ISTANBUL TECHNICAL UNIVERSITY ★ GRADUATE SCHOOL

ROCKET ENGINE ALTITUDE TEST FACILITY DESIGN AND 1D ALTITUDE SIMULATION OF LOX/LH2 PROPELLANT ROCKET ENGINE

M.Sc. THESIS

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ISTANBUL TEKNİK ÜNİVERSİTESİ ★ LİSANSÜSTÜ EĞİTİM ENSTİTÜSÜ

ROKET İRTİFA TEST DÜZENEĞİ TASARIMI VE LOX/LH2 YAKITLI ROKET MOTORUNUN 1D İRTİFA SİMÜLASYONU

YÜKSEK LİSANS TEZİ

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İsmail ÖZCAN, a M.Sc. student of ITU Graduate School of Science Engineering and Technology 511201101, successfully defended the thesis entitled "ROCKET ENGINE ALTITUDE TEST FACILITY DESIGN AND 1D ALTITUDE SIMULATION OF LOX/LH2 PROPELLANT ROCKET ENGINE", which he prepared after fulfilling the requirements specified in the associated legislations, before the jury whose signatures are below.

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To my family,

FOREWORD

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ABBREVIATIONS

AD	:	Anno Domini
ASME	:	American Society of Mechanical Engineers
CCLD	:	Constant Current Line Drive
CFD	:	Computational Fluid Dynamics
CONOPS	:	Concepts of Operations
DAQ	:	Data Acquisition
DLR	:	German Aerospace Center
ECU	:	Engine Control Unit
EN	:	European Standard
ESA	:	European Space Agency
JAXA	:	Japan Aerospace Exploration Agency
JP-10	:	Jet Propellant-10
JP-8	:	Jet Propellant-8
K	:	Kelvin
kN	:	Kilo Newton
LH2	:	Liquid Hydrogen
LOX	:	Liquid Oxygen
mbarA	:	millibar Absolute
MIL-STD	:	Military Standard
N	:	Newton
NASA	:	National Aeronautics and Space Administration
RP-1	:	Refined Petroleum-1
RP-2	:	Refined Petroleum-2
RPM	:	Revolution per Minute
RTD	:	Resistance Temperature Detector
SAE	:	Society of Automobile Engineers
SSME	:	Space Shuttle Main Engine

SYMBOLS

$\boldsymbol{\varepsilon}$	:	Expansion Ratio
$\boldsymbol{\delta\dot{W}}$	:	Work in Unit Time
$\boldsymbol{\delta\dot{Q}}$	:	Heat Exchange
$\boldsymbol{\gamma}$	:	Specific Heat Ratio
$\boldsymbol{m}$	:	Mass
$\boldsymbol{a}$	:	Acceleration
$\boldsymbol{M}$	:	Mach Number
$\boldsymbol{\upsilon_j}$	:	Stoichiometric Coefficient
$\boldsymbol{\zeta}$	:	Minimum Singular Pressure Drop Factor
$\boldsymbol{w}$	:	Pressure Ratio in Driving Side
$\boldsymbol{u}$	:	Velocity through the axis
$\boldsymbol{U}$	:	Internal Energy
$\boldsymbol{t}$	:	Time
R	:	Gas Constant per Unit Weight
r	:	Mixture Ratio
$\boldsymbol{R}$	:	Rate Of the Global Reactions
P	:	Ratio in Suction Side
$\mathbf{n_i''}$	:	Mole of Products
$\mathbf{n_i'}$	:	Mole of Reactants
$\boldsymbol{N}$	:	Pressure Ratio
$\boldsymbol{M_j}$	:	Molar Mass for the Species
$\boldsymbol{Kv}$	:	Maximum Flow Capacity Coefficient
$\boldsymbol{h}$	:	Specific Enthalpy
$\boldsymbol{Cv}$	:	Maximum Flow Capacity Coefficient
$\boldsymbol{Cq}$	:	Maximum Flow Coefficient
$\mathbf{A}\boldsymbol{i}$	:	General Chemical Species
$\boldsymbol{\mu}$	:	Mass Flowrate Ratio

$\Delta H(T)$	:	Heat of Reaction
v_{exit}	:	Exit Velocity
v_2	:	Exhaust Velocity
m_o	:	Oxidizer mass flowrate
m_f	:	Fuel mass flowrate
$\dot{m}$	:	Mass Flowrate
g_0	:	Standard sea-level Gravitational Acceleration
T_c	:	Chamber Temperature
P_{exit}	:	Exit Pressure
$P_{environment}$	:	Environment Pressure
P_c	:	Chamber Pressure
$P_{ambient}$	:	Ambient Pressure
I_{sp}	:	Specific Impulse
A_{throat}	:	Throat Area
A_{nozzle}	:	Nozzle Area
$\dot{x}_{out}$	:	Outlet Mass Fractions
$\dot{x}_{in}$	:	Inlet Mass Fractions
$\dot{m}_{out}$	:	Outlet Mass Flowrate
$\dot{m}_{in}$	:	Inlet Mass Flowrate
P_{sep}	:	Static pressure at flow separation point
P_{atm}	:	Atmospheric pressure
M_{sep}	:	Separation point Mach number
P_0	:	Static pressure at extremity of the nozzle
T_0	:	Static temperature at extremity of the nozzle
A_1	:	Cross-sectional area of the point of interest
M_1	:	Mach number of point of interest
A_T	:	Cross-sectional area of the throat
M_T	:	Mach number of the throat
T_{sep}	:	Separation point temperature
v_{sep}	:	Velocity at separation point
temp2	:	Hot gases temperature at the nozzle inlet
ρ_{sep}	:	Separation point density

A_{sep}	:	Cross-sectional area of separation point
dm1	:	Mass flowrate
P	:	Static Pressure fluid
ρ	:	Density of fluid
V	:	Velocity of fluid
g	:	Gravitational Acceleration
h	:	Elevation

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ROCKET ENGINE ALTITUDE TEST FACILITY DESIGN AND 1D ALTITUDE SIMULATION OF LOX/LH2 PROPELLANT ROCKET ENGINE

SUMMARY

The interest in space and development studies started in B.C. for humanity. Astronomical observations began to understand the dynamics between planets and stars. Reaching space and flying around the world have been the most considered areas over the years. Before the 20th century, theoretical and experimental studies were conducted; however, scientific research increased after the 20th century. Tsiolkovsky and Goddard contributed to rocketry both theoretically and experimentally. During World War II, the Germans developed V2 rockets, which were the predecessors of space rockets in the following years. The Saturn V rocket was launched in 1967, and it remained the most powerful rocket for humankind up to the 21st century before Starship Heavy.

In today's world, the development of liquid-fueled rocket engines is progressing rapidly. Many companies and agencies are designing launch vehicles, payloads, and rocket engines all over the world. Today, NASA, ESA, SpaceX, JAXA, and others have numerous studies on rocket engines and their components. Nowadays, research and development of rocket engines have reached the most powerful rocket engine, the Raptor Engine by SpaceX. Test system designs and test facility setups are constructed even for the most powerful rocket engine designs. Various test setups and infrastructures are being established for testing numerous subcomponents and equipment as part of pre-flight verification for liquid-fueled rocket engines. It is worth noting that rocket engines with turbopump feed systems undergo ground tests and altitude simulation tests before launch in many industrial and academic studies.

Today, there is ongoing development of various test infrastructures for the feed systems of turbopumps, including valve characterization test setups, cryogenic valve test setups, injector test setups, turbine test setups, pump test setups for oxidizer and fuel, leak test setups, gas generator and combustion chamber test setups, and more. Test setups for altitude testing of first-stage and second-stage rocket engines have been developed and are used to verify thrust and turbopump efficiency during the rocket's flight. Altitude test setups for rocket engines are available in various countries such as France, the United Kingdom, the United States, Germany, and South Korea, and they are used for altitude tests of rocket engines with different thrust capacities.

Due to various factors such as different types of oxidizer and fuel feed systems, thrust capacities, rocket engine sizes, exit temperatures, and flow rates, unique system developments have been made and continue to be pursued. The primary sub-components of the rocket altitude test setup, as described in the literature, include motor feed tanks, rocket engines, vacuum chambers, diffusers, cooling water injectors, deflectors, ejector structures, and condensing equipment. In addition to these

components, instruments such as pressure sensors, flow meters, temperature sensors, accelerometers, and load cells are used during performance testing to measure and verify performance values.

Design considerations for the 1D modeling of the rocket engine followed established processes from the literature in this study. The most common programs for 1D modeling are EcosimPro and Simcenter Amesim. Simcenter Amesim was preferred for studying 1D system modeling for both the rocket engine and the altitude test facility design. In the first step, a 1D model of the rocket engine was created. A 1D model of the LOX/LH2 liquid-fueled Expander Cycle Rocket Engine VINCI’s full model was developed, and a simplification method was used to reduce system unknowns. For the simplification method, the boundary conditions at the system inlet were kept the same, while the outlet conditions of the pump and turbine became the new inlet conditions for the thrust chamber. The pressurized oxidizer and fuel conditions remained stable with the addition of orifices, and the performance of the combustion chamber and nozzle was monitored. After completing the rocket engine design, the design of the altitude test facility was undertaken. Piping, chambers, divergent-convergent structures, steam ejectors, and steam feeding boundary conditions were added to the system to control the results. A vacuum system model was created to observe the VINCI rocket engine's performance under vacuum conditions lower than 50 mbarA.

In this thesis study, the performance of the VINCI rocket engine and the altitude test facility were considered. The desired version of the VINCI was the 180 kN thrust in vacuum. It was observed that the thrust levels and performance data produced by the VINCI rocket engine were satisfactory during simulation. The thrust of the 1D modeled VINCI engine was found to be nearly 178 kN. The combustion chamber pressure manufacturer data was given as 60 barA, and the 1D model of the VINCI simulation result was found to be 59.78 barA. The feeding lines of the fuel and oxidizer mass flow rates were given as 5.59 kg/s and 34.11 kg/s according to the manufacturer. Simulation results for the fuel and oxidizer were found to be 5.5 kg/s and 34.31 kg/s, respectively. The manufacturer's specific impulse data was given as 457.2 s, and the 1D model of the VINCI simulation result was found to be 456.17 s. The main consideration for the vacuum conditions was set at 50 mbarA, and the 1D simulation results were found to be 42.93 mbarA with an increased steam inlet mass flow rate of the second-stage steam ejector. For the first stage of the steam ejector, literature information was followed, and a 110 kg/s steam condition was satisfied. For the second ejector, 141 kg/s of steam was supplied, creating a vacuum of 42.93 mbarA. The manufacturer's second-stage steam mass flow rate was given as 118 kg/s, suggesting that better vacuum conditions could be achieved according to the simulation results. Future work includes creating a digital twin with the 1D model, conducting 3D model-based CFD analysis for the altitude test facility design, comparing real-time test data between altitude test facilities and 1D simulation models, and running performance scenarios in the 1D altitude test model before conducting tests.

ROKET İRTİFA TEST DÜZENEĞİ TASARIMI VE LOX/LH2 YAKITLI ROKET MOTORUNUN 1D İRTİFA SIMÜLASYONU

ÖZET

Havacılık endüstrisi ve havacılık araştırma ve geliştirme çalışmaları, insanlık tarihi açısından milattan önce başlamıştır. Gezegenler ve yıldızlar arasındaki dinamikleri anlamak amacıyla astronomik gözlemler yapılmaya başlanmıştır. Uzaya ulaşmak ve dünya etrafında uçmak, yıllar boyunca en çok üzerinde durulan alan olmuştur. 20. yüzyıldan önce teorik ve deneysel çalışmalar yapılmışsa da bilimsel araştırmalar 20. yüzyıldan sonra artmıştır. Tsiolkovsky ve Goddard, roketçilik alanında teorik ve deneysel olarak katkıda bulunmuşlardır. II. Dünya Savaşı sırasında Almanlar, V2 roketlerini geliştirmiştir ve bu roketler, sonraki yıllarda uzay roketlerinin atası olmuştur. Saturn V roketi, 1967'de fırlatılmıştır ve bu roket, 21. yüzyıla kadar insanlık için en güçlü roket olmuştur.Günümüz dünyasında, uzay çağında gelişmeler büyük bir ivmelenme ile gelişimini sürdürmektedir. On dokuzuncu yılın sonlarında ve yirminci yüzyılın başlangıcından itibaren uzaya ulaşmak için kullanılan ve ilk tercih edilen fırlatma araçları olan roket teknolojisi geçmişten bugüne gelişimini sürdürmektedir. Kimyasal reaksiyonların kontrollü olması amacıyla özellikle ikinci dünya savaşı zamanlarında gelişimi ivmelenen sıvı yakıtlı roket motorları günümüzde de aktif olarak kullanılmaktadır. Sıvı yakıtlı roket motorları literatürde üç farklı başlık altında incelenmekle birlikte üç farklı tip roket motoru günümüzde farklı metotlar da kullanılarak yeni modeller doğurmuştur. Sıvı yakıtlı roket motorları genişleme çevrimi, gaz jeneratörü çevrimi ve aşamalı yanma çevrimi olarak üç farklı başlık altında incelenmektedir. Farklı görevleri yerine getirmek amacıyla geçmişten günümüze Vulcain, VINCI, RL10 gibi farklı roket motorları fırlatma araçlarına güç vermek amacıyla kullanılmıştır. Genişleme çevrimine sahip olan roket motorları günümüzde aktif olarak yaygın kullanıma sahip olmamakla birlikte birçok zorluk içermektedir. Özellikle sıvı hidrojen yakıtı ile yapılan sıvı yakıtlı roket motorlarında depolama tankı tasarımı, lülenin yüksek sıcaklıklara karşı korunumu için yakıtın bir miktarının kontrol vanalarıyla bir miktarının soğutma için ayrılması gibi kontrol mekanizmalarının kritikliği sebebiyle gaz jeneratörü çevrimli ve aşamalı yanma çevrimi roket motorları daha yaygın kullanılmaktadır.

Sıvı yakıtlı roket motorlarının fırlatmadan önce çeşitli alt bileşenlerin ve ekipmanların test edilmesi için çeşitli test düzenekleri ve altyapılar kurulmaktadır. Turbopompalı besleme sistemine sahip roket motorları, birçok endüstriyel ve akademik çalışma ile fırlatmadan önce yer testlerine ve irtifa simülasyon testlerine tabi tutulmaktadır. Günümüzde, turbopompa besleme sistemlerinin çeşitli test altyapıları geliştirilmektedir; bu altyapılar arasında valf karakterizasyon test düzenekleri, kriyojenik valf test düzenekleri, enjektör test düzenekleri, türbin test düzenekleri, oksitleyici ve yakıt için pompa test düzenekleri, sızdırmazlık test düzenekleri, gaz üreteci ve yanma odası test düzenekleri, ısı eşanjörü test düzenekleri ve farklı

komponentlerin testlerini gerçekleştirmek için benzer test düzenekleri tasarlanmaktadır. Roketin uçuşu sırasında üretilen itki ve turbopompa verimliliğinin doğrulandığı bilinmektedir. Roket motorlarının irtifa test düzenekleri Fransa, Birleşik Krallık, Amerika Birleşik Devletleri, Almanya ve Güney Kore gibi çeşitli ülkelerde mevcuttur ve farklı itki kapasitelerine sahip roket motorlarının irtifa testleri için kullanılmaktadır. Farklı oksitleyici ve yakıt besleme sistemleri, itki kapasiteleri, roket motoru boyutları, çıkış sıcaklıkları ve akış hızları gibi çeşitli faktörlerden dolayı, benzersiz sistem geliştirmeleri yapılmış ve yapılmaya devam edilmektedir. Literatür örneklerinde roket irtifa test düzeneklerinin başlıca alt bileşenleri, motor besleme tankları, roket motorları, vakum odaları, difüzörler, soğutma suyu enjektörleri, yansıtıcılar, ejektör yapıları ve kondenzasyon ekipmanlarını içermektedir. Motor besleme tankları özellikle yer testlerinin kullanımı açısından büyük bir öneme sahip olmasa da kriyojenik sıcaklıklarda depolama yapılmasından ötürü özeli bir tasarım gerekmektedir. Çift cidarlı tank ve vakum izoleli boşluklar ile birlikte iletim ve konveksiyon temelli ısı kayıpları önemli ölçüde azaltılmaktadır. Tank malzemeleri için birçok malzeme kullanılmakla birlikte tankların basınçlı olup olmadığına bağlı olarak paslanmaz çelik, alüminyum, kompozit ve farklı çelik alaşımları genel olarak kullanılmaktadır. Roket motorlarının yerleşimi ve doğru ölçüm yapılması için bremzeler kurularak farklı performans testlerine hitap etme özelliğine sahip yerleşim şaseleri yaygın olarak kullanılmaktadır. İrtifa test düzeneklerinde kullanılan buhar ejektörleri ise vakum ejektörleri olarak endüstride ve havacılıkta yaygın olarak kullanılmaktadır. Özellikle hatlarda biriken su, yakıt ve yağ gibi zamanla korozyona sebep verebilecek maddelerin hatlardan tamamen çıkarılması için de kullanımı söz konusudur. Borulama hatlarında ise geçen akışkan özelliklerinin önemi söz konusudur. Kriyojenik sıcaklıklarda genellikle paslanmaz serisi boru malzemeleri yaygın şekilde kullanılmaktadır. Borulamalardan da kayıp yaşanmaması adına boru tasarımları vakum izoleli olarak yapılmaktadır. Kondensatörler ise genellikle buhar ejektörlerinin kullandığı yüksek miktardaki suyu tekrar kullanmak için kullanılmaktadır. Tekrar kullanılacak olan suyun içerisinde yer alan yakıt ve oksitleyici sebebiyle çeşitli filtreler kullanılmakla birlikte kullanılan oksitleyici ve yakıtın seçimine uygunluk önemli bir kriter olarak görülmektedir. Bu bileşenlere ek olarak, basınç sensörleri, akış ölçerler, sıcaklık sensörleri, ivmeölçerler ve yük hücreleri gibi enstrümanların performans testlerinde kullanıldığı ve performans değerlerini ölçmek ve doğrulamak için kullanıldığı bilinmektedir. Seçilen enstrümantasyon ve test düzeneği ekipmanlar ilgili motorların tasarımı için seçilmekle birlikte yüksek sıcaklık, düşük sıcaklık gibi koşullar göz önünde bulundurularak ilgili arayüzler tasarlanmaktadır.

Roket motorlarının modellenmesi konusunda farklı kaynaklar kullanılarak yetkin programlar aracılığıyla ilgili görevlere yönelik tasarımlar yapılmaktadır. Roket motorları için kullanılan turbopompaların tüm komponentlerinin geliştirilme süreçleri farklı analiz programları ile yürütülmektedir. Roket motorlarının tasarımında pompa performanslarını incelemek amacıyla CFTurbo ve benzeri programlar aracılığıyla çalışmalar yürütülmektedir. Türbin performanslarını incelemek ve tasarımını yapmak amacıyla Concepts NREC ve benzeri programlar aracılığı ile tasarımlar yapılmaktadır. Özellikle sızdırmazlık elemanları için mil, türbin ve pompalar arasındaki ikincil akış çalışmaları için GFSSP, Flownex ve özgün yazılımlar sıklıkla kullanılmaktadır. Turbopompa elemanının ve roket motorunun modellenmesi için günümüzde ESA tarafından literatürde sıklıkla kullanılan EcosimPro yazılımı kullanılmaktadır. EcosimPro yazılımına ek olarak Flownex ve Simcenter Amesim yazılımları günümüzde roket motorlarının 1D modellenmesinde sıklıkla kullanılmaktadır.

EcosimPro ve Simcenter Amesim programlarında RL10A-3-3A roket motorunun modellenmesiyle birlikte endüstride birçok firma bu iki yazılımı kullanmaktadır.

Roket motorunun 1D modelleme tasarım düşünceleri literatürdeki yerleşik süreçleri göz önünde bulundurularak çalışmalar gerçekleştirilmiştir.. 1D modelleme için en yaygın programlar EcosimPro ve Simcenter Amesim'dir. Hem roket motoru hem de irtifa test tesisi tasarımının 1D sistem modellemesi çalışmalarında Simcenter Amesim tercih edilmiştir. İlk adımda, roket motorunun 1D modeli oluşturulmuştur. LOX/LH2 sıvı yakıtlı Genişleme Çevrimli Roket Motoru olan VINCI'nin tam modelinin 1D modeli geliştirilmiş ve sistem bilinmeyenlerini azaltmak için bir basitleştirme yöntemi kullanılmıştır. Basitleştirme yöntemi için, sistem girişindeki sınır koşulları aynı tutulmuş, pompa ve türbinin çıkış koşulları ise itki odasının yeni giriş koşulları olmuştur. Basınçlı oksitleyici ve yakıt koşulları orifis eklemeleri ile sabit tutulmuş ve yanma odası ile nozülün performansı izlenmiştir. Roket motoru tasarımının tamamlanmasının ardından, irtifa test tesisi tasarımına başlanmıştır. Sistemin sonuçlarını kontrol etmek için borular, odalar, divergent-konvergent yapılar, buhar ejektörleri ve buhar besleme sınır koşulları sisteme eklenmiştir. VINCI roket motorunun performansını 50 mbarA'nın altındaki vakum koşullarında gözlemlemek için bir vakum sistem modeli oluşturulmuştur.

Bu tez çalışmasında, VINCI roket motorunun ve irtifa test tesisinin performansı incelenmiştir. Vakumda 180 kN itki gücü sağlayan VINCI versiyonu hedeflenmiştir. VINCI roket motoru tarafından üretilen itki seviyeleri ve performans verilerinin simülasyon sırasında tatmin edici olduğu gözlemlenmiştir. 1D modellenen VINCI motorunun itki gücü yaklaşık 178 kN olarak bulunmuştur. Yanma odası basıncı üretici verisi 60 barA olarak verilmiş olup, VINCI simülasyon sonucunun 1D modeli 59.78 barA olarak bulunmuştur. Yakıt ve oksitleyici kütle akış oranlarının besleme hatları, üreticiye göre 5.59 kg/s ve 34.11 kg/s olarak verilmiştir. Yakıt ve oksitleyici simülasyon sonuçları ise sırasıyla 5.5 kg/s ve 34.31 kg/s olarak bulunmuştur. Üretici tarafından verilen spesifik itki verisi 457.2 s olarak verilmiş olup, VINCI simülasyon sonucunun 1D modeli 456.17 s olarak bulunmuştur. Vakum koşulları için ana kriter 50 mbarA olarak belirlenmiş olup, ikinci aşama buhar ejektörünün artırılmış buhar giriş kütle akış hızı ile 1D simülasyon sonuçları 42.93 mbarA olarak bulunmuştur. İlk aşama buhar ejektörü için literatür bilgisi takip edilmiş ve 110 kg/s buhar koşulu sağlanmıştır. İkinci ejektör için 141 kg/s buhar sağlanmış ve 42.93 mbarA vakum oluşturulmuştur. Üreticinin ikinci aşama buhar kütle akış hızı 118 kg/s olarak verilmiş olup, simülasyon sonuçlarına göre daha iyi vakum koşullarının sağlanabileceği öngörülmektedir. Gelecek çalışmalar arasında 1D model ile dijital ikiz oluşturulması, irtifa test tesisi tasarımı için 3D model tabanlı CFD analizi, irtifa test tesisi ve 1D simülasyon modelleri arasında gerçek zamanlı test verisi karşılaştırması, ve testlerden önce 1D irtifa test modelinde performans senaryolarının çalıştırılması bulunmaktadır.

1. INTRODUCTION

From the early 1950s to today, rockets have become one of the essential aerial vehicles for transferring satellites, probes, rovers, robots, and humans. Even though mission failures have resulted in losses, including deaths, the development of rocket technology has not diminished its impact on the space industry. The history of rocketry began in China around AD 970. These early rockets were the ancestors of simple rockets later developed by the Mongols for military use in the 1300s. After encountering rockets, Western civilizations developed modern systems. Modern rocket technology started with Tsiolkovsky, who formulated the rocket equation and designed the architecture for a more powerful thrust system than the powder-based systems. This design, called the liquid propellant rocket engine, was intended for space travel. After Tsiolkovsky's developments were published, Herman Oberth published his doctoral thesis on liquid fuel-based engines using alcohol and LOX in 1923. Robert Goddard was the first person to experimentally fire liquid propellant engines. He solved many problems experimentally, such as storage, combustion, and thrust system development. The first practical liquid propellant engine, the A4 (known as V-2), was successfully developed by Wernher von Braun, based on concepts from Goddard's studies. After the Second World War, the U.S. and Soviet Union started their space programs, accelerating the development of modern liquid propellant rocket engines. The first programs for modern liquid propellant engines were R7 and Atlas. Over the years, engines like Titan, Saturn, and RL10 were developed, becoming the ancestors of new modern liquid propellant rocket engines [1] [2] [3] [4]. Today, many liquid propellant rocket engines are actively developed and used in space programs, including Merlin, RD107, Raptor, LE-X, and VINCI.

Today, solid, hybrid, and liquid propellant rocket engines are frequently used in modern space programs for different missions. For liquid propellant rocket engines, there are two types of feeding systems: pressurized systems and turbopump systems. Both feeding systems are challenging and must be modeled for rocket engines. After selecting the proper feeding system, the thrust chamber is designed and combined with the feeding system. Liquid propellant engines have many sub-components, including valves, pipes, propellant tanks, injectors, nozzles, gas generators, combustion chambers, pneumatic controllers, and hydraulic controllers. All sub-components and main components are simulated to verify the engine design. 1D simulations can be conducted with different programs, optimizing the rocket engine design with all components. Depending on the purpose, combustion chamber components, fluid components, nozzle components, pump components, turbine components, mechanical interface components, gas generator components, piping components, and other components are developed via 1D design programs.

Test facilities and systems are developed and used to verify the sub-component and component design of liquid propellant engines. Various test systems are used, including hydraulic and pneumatic test rigs for valves, pump test rigs for pump performance, turbine test rigs for turbine performance, cryogenic-pressurized tank test rigs for propellant tanks, sealing elements test rigs, hot firing test rigs, stage test rigs, and altitude test rigs.

The novelty of this study is combining a liquid propellant rocket engine with an altitude test facility in a 1D simulation program. For the 1D rocket engine simulations, reference vacuum conditions are defined for the engine, and performance is analyzed in the simulation program. In real life, altitude test facilities are used to verify rocket engine performance, and different conditions can occur. Vacuum conditions can be simulated in various scenarios with an altitude test facility, and rocket engine design can be optimized under different conditions before flight. The selection and design of altitude test facility components can be optimized with 1D simulation, creating a digital twin of test conditions to reduce the costs of engine tests.

1.1 Purpose and Importance of Test Systems

The space technology market has been growing over the years, with countries around the world demanding various payloads in orbit around the Earth. The reduction of costs for payloads and launch vehicles has led to increased investments in space technology, as seen in Figure 1.1.

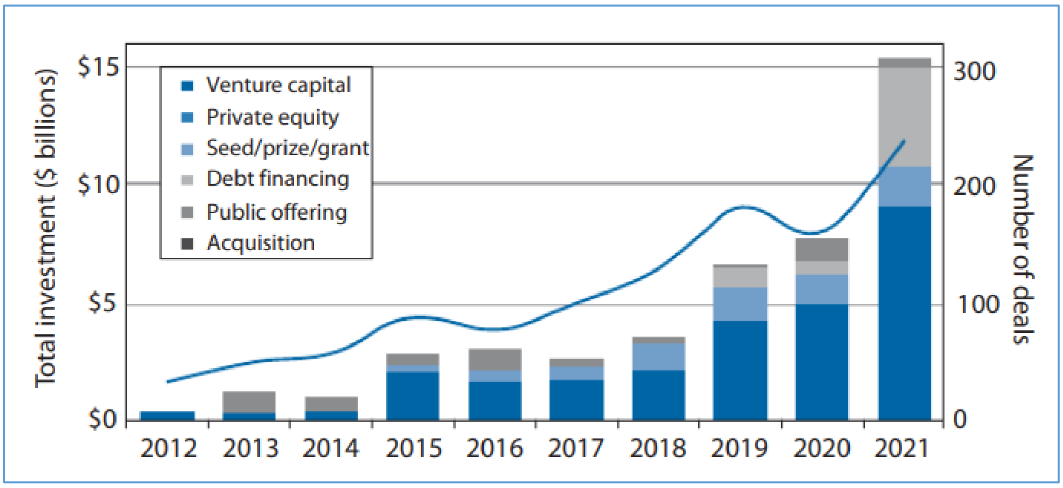


Figure 1.1: Space Related Companies Investments [5]

Research and development in space technology are supported by investments in test systems, optimized software, and academic research. In the aerospace industry, there are numerous test system examples where specific components for aerial vehicles can be directly tested. Aerial vehicle engine tests are conducted under two conditions: sea level tests and altitude tests. In sea level tests, the test rig for the engines is under standard conditions that humans experience daily. Engine behavior is examined by test engineers, and the collected data is used to optimize software for mechanical analysis, CFD analysis, ECU analysis, and other programs. In altitude tests, the test rig for the engines is under low pressure or vacuum conditions, simulating high altitude and space conditions. Similar procedures provide engineers with information to develop optimized aerial vehicle engines. Rocket engine behavior is examined based on similar aerial vehicle engine test procedures. Generally, most launch vehicles have more than one stage to reach Earth orbit, requiring rocket engines to be tested under different conditions before flight.

For the first stage of launch vehicles, the rocket engine operates under atmospheric conditions, similar to sea level conditions, progressing to low pressure during flight. After the first stage, the launch vehicle enters the second phase of flight, where the second stage rocket engine is fired. The second stages of rocket engines typically operate under vacuum conditions, simulating the presence of space. Rocket engines are usually used only once during flight, and errors can result in failure and explosions. Companies and agencies conduct extensive tests on rocket engines before flight. Test systems are designed by engineers following standards like MIL-STD, SAE, ASME, and EN. These standards are set by governing bodies, countries, and agencies to ensure safety during tests. Performance tests for the entire engine, main components, and sub-components are conducted using test rigs. Test rigs provide critical information for the tested component, measuring acceleration, pressure, temperature, mass flow rate, dew point, force, displacement, torque, power, RPM, flame detection, and other measurable values via sensors or probes.

As this thesis concentrates on the altitude test system design and 1D simulation of rocket engines, test system components and rocket engine specifications are widely used.

1.2 Measuring the Performance

Altitude test facilities are used for testing rocket engine second stage before the flight of the launch vehicles. The common measured values for the altitude test facilities are exhaust gas temperature, pressure around the rocket engine, mass flowrates of the oxidizer and fuels, characteristic velocity of the nozzle exhaust gas, valve responses, acceleration of rocket engine and rocket engine's generated thrust. Rocket engine includes stages in its structure and all stages have instruments to check problem between stages.

Sensors to measure given common measured values have to be selected carefully that accuracy of the measured value importance. There are also DAQ systems that collects and process signals from the sensors. Sensor assemblage on the measurement region is an important case that extremely cold and extremely hot regions could result in failure. Failure can be prevented by sensor's materials, placing far away location via tubes, cooling-heating around the sensors and proper sensor selections Pressure measurements are generally measurement by pressure transmitters. Typical simple pressure transmitter can be seen in Figure 1.2.

Figure 1.2: Pressure Transmitter [6]

Temperature measurements are generally measured by thermocouples and RTDs. Typical simple thermocouple can be seen in Figure 1.3.

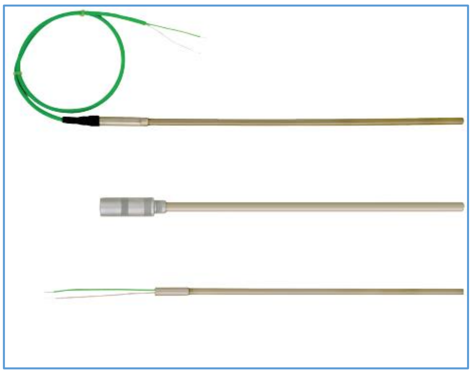


Figure 1.3: Mineral Insulated Thermocouple [7]

Mass flowrate measurements are generally measured by flowmeters. Flowmeters have various types for different fluid types. For the general applications, coriolis and turbine type flowmeters are widely used. Typical coriolis flowmeter can be seen in Figure 1.4.

Figure 1.4: Coriolis Type Flowmeter [8]

Velocity measurements are very hard to measure for exceeding speed of sound. Total pressure probes are widely to measure total pressure and measured values shows after shock interactions. There are some correlation methods are valid for bow shock and oblique shock interactions. Taylor-Maccoll is typical known correlation method for conic probes [9].

Acceleration that can be stated as vibration measurements are generally measured by accelerometers. Accelerometers have various types for different purposes and typical simple CCLD type accelerometer can be seen in Figure 1.5.

Figure 1.5: Accelerometer [10]

Thrust measurements are generally measured by loadcells. Loadcells have to get force from plates and convert this reaction to force by strain gages. Loadcells have various types for customers and typical S type loadcell can be seen in Figure 1.6.

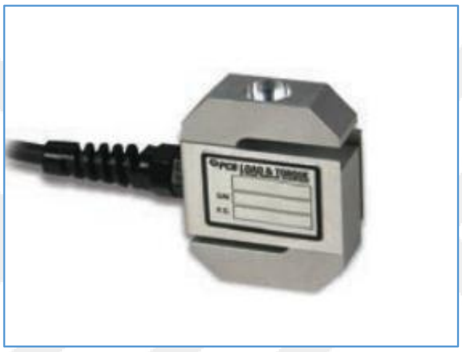


Figure 1.6: Loadcell [11]

1.3 Literature Overview

For the altitude test facilities there are many different design methodologies for different rocket engines. German based test facility is seen Figure 1.7

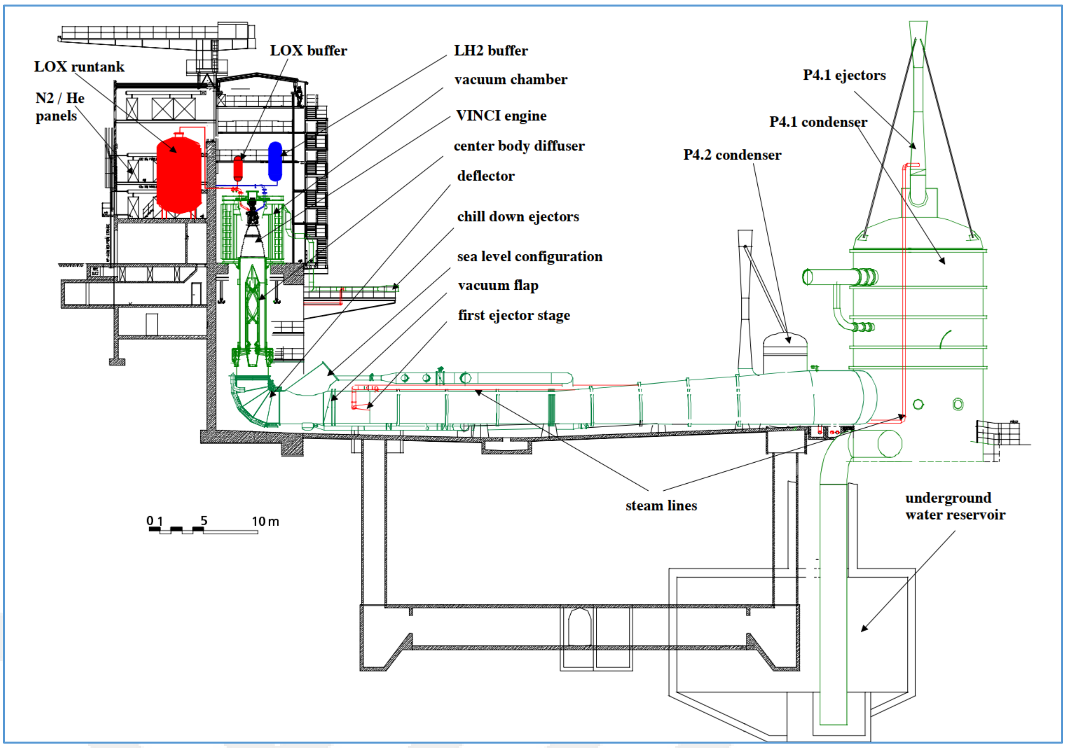


Figure 1.7: P4.1 Test Facility [12]

P4.1 test facility is used for VINCI rocket engine that is designed by DLR. In this test facility different versions of the altitude test facilities are available according to related studies. As it is seen Figure 1.7, there are a number of components that are related with engine performance. LOX consumption of the VINCI is considered huge and run tank volume is seen higher than buffer tank. LH2 consumption of the VINCI rocket engine is considered lower that run tank is not added for design. Vacuum chamber is used to create vacuum around the engine and vacuum is created by the ejectors. Center body diffuser is used to control of the velocity which is generated by the engine. There are other important considerations which are heat loads on component and dimension limitations on the center body design [12] [13]. Center body diffuser has cooling channels and water feeding systems on its design as it is seen in Figure 1.8.

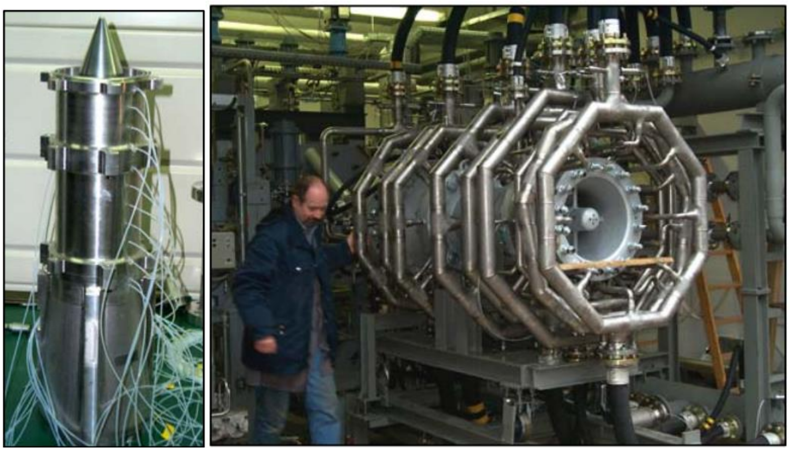


Figure 1.8: Diffuser for Cold and Hot Tests [13]

There are a number studies were held in USA. NASA conducts altitude test simulation bench designs in U.S and construction documents were shared. As it is seen in German based altitude test facility design, NASA's altitude test simulation system includes diffuser and ejector systems. This test stand is called A3 and it is seen in Figure 1.9. [14]

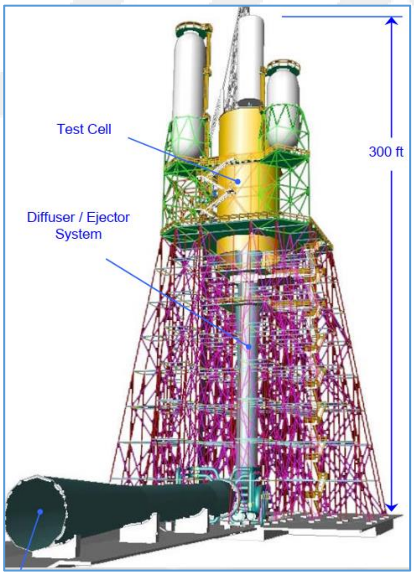


Figure 1.9: A3 Test Stand for J2-X Engine [14]

3D model of the A3 test stand is detailed for the construction of altitude test bench. In this test bench, feeding run tanks for LOX LH2, feeding lines, 1st stage ejectors, diffuser, diffuser cooling feeding, 2nd stage ejectors, steam generators and outlet diffuser to atmosphere is seen in Figure 1.10 and Figure 1.11

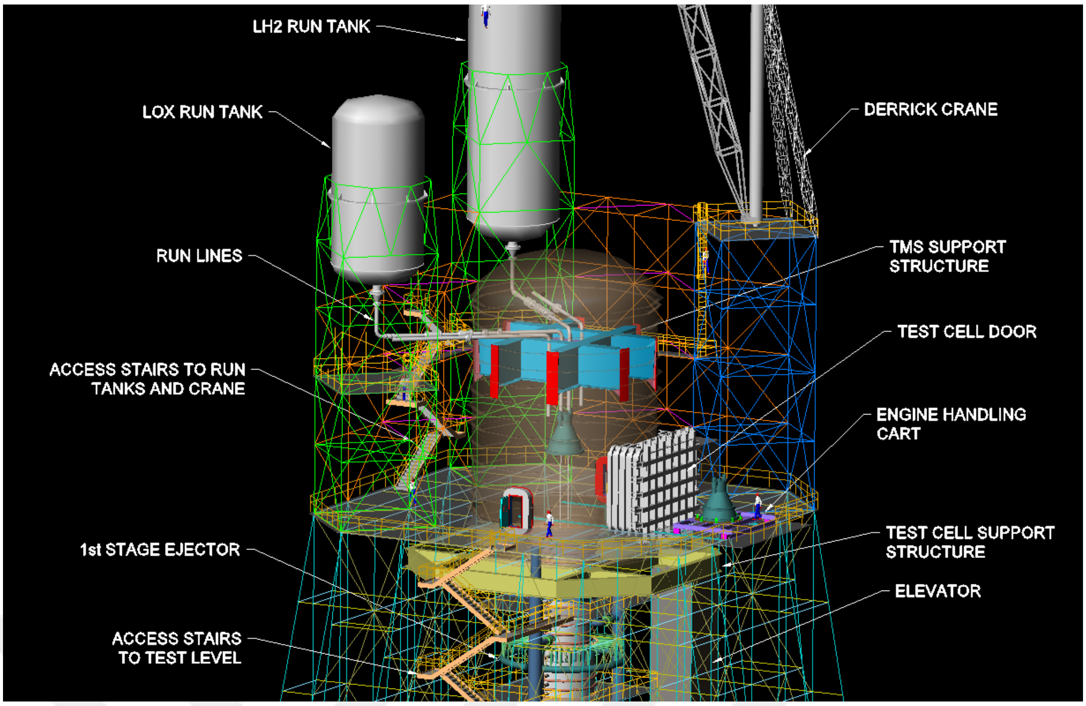


Figure 1.10: A3 Test Stand Detailed Construction [14]

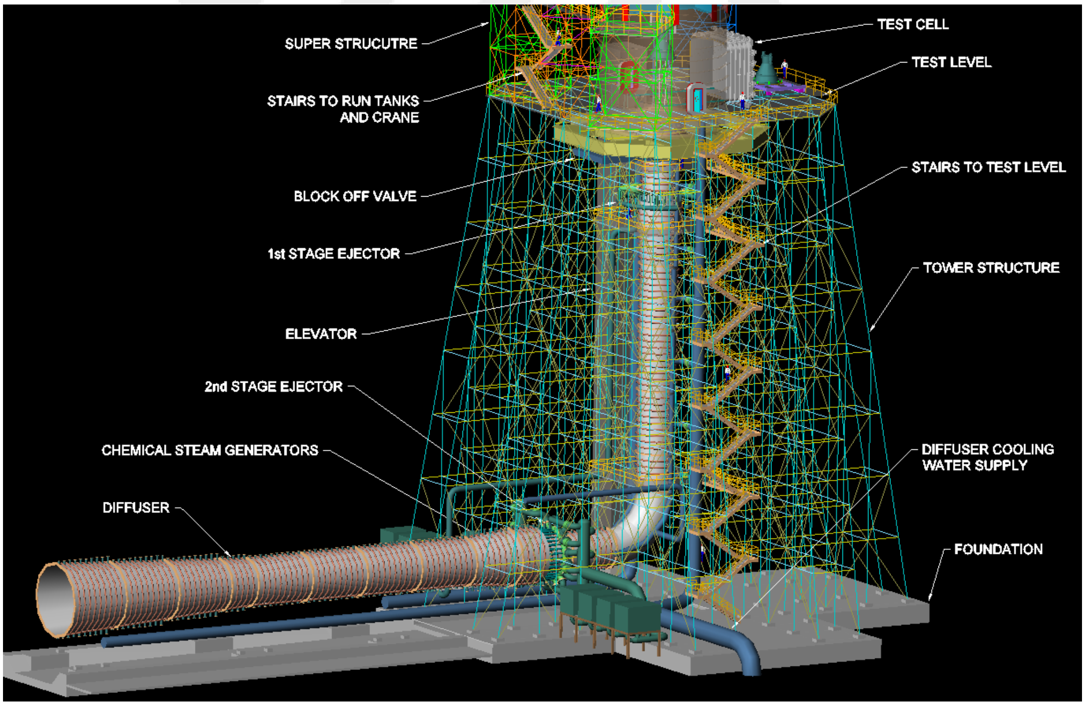


Figure 1.11: A3 Test Stand Detailed Construction [14]

Titan III's rocket engine is also tested by US. The test cell for the rocket engine that is called J-3 was developed and used for the altitude simulations. In this test cell, various altitude test simulation is conducted between 100,000 – 200,000 fts that is nearly 30 km to 60 km altitude test values in metric units. Vacuum is created via steam ejectors and without steam ejectors variation of the test cell pressure can be problem in here. J-3 test stand design clarify that spray cooling water is important to remove big amount of thermal energy of exhaust gases. Test cell construction of the J-3 is seen in Figure 1.12.

Figure 1.12: J-3 Test Stand [15]

Developed version of the J-3 test stand was used for LR 91, RL10 and various rocket engines in NASA’s test sites that is called J-4. Similar design of the J-3 and A3 test stand construction is seen as it is expected. Developed version of the J-4 test cell is seen in Figure 1.13 [16].

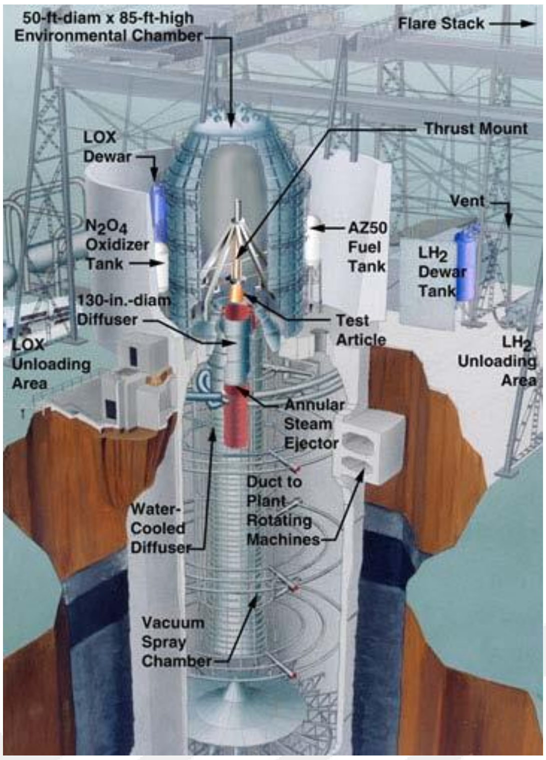


Figure 1.13: J-4 Test Stand [16]

Another test stand is seen in Westcott Test Facility in United Kingdom. In this test system approximate vacuum condition is satisfied between 35-50 mbarA pressure. Tested engines are small and thrust capability of engines are 1-2 kN in this study. Design and model of the test stand is seen in Figure 1.14

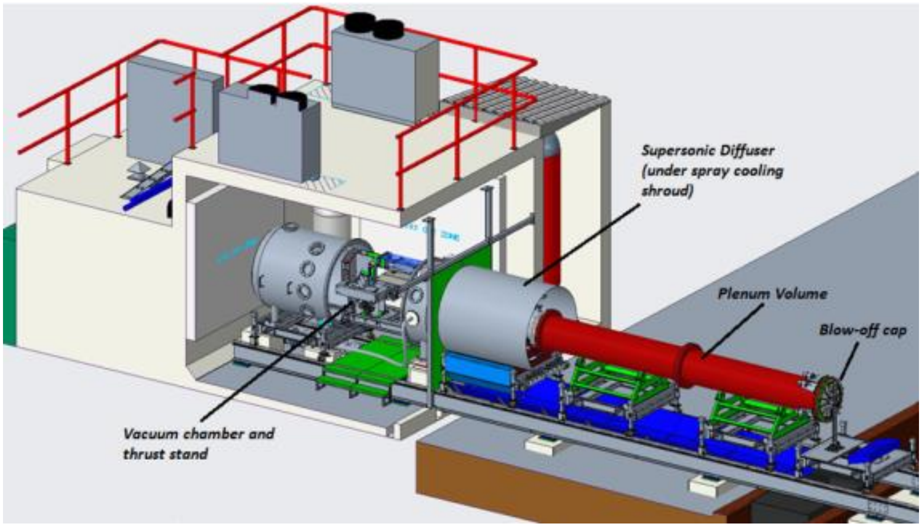


Figure 1.14: J3 Test Stand in UK [17]

Korean rocket engine test facilities also include altitude test facilities and give some information about altitude test facility design. Small class of liquid rocket engine which is stated as 500N is tested in altitude test facility. Simulation condition for the vacuum is 27 mbarA for this rocket and test stand design can be seen in Figure 1.15.

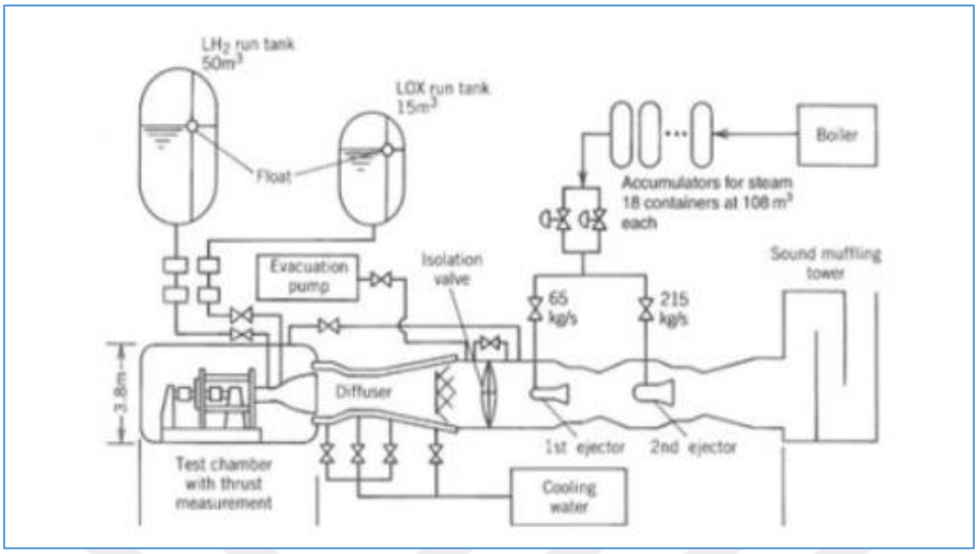


Figure 1.15: Korean Small Scale Engine Altitude Test Facility [18]

1.4 Purpose of Study

In this study, altitude test facility design concepts, 1D modeling of rocket engine design concepts, 1D modeling of an altitude test facility, and 1D simulation of the rocket engine and its altitude test facility are presented. To validate the rocket engine performance under different conditions (Start-stop behaviors and different atmosphere performance), the manufacturer's datasheet is used to verify the performance values of the simulated rocket engine, which is VINCI. VINCI has its altitude test facility, and some of the experimental data have been shared by the manufacturer. This study does not include all components of expander cycle, staged combustion cycle, and gas generator cycle rocket engines via a 1D simulation program. The main focus of this study is combining rocket engine design with altitude test facility design in a 1D simulation program. This system can be used for pre- and post-processing of rocket engine performance in altitude test facilities, potentially reducing the number of altitude tests required through simulation validation. For future studies, experimental results can be incorporated into the 1D simulation program for optimization. A digital twin of the rocket engine at different altitudes may be simulated, allowing for design optimizations without extensive testing.

2. LIQUID PROPELLANT ROCKET ENGINES & SOME EXAMPLES

2.1 Physics of Rocket Engine

2.1.1 Theory of thrust

Rockets operate according to Newton's Third Law, which states that every action result in an equal and opposite reaction. In simple terms, this means that when gases are expelled from a rocket engine at high speeds, they push the rocket in the opposite direction with an equal force. This propulsion is achieved through the combustion of propellants (fuel and oxidizer) inside the engine's combustion chamber, where they react to produce hot gases. These gases are then expelled through the engine nozzle at high velocities, generating thrust and propelling the rocket forward. Pressure acting on rocket engine is seen in Figure 2.1 [1] [4].

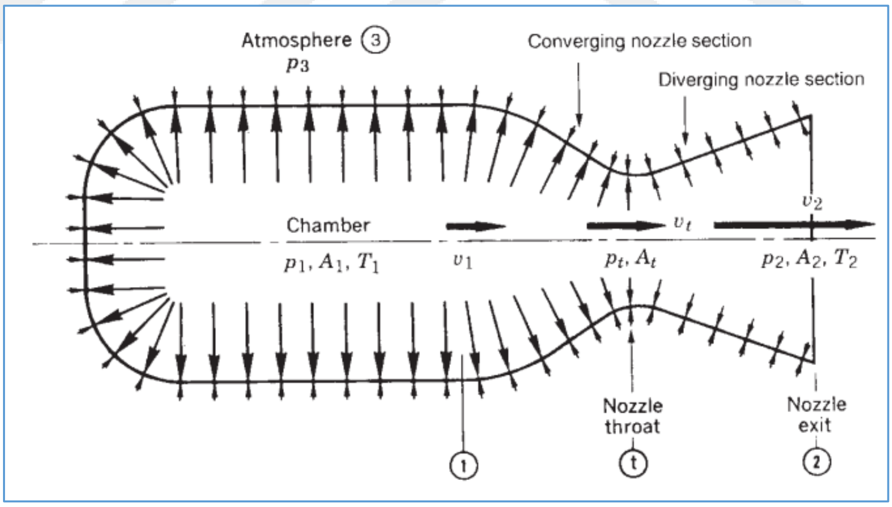


Figure 2.1: Pressure acting on rocket engine [4]

Rocket engine is accelerated via change of momentum of the outgoing gasses and thrust occurred. Newton's Second Law, which states force is equal to the rate of the change of momentum. Generally, Newton's Second Law is known as equation 2.1 in the modern physics lectures [1] [4].

$$F = m * a \tag{2.1}$$

In modern rocket technology, mass of the rocket is decreased due to burning of the fuel and oxidizer according to the time. Equation 2.1 forms in to equation 2.2 for the rocket behaviour [1] [4].

$$F = m(t) * \frac{du}{dt} + u * \frac{dm}{dt} \tag{2.2}$$

Control volume approach can be applied to the Figure 2.1. According to the control volume approach in inviscid fluid dynamics, control volume force equation can be modified as equation 2.3

$$F = \dot{m} * v_{exit} + (P_{exit} - P_{environment}) * A_{nozzle} \tag{2.3}$$

2.1.2 Rocket engine performance calculations

Rocket engines are various performance metrics. For the rocketry, performance of the rockets is generally compared according to their thrust, specific impulse, mixture ratios. To compare these values there are many equations that are generated to calculate each of performance metrics such as expansion ratio, exhaust velocity, Mach number and the other parameters.

Specific impulse is a measure of how efficiently a rocket engine utilizes its propellant. It represents the total impulse produced by the engine per unit mass of propellant consumed. In simpler terms, it tells us how effectively a rocket engine converts its fuel into thrust. A higher specific impulse means the engine can generate more thrust with less fuel, indicating greater efficiency [1] [4]. Specific impulse is shown by I_{sp} and it could be calculated in equation 2.4.

$$I_{sp} = \frac{F}{\dot{m} * g_0} \tag{2.4}$$

Mixture ratio is the measure of how efficiently oxidizer and fuel burn in combustion chamber of rocket engine. It is very important parameter for launch vehicle design because rocket engine storage tanks volume, total mass of the launch vehicle determination and center of gravity position to maneuvers are affected by consumption values of fuel-oxidizer. In literature, the representation may be different and *r* is used generally for the mixture ratio as it is calculated in equation 2.5 [1] [4].

$$r = \frac{m_o}{m_f} \tag{2.5}$$

Expansion ratio of the rocket is an important parameter to design nozzle of the rocket engines. This parameter is used for determining throat are to nozzle area of the outlet that throat area limits mass flowrate of the combustion chamber products and expanded area changes the velocity of the exhaust gas. In literature, the representation may be different and ε is used generally for the expansion ratio as it is calculated in equation 2.6 [1] [4].

$$\varepsilon = \frac{A_{nozzle\ outlet}}{A_{throat}} \tag{2.6}$$

Exhaust velocity indicates the rate at which the engine expels the hot gases from its combustion chamber. This factor significantly influences the engine's effectiveness and directly impacts the thrust it generates. Generally, higher exhaust velocities lead to increased thrust and efficiency in the rocket. In literature, the representation may be different and v_2 is used generally for the exhaust velocity as it is calculated in equation 2.7 [1] [4].

$$v_2 = \sqrt{\frac{2 * \gamma}{\gamma - 1} * R * T_c * \left[1 - \frac{P_{ambient}}{P_c}\right]^{\frac{(\gamma-1)}{\gamma}}} \tag{2.7}$$

Mach number indicates how fast the rocket is moving compared to the speed of sound. Designing propulsion systems to ensure effective flight across different speed ranges, including subsonic, supersonic, and hypersonic speeds. Mach number is also important for the following shock effects after the nozzle outlet. Mach number is also an important parameter for the nozzle throat that Mach number at the throat is equal to one in ideal condition. In literature, the representation M is used generally for the exhaust velocity as it is calculated in equation 2.8 [1] [4].

$$M = \sqrt{\gamma * R * T} \tag{2.8}$$

2.2 Liquid Propellant Rocket Engines

Liquid rocket engines are vital components of space propulsion systems, functioning by igniting liquid propellants, typically a blend of fuel and oxidizer, within a combustion chamber. This process results in the creation of high-pressure, high-velocity exhaust gases that are expelled through a nozzle, thereby generating thrust in accordance with Newton's Third Law of Motion. These engines play an indispensable role in modern rocketry, being essential for launching spacecraft, satellites, and conducting interplanetary missions.

Compared to solid rocket engines, liquid rocket engines generally provide superior specific impulse, a key metric for propellant efficiency crucial for achieving the velocities required for space travel. This heightened specific impulse translates to greater thrust per unit of propellant, rendering liquid engines indispensable for missions necessitating extended durations and deep-space capabilities. An advantageous feature of liquid rocket engines is their capacity for thrust control, enabling them to be throttled, shut down, and restarted as needed. This flexibility facilitates intricate maneuvers like orbital insertion, precise spacecraft adjustments, and gentle landings on celestial bodies, all vital for meeting the precision requirements of modern space exploration missions. Liquid rocket engines are adaptable to a wide array of propellant combinations, such as liquid oxygen (LOX) with kerosene (RP-1) or liquid hydrogen (LH2), allowing engineers to tailor performance to specific mission parameters. For instance, LOX/LH2 offers exceptional specific impulse suited for upper stages and interplanetary missions, while LOX/RP-1 provides simplicity and ease of storage advantageous for lower stages of launch vehicles. These engines boast a high thrust-to-weight ratio, enabling them to generate substantial thrust relative to their mass. This characteristic is particularly critical for lifting heavy payloads into orbit and beyond, rendering liquid rocket engines indispensable for both crewed and uncrewed space missions. Advanced designs, such as staged combustion cycles, further enhance the performance of liquid rocket engines. In these configurations, a portion of the propellant is burned in a pre-burner to drive turbopumps before undergoing complete combustion in the main chamber. This method maximizes efficiency and thrust, exemplified by engines like the Space Shuttle Main Engine (SSME) and the Russian RD-180, thereby enabling enhanced performance and efficiency in space missions [1] [2] [4] [19].

2.2.1 Liquid rocket engine components

Liquid rocket engines have a number of components in its structures. To get precise thrust force for the launch vehicle, combined version of the all components has to be worked together. In literature, there are many assemble version of the rocket engine is available and details are shared by the manufacturers. A detailed version of the rocket engine is seen in Figure 2.2.

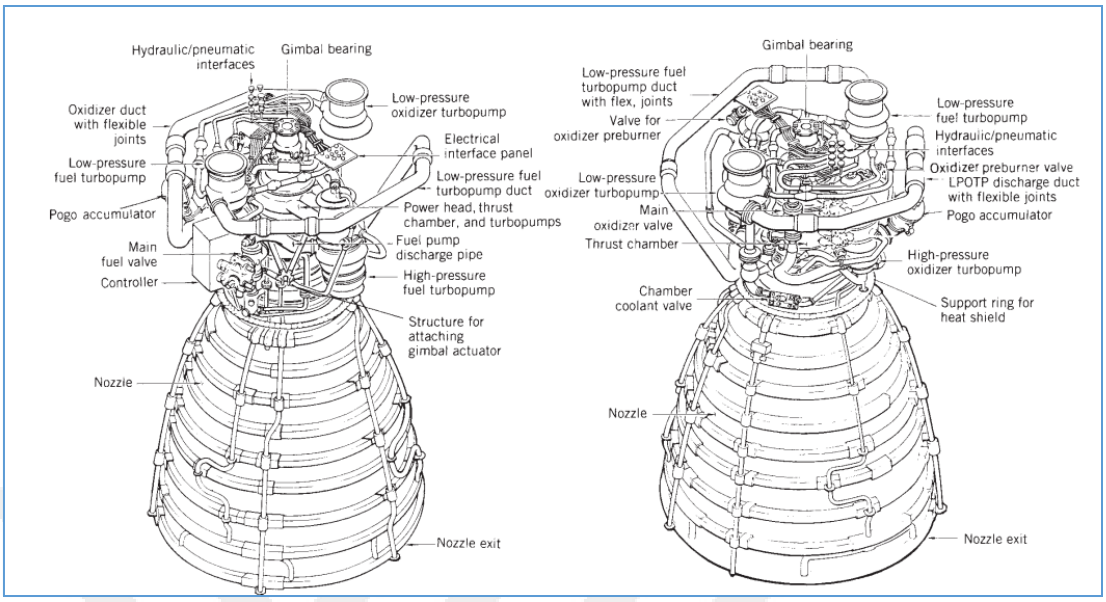


Figure 2.2: RS-25 Rocket Engine [4]

A schematic for a rocket engine can be seen in Figure 2.3.

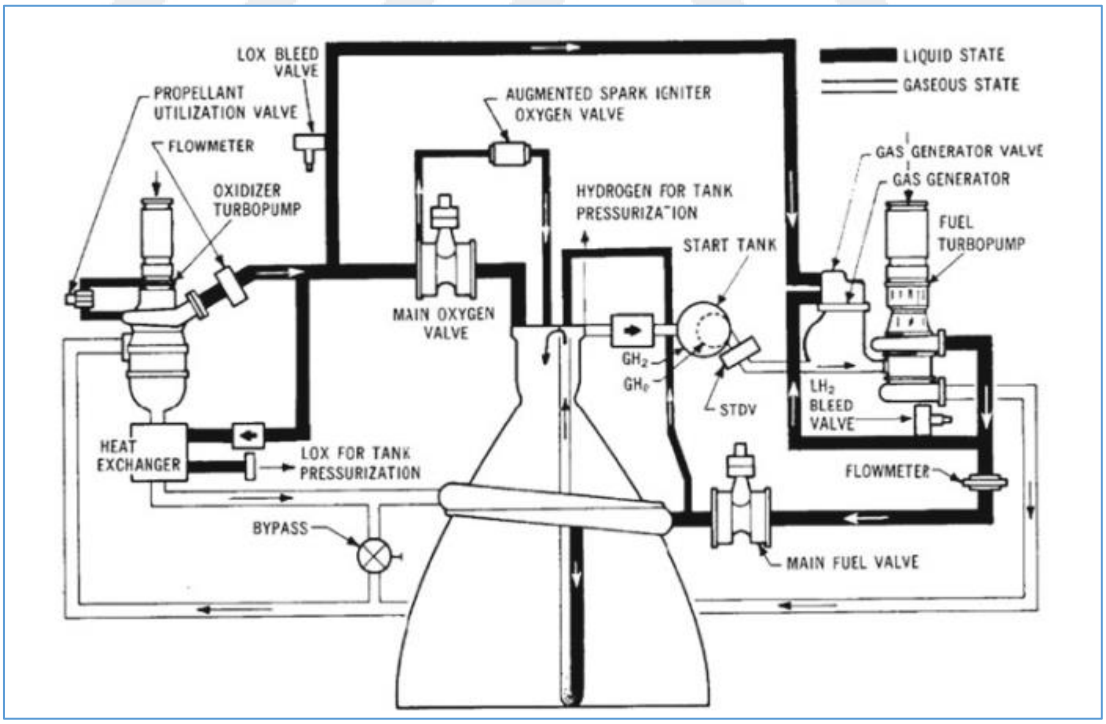


Figure 2.3: J-2 Rocket Engine [2]

2.2.1.1 Propellant tanks

Propellant tanks play a fundamental role in the configuration of liquid rocket engines, serving as essential reservoirs for storing liquid fuel and oxidizer until required for combustion. Constructed from materials carefully chosen for their combination of strength and lightweight properties, these tanks are engineered to endure the substantial pressures and extreme cold associated with propellant containment. Their primary objective is to maintain the stability and operational preparedness of the engine system, thereby facilitating optimal conditions for ignition and propulsion maneuvers during space missions. A structure of the propellant engine differs from common storage tanks and an example of the structure of propellant tank can be seen in Figure 2.4[1] [2] [4] [19].

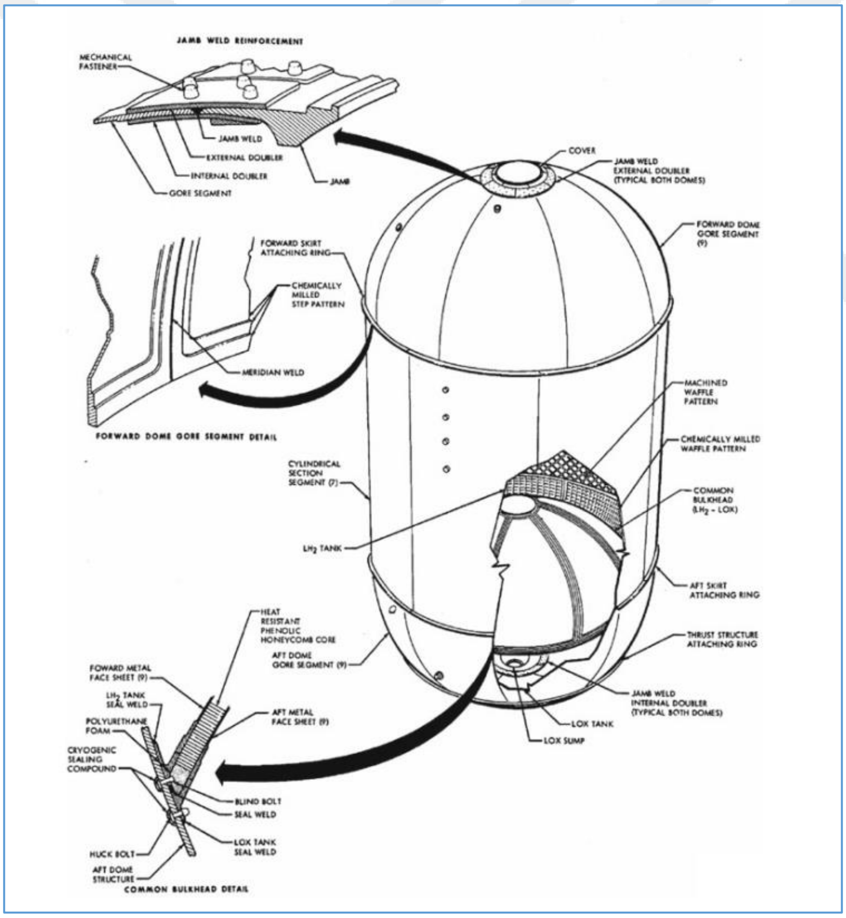


Figure 2.4: S-IVB Propellant Tank [2]

2.2.1.2 Turbopumps

The turbopump, a critical element in liquid rocket engine setups, consists of turbines, impellers, seals, bearings, and supplementary parts. Turbines, often operated by gas generators or staged combustion cycles, convert energy to propel the pump mechanism. Impellers are key in boosting fluid speed and pressure as they rotate within the pump. Sealing elements maintain fluid integrity, while bearings aid in the smooth rotation of components. Turbine feeding manifold is also important component of the turbopump that feeding supersonic speed gas to turbine for rotating. Collectively, these elements collaborate to pressurize and distribute propellants, guaranteeing top-notch engine functionality and dependability. A simple turbopump model can be seen in Figure 2.5 [1] [2] [4] [19].

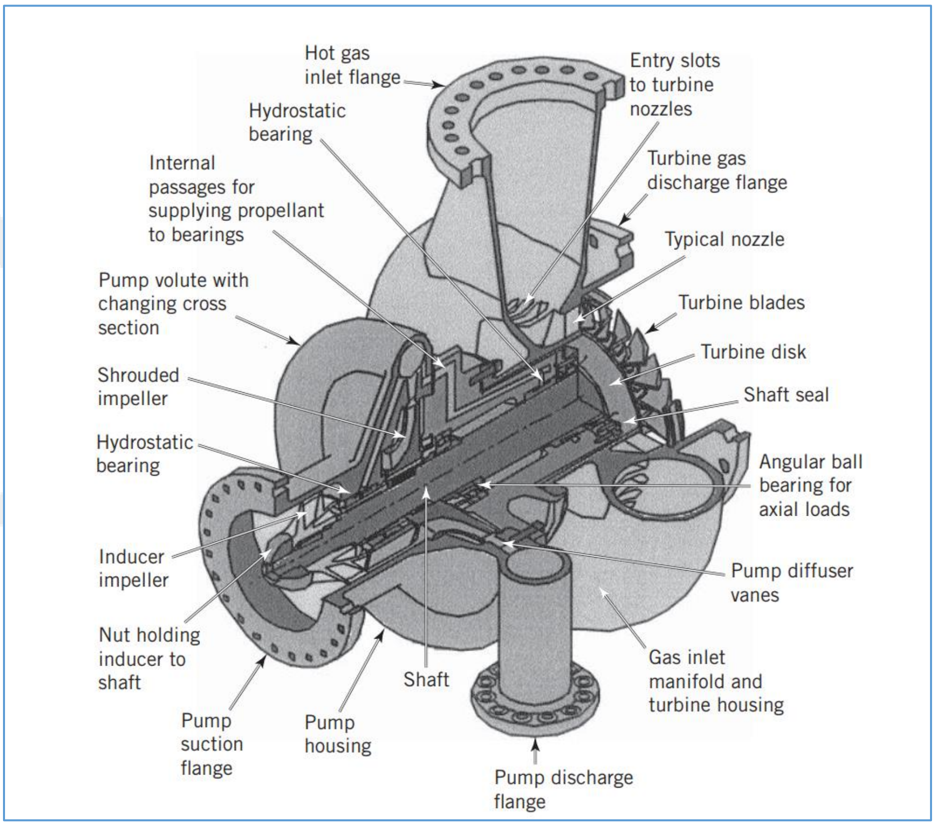


Figure 2.5: Simple turbopump model [4]

2.2.1.3 Combustion chamber

The combustion chamber serves as the core of the liquid rocket engine, where fuel and oxidizer combine and ignite to generate thrust. Crafted from materials engineered to endure intense temperatures and pressures, this chamber enables efficient combustion while limiting heat transfer to surrounding engine parts. Crucial for producing the high-pressure, high-velocity exhaust gases essential for propulsion, the combustion chamber plays a vital role in engine operation [1] [2] [4] [19].

2.2.1.4 Injector

Injectors represent intricate assemblies positioned at the inlet of the combustion chamber. Their function is to atomize the liquid propellants and evenly distribute them to promote swift and thorough combustion. By ensuring meticulous blending of the fuel and oxidizer, injectors facilitate efficient combustion, thereby reducing pressure losses and instability within the combustion chamber [1] [2] [4] [19].

2.2.1.5 Nozzle

The nozzle serves as the exhaust point of the combustion chamber, where the high-pressure exhaust gases expand and gain acceleration, resulting in thrust production. Different nozzle configurations are employed to enhance the velocity and efficiency of the exhaust gases, tailored to meet specific mission demands and operational circumstances. The nozzle design holds paramount importance in maximizing thrust generation and optimizing the overall performance of the engine [1] [2] [4] [19].

2.2.1.6 Valves:

Valves play a critical role as indispensable components governing the flow of propellants and fluids within the engine system. They oversee flow rates, direction, and pressure, ensuring meticulous operation throughout the mission duration. Valves encompass various types such as isolation valves, check valves, and control valves, each with designated functions aimed at preserving the integrity and dependability of the engine system [1] [2] [4] [19].

2.2.1.7 Pipelines

Pipelines, also termed feed lines, are responsible for transporting liquid propellants, pressurized gases, and coolant fluids among various engine components. These interconnected networks of tubes and ducts are designed for accurate flow control, minimal pressure losses, and resilience against corrosion and erosion. Pipelines play a vital role in ensuring the effective and dependable transfer of fluids within the engine system [1] [2] [4] [19].

2.2.1.8 Filters

Filters are incorporated into propellant lines and subsystems to eliminate contaminants and debris from propellants and fluids, thereby upholding system cleanliness and reliability. These filters encompass a range of designs such as mesh screens, porous cartridges, and centrifugal separators, each customized to meet specific filtration needs and flow rates. Their primary function is to prevent clogging and safeguard engine components from damage during operation, making them indispensable for ensuring optimal engine performance [1] [2] [4] [19].

2.2.1.9 Pressure regulators

Pressure regulators are tasked with upholding the desired pressure levels within the engine system by regulating flow rates through valve adjustments. Their function is to avert over pressurization and cavitation, thereby ensuring safe and efficient engine operation. Pressure regulators assume a pivotal role in sustaining system stability and performance throughout space missions [1] [2] [4] [19].

2.2.1.10 Heat exchangers

Heat exchangers play a vital role in facilitating the transfer of thermal energy between fluids within the engine system, thereby regulating temperatures and managing thermal gradients. These components encompass various forms such as regenerative cooling channels, radiators, and heat sinks, each designed to dissipate excess heat generated during combustion. Heat exchangers are indispensable for maintaining component temperatures within acceptable limits, thereby guaranteeing optimal engine performance and reliability [1] [2] [4] [19].

2.2.2 Propellant feed systems for liquid propellant rocket engines

Liquid propellant rocket engines require propellant to maintain mission during the flight. There are many sub-feed systems for the liquid rocket engines however the main two subjects are pressurized feeding systems and turbopump based feeding systems. The detailed family of the feeding systems for the liquid propellant rocket engines can be seen in Figure 2.6 [4].

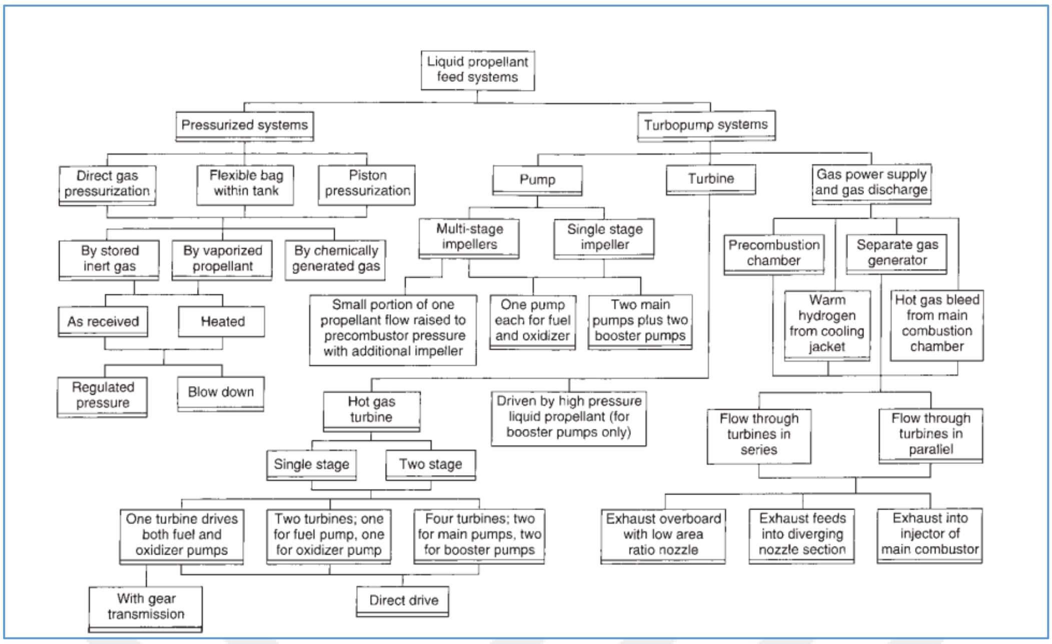


Figure 2.6: Liquid Propellant Rocket Engine Feeding Systems [4]

2.2.2.1 Gas pressurized feeding systems

The gas pressure system stands as one of the oldest and most reliable methods for propelling liquid propellants. This system comprises two main configurations which are regulated pressure and blow-down feed systems. In the regulated pressure arrangement, a gas pressure regulator maintains a constant tank pressure, ensuring consistent thrust during engine operation. Essential components of this system include high-pressure gas tanks, regulators, propellant tanks, valves, and feed lines, with additional features like check valves and pressure sensors enhancing system robustness. After propellant depletion, the pressurizing gas can be employed to clear residual propellant from lines and valves. In contrast, the blow-down feed system operates by utilizing the expansion of gas within propellant tanks to expel propellants. While blow-down systems are lighter than regulated pressure setups, they experience declining gas temperatures, pressures, and thrust as propellants are consumed. Various bipropellant pressurization methods and optional features are discussed in references, with monopropellant systems offering simplified designs due to utilizing only one propellant. Simple diagram of a blow-down type gas pressure feed system can be seen in Figure 2.7 [4] [19].

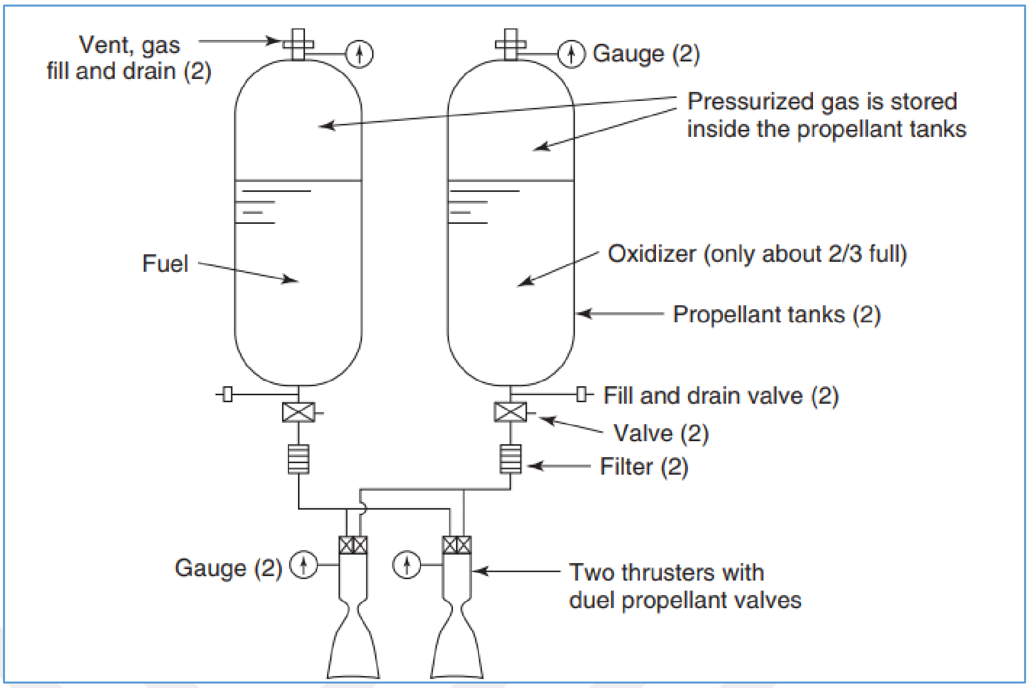


Figure 2.7: Blow-down Type Gas Pressure Feeding System [4]

2.2.2.2 Turbopump feed systems

The turbopump feed system represents an advanced technique employed in rocket propulsion to pressurize propellants. It involves individual pumps propelled by turbines energized by high enthalpy gases. These systems are favored for their effectiveness in high-thrust, extended-duration scenarios. Compared to alternative systems, turbopump feed systems are generally lighter, rendering them well-suited for applications such as space launch vehicles and long-range missiles. They are engineered to adapt to diverse application needs and are particularly adept in situations demanding substantial total impulse [4] [19].

Rocket engines utilizing a turbopump feed system typically operate through various engine cycles, extensively applied in real-world scenarios. These cycles entail specific pathways for propellant flow within engine components, methods for delivering hot gas to turbines, and procedures for managing turbine exhaust gases. Engine cycles are broadly classified into two categories which are open and closed cycles. In open cycles, the turbine's working fluid is either released into the nozzle exit section of the thrust chamber or discharged overboard after expansion. Closed cycles, also known as topping cycles, involve directing all turbine working fluid into the combustion chamber to optimize energy utilization. This method generally results in superior performance compared to open cycles, where turbine exhaust gases undergo limited expansion [4] [19].

For the engine cycles, there are three main design concepts are available which are expander cycle, gas generator cycle and staged-combustion cycle. Schematics for the three types of cycles can be seen in Figure 2.8.

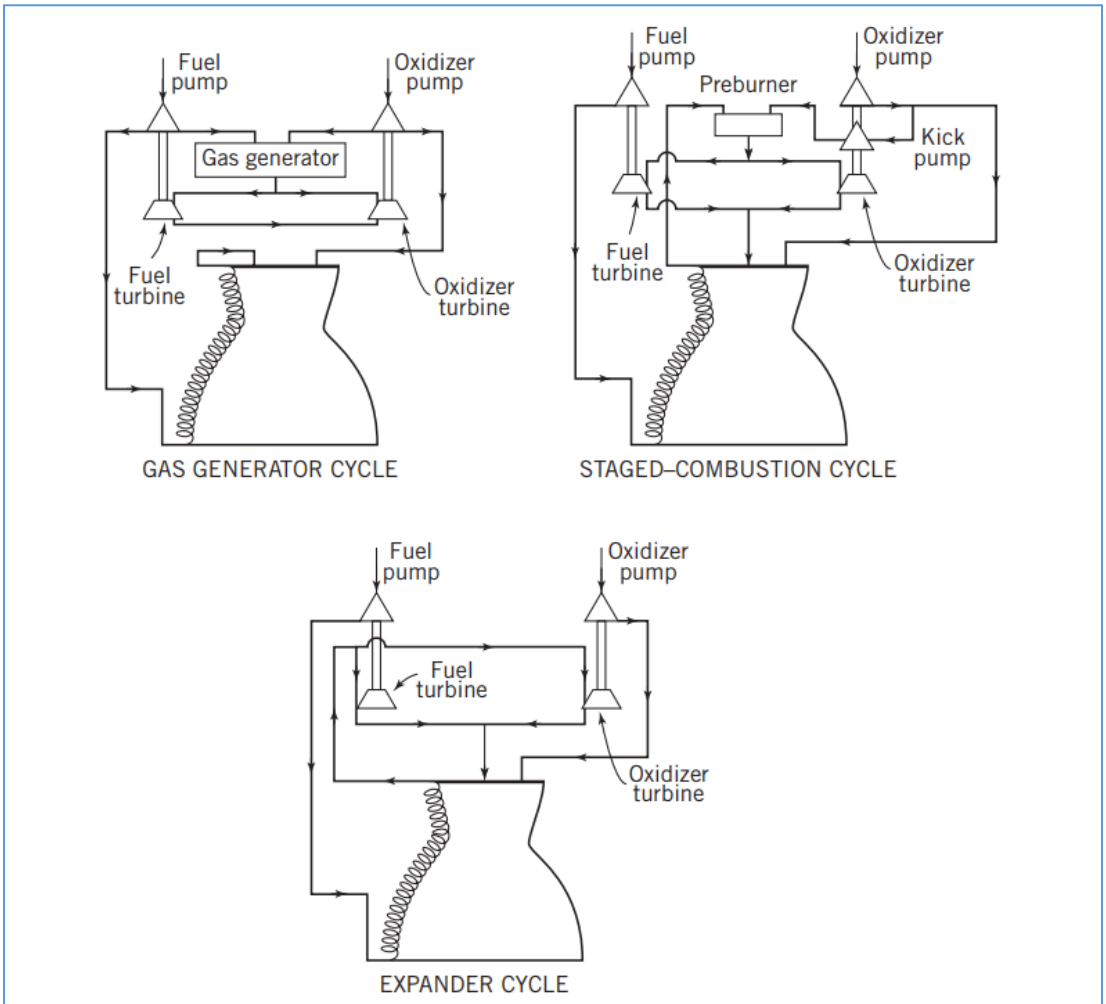


Figure 2.8: Liquid Propellant Engine Cycles [4]

The expander cycle, notably employed in rocket engines like the RL10 LOX/LH2 engine series and the VINCI engine examined in this thesis, utilizes various cryogenic fuels such as hydrogen and methane. This cycle entails heating and evaporating the fuel before directing it into low-pressure-ratio turbines via the engine's cooling system. A portion of the coolant is diverted past the turbine and reenters the exhaust flow, eventually reaching the injector or combustion chamber. This cycle offers advantages including high specific impulse, the absence of a gas generator, and relatively lightweight engine construction. RL10 engine variations, including the RL10B-2 featuring an extendible nozzle skirt, have proven effective in upper stages of diverse space launch vehicles. Thrust regulation in these engines involves controlling fuel gas flow to the turbine and bypassing it to maintain consistent chamber pressure [4] [19].

The gas generator cycle, most commonly employed among the three, is known for its simplicity, lower pressures, and reduced inert mass and engine cost compared to other cycles. In this system, turbine inlet gas is sourced from a separate gas generator, supplied either by pressurized propellant tanks or drawn from main propellant pump discharges. Turbine exhaust gases are typically discharged overboard through uncooled ducts and small nozzles, maintaining a fuel-rich mixture to enable the use of uncooled turbine blades and nozzle exit segments. An example of an engine operating on the gas generator cycle is the LOX/LH2 RS-68 rocket engine, utilized in the Delta IV Heavy launch vehicle. Unlike alternative cycles, the gas generator cycle features a parallel turbine flow-path with the thrust chamber flow-path, simplifying its development and operation. However, this configuration incurs a performance penalty due to factors such as incomplete expansion of turbine-drive gases and mixture-ratio shifts in the main thrust chamber. Despite these drawbacks, the inherent simplicity of the gas generator cycle makes it a preferred choice for both booster and space engines [4] [19].

The staged-combustion cycles utilize a coolant flow path similar to the expander cycle, alongside a pre-burner acting as a high-pressure gas generator. This pre-burner ignites fuel with a portion of the oxidizer to produce high-enthalpy gas for the turbines. The entirety of the turbine exhaust gas is then directed into the main combustion chamber, where it combines with the remaining oxidizer. This setup allows for operations at high chamber pressures, facilitating relatively compact thrust chamber designs. However, the added pressure drop in the pre-burner and turbines necessitates heavier and more intricate pumps, turbines, and piping. Despite these complexities, staged combustion cycles offer high specific impulses, although they come with increased engine intricacy and massiveness. Variations of this cycle have been utilized in engines like SSME and the RD-191 engine. The staged combustion cycle resembles the expander cycle with the turbine expelling directly into the thrust chamber, while combustion takes place upstream of the turbine, similar to the gas-generator cycle. Although offering superior performance potential, this cycle is also more intricate, challenging to develop, and heavier compared to its counterparts [4] [19].

2.2.3 Liquid propellant rocket engines propellants

Liquid propellant engines use different configurations according to their cycle, thrust generations, stages and launch vehicle design. Storage of the oxidizer and fuel is important as it is stated and various oxidizer-fuel types are used in rocket industry. As it is seen in Sutton's book, commonly used liquid oxidizers and liquid fuel information is given in Figure 2.9.

Propellant	Liquid Oxygen	Nitrous oxide	Nitrogen Tetroxide	Nitric Acid[a] (99% pure)	Rocket Fuel RP-1, RP-2	Liquid Hydrogen	Liquid Methane	Monomethyl-hydrazine	Hydrazine	Unsymmetrical Dimethyl-hydrazine	Water
Chemical formula	O_2	N_2O	N_2O_4	HNO_3	Hydrocarbon $CH_{1.97}$	para–H_2	CH_4	CH_3NHNH_2	N_2H_4	$(CH_3)_2NNH_2$	H_2O
Molecular mass	31.988	44.013	92.016	63.016	~175	2.016	16.04	46.072	32.045	60.099	18.02
Melting or freezing point (K)	54.8	182.29	261.95	231.6	225	14.0	90.67	220.7	275.16	216	273.15
Boiling point (K)	90.2	184.67	294.3	355.7	460–540	20.27	111.7	360.8	387.46	335.5	373.15
Heat of vaporization (kJ/kg)	213	374.3 (at 1 atm)	413[2]	480	246[b]	446	510[b]	808	1219[b]	543	2253[2]
Specific heat (kcal/kg-K)	0.4 (65 K)	0.209	0.374 (290 K) 0.447 (360 K)	0.042 (311 K) 0.163 (373 K)	0.48+ (298 K)	2.34[b] (20.27 K) —	0.835[b] —	0.700 (298 K) 0.735 (393 K)	0.736 (293 K) 0.758 (338 K)	0.704 (298 K) 0.715 (340 K)	1.008 (273.15 K)
Specific gravity[c]	1.14 (90.4 K) 1.23 (77.6 K)	1.23[b]	1.38 (293 K) 1.447 (322 K)	1.549 (273.15 K) 1.476 (313.15 K)	0.58 (422 K) 0.807 (289 K)	0.071 (20.4 K) 0.076 (14 K)	0.424 (111.5 K) —	0.8702 (298 K) 0.857 (311 K)	1.0037 (298 K) 0.952 (350 K)	0.7861 (298 K) 0.784 (244 K)	1.002 (373.15 K) 1.00 (293.4 K)
Viscosity (centipoises)	0.87 (53.7 K) 0.19 (90.4 K)	0.0146 (gas at 300 K)	0.47 (293 K) 0.33 (315 K)	1.45 (273 K)	0.75 (289 K) 0.21 (366 K)	0.024 (14.3 K) 0.013 (20.4 K)	0.12 (111.6 K) 0.22 (90.5 K)	0.775 (298 K) 0.40 (344 K)	0.97 (298 K) 0.913 (330 K)	0.492 (298 K) 0.48 (300 K)	0.284 (373.15 K) 1.000 (277 K)
Vapor pressure (MPa)	0.0052 (88.7 K)	5.025 (293 K)	0.1014 (293 K) 0.2013 (328 K)	0.0027 (273.15 K) 0.605 (343 K)	0.002 (344 K) 0.023 (422 K)	0.2026 (23 K) 0.87 (30 K)	0.033 (100 K) 0.101 (111.7 K)	0.0066 (298 K) 0.638 (428 K)	0.0019 (298 K) 0.016 (340 K)	0.0223 (298 K) 0.1093 (339 K)	0.00689 (312 K) 0.03447 (345 K)

Figure 2.9: Commonly Used Liquid Propellants Specifications [4]

2.2.3.1 Oxidizers for the liquid propellant rocket engines

There are many important oxidizers is available to select for liquid propellant rocket engines. Oxidizer selection is related with many considerations such as storage capabilities, densities, toxicity and hazards for the environment. LOX is the most used oxidizer for the liquid propellant engine all over the world. Other oxidizers that are used today are hydrogen peroxide, nitric acid, nitrogen tetroxide and nitrous oxide. All of the oxidizers have different advantages however reaching the LOX is easy and generally cheaper [4].

Liquid oxygen (LOX) is a widely utilized oxidizer in rocket propulsion, known for its bright white-yellow flame when burned with most hydrocarbon fuels. It has been successfully used with alcohols, kerosene-based jet fuels such as RP-1, gasoline, and hydrogen, making it a preferred choice for large rocket engines due to its high-performance potential. In this study, LOX is the main oxidizer for the VINCI rocket engine and is also employed in SpaceX's Raptor engines. Although LOX typically does not ignite spontaneously with organic materials at ambient pressures, confined mixtures under pressure can result in combustion or explosions. LOX enhances and accelerates the combustion of other materials, requiring that all contact surfaces be clean to ensure safe handling and storage [4].

2.2.3.2 Fuels for the liquid propellant rocket engines

There are also many types of fuels that are available for the liquid propellant rocket engines. Fuel selection is related with cost, storage capabilities, characteristic velocity, I_{sp} values and combustion efficiencies. There are many types of liquid fuels are used today and all the different fuels are used for indicated missions by the companies that develops liquid propellant engines. Hydrocarbon based fuels are used widely in aerospace industry. Jet fuel types JP-8 and JP-10 are used in turbojet engines widely. Kerosene-based hydrocarbons are used widely such as RP-1 and RP-2. Hydrazine, unsymmetrical dimethylhydrazine, monomethyl hydrazine are the other common fuels for the liquid propellant rocket engines. Liquid methane usage is popular in today's world. Raptor engines use liquid methane for powering the rocket engines and launch vehicle that is named Starship will be used for future Mars and Moon mission according to SpaceX company. Liquid methane can be found in huge amount and relatively low-cost fuel for the liquid propellant rocket engines [4].

LH2 produces a high-performance colorless flame when burned with LOX. It is an excellent regenerative coolant and is the lightest and coldest fuel, with a specific gravity of 0.07 and a boiling point around 20 K. This thesis details the VINCI rocket engine, which also uses LH2 in its design. Another notable rocket engine using LH2 is the RL10, which is modeled in 1D simulations. Due to its low density, LH2 requires large fuel tanks, leading to increased vehicle volumes. It's extremely low temperatures limit material options for pumps, cooling jackets, tanks, and piping due to brittleness. LH2 is produced from gaseous hydrogen through compression, cooling, and expansion, and must be converted to its stable form, parahydrogen, during liquefaction to prevent excessive boil-off. Storing LH2 is a challenge for space companies, despite its abundant availability [4].

2.3 VINCI Rocket Engine

2.3.1 VINCI rocket engine background

VINCI is a liquid propellant rocket engine that rocket development progress is maintained by ESA. VINCI rocket engine is the upper stage rocket engine which is used for high altitudes. VINCI is an expander cycle rocket engine that oxidizer of the VINCI is LOX and fuel is LH2 for this liquid propellant engine. The program of the VINCI development process started in 1998. Overall design of the engine and integration is conducted by Snecma company. VINCI has many components that responsibilities of the manufacture and procurement processes are shared between European companies [20]. The first design of the VINCI engine is called M1 and CAD model of the first assembled VINCI engine can be seen in Figure 2.10.

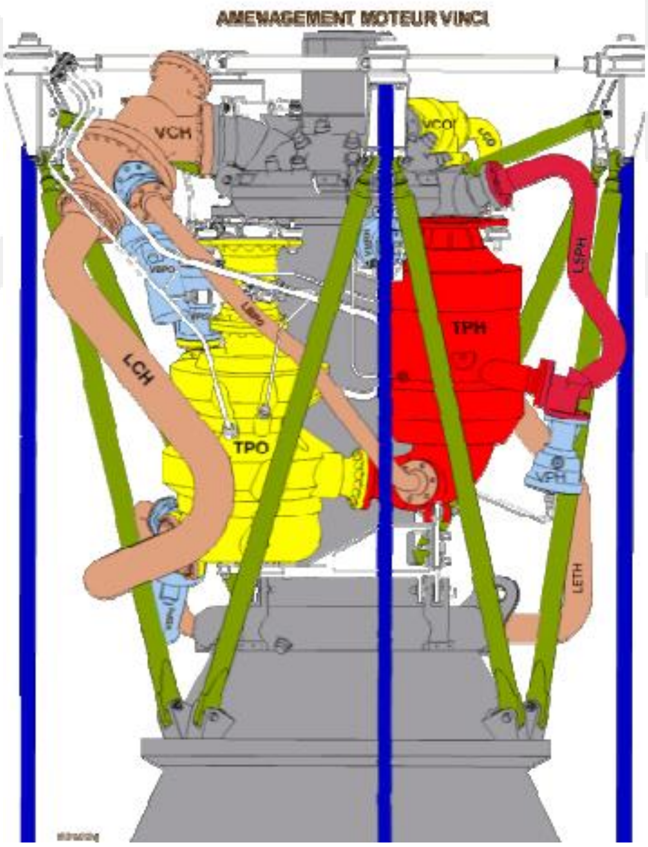


Figure 2.10: CAD Model of the Integrated VINCI Engine [20]

The first model of assembled VINCI rocket engine is called M1 and it is seen in Figure 2.11.

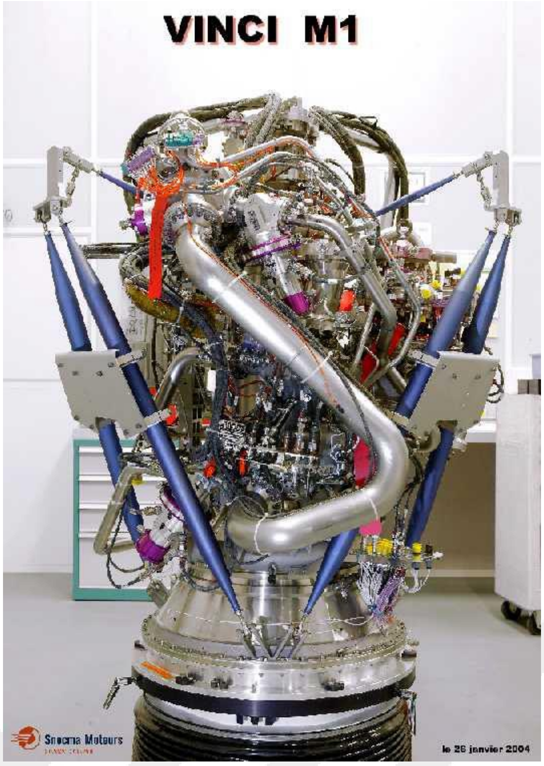


Figure 2.11: Integrated VINCI Engine [20]

In literature, the last version of the VINCI prototype is called M7 [21]. New VINCI models will be used for Ariane 6 upper stage according to the latest literature review. VINCI rocket engine's flow schematic can be seen in Figure 2.12 [22].

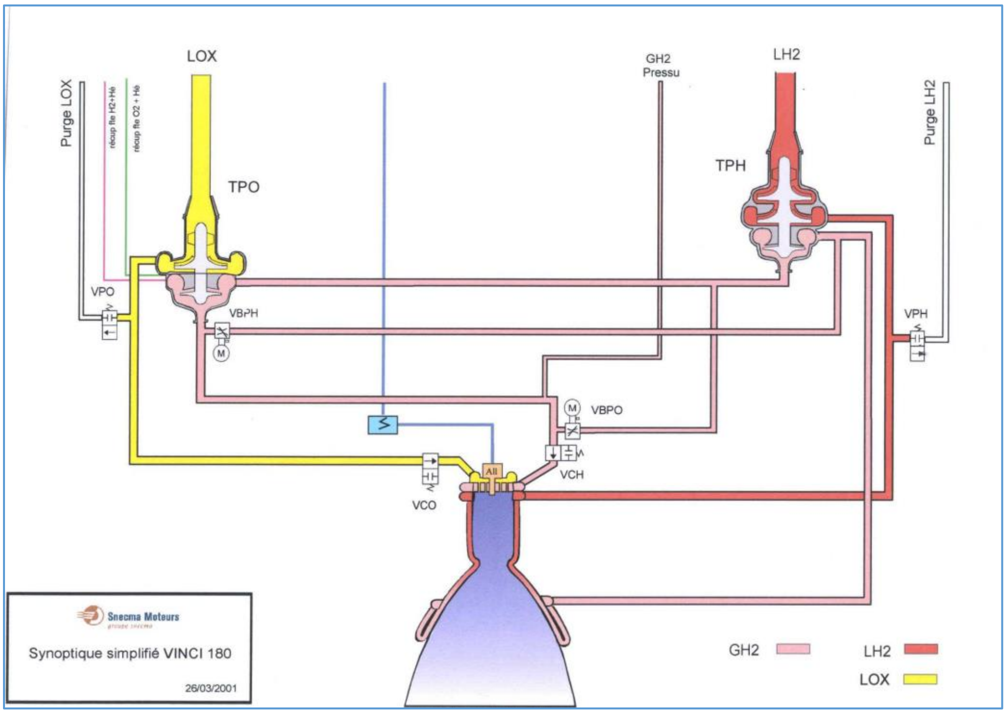


Figure 2.12: Flow Schematic of VINCI [22]

The last version of the VINCI rocket engine design can be seen in Figure 2.13 . VINCI will serve in Ariane 6 upper stage and possible Ariane 5+ program according to literature information [21].

Figure 2.13: Integrated VINCI Engine [21]

2.3.2 VINCI rocket engine performance specifications

VINCI has two different types for the performance according to related studies. 180 kN vacuum thrust is the first type and 130 kN vacuum thrust is the second type. Performance metrics of the VINCI can be seen in Figure 2.14 [23].

SPECIFICATIONS		
Cycle	Expander	
Vacuum thrust (kN)	180	130
Specific impulse (s)	457.2	458.2
Combustion pressure (bar)	60	45
Mixing ratio	6.1	5.5
Propellants	LOX - LH2	
Propellant flow rate (LOX/LH2, kg/s)	34.11 / 5.59	24.20 / 4.40
Nozzle outlet diameter (m)	1.84	
Operating points	180 kN	130 kN
TPH rotation speed (rpm)	90,000	70,000
TPO rotation speed (rpm)	18,000	14,000
Height (m)	3.22	

Figure 2.14: VINCI Engine Performance Specifications [23]

2.3.3 VINCI rocket engine performance tests

The development process of the VINCI comprises years and many tests were conducted during the years. The main test facilities for the VINCI are combustion chamber test facility, high altitude test facility, stage test facility, inducer test facility, bearing test facility, cryogenic spin test rig facility, cold flow test facility, injector test facility turbopump test facility, turbine test facility and other required test facilities [24] [25] [26] [27] [28].

VINCI turbopump test facility and fuel turbopump tests were held by test facility in Vernon, France. For the fuel test pump related turbopump design was used that is seen in Figure 2.15

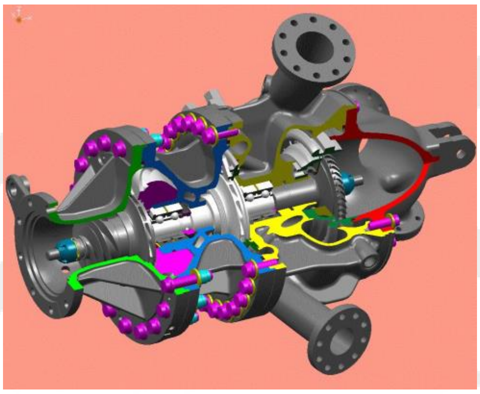


Figure 2.15: Fuel Turbopump of VINCI [24]

In turbopump test bench, critical speed observations, various rpm values performance, chill-down process observations have been satisfied in test bench that is seen

Figure 2.16: Turbopump Test Bench [24]

In inducer test bench, pump tests are conducted via water. The main consideration of the tests is observation of pump head rise in inducer and cavitation. It is very important test case for pump performance and maintenance for turbopump and rocket engine. Cavitation limits and pump head rise test results show that first prototype is seen successful for the VINCI [24].

Bearing is an important component for the turbopumps. It maintains rotation during powered shaft movement and bearing test validations are conducted in all turbomachinery areas. For VINCI's bearing test rig, radial and axial loads are applied between 75,000 to 120,000 rpm values. According the results of the performance, operation time of the bearing reaches 500 seconds. Bearing test rig configuration can be seen in Figure 2.17.

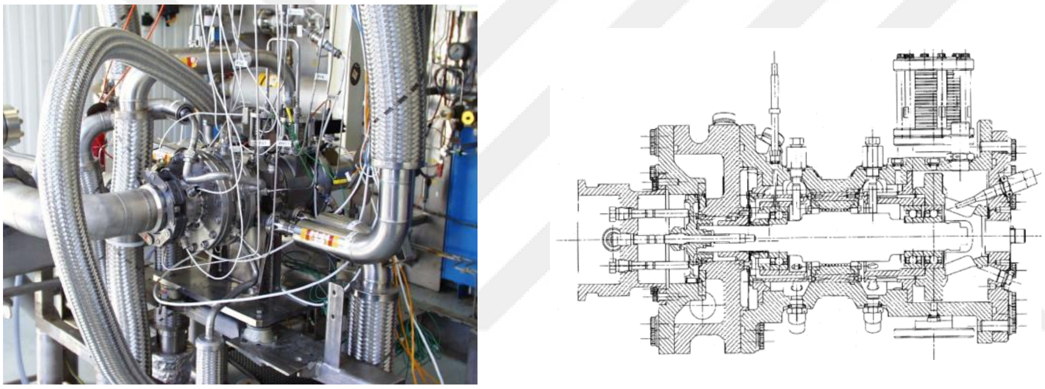


Figure 2.17: Bearing Test Rig [24]

Impeller spin tests for the VINCI was conducted via SCHENCK designed spin test rig. Burst limits in low temperature is measured via test rig that is seen in Figure 2.18 [24]

Figure 2.18: Spin Test Rig [24]

VINCI rocket engine has altitude test facility for its design and assemblage of the altitude test facility can be seen in Figure 2.19.

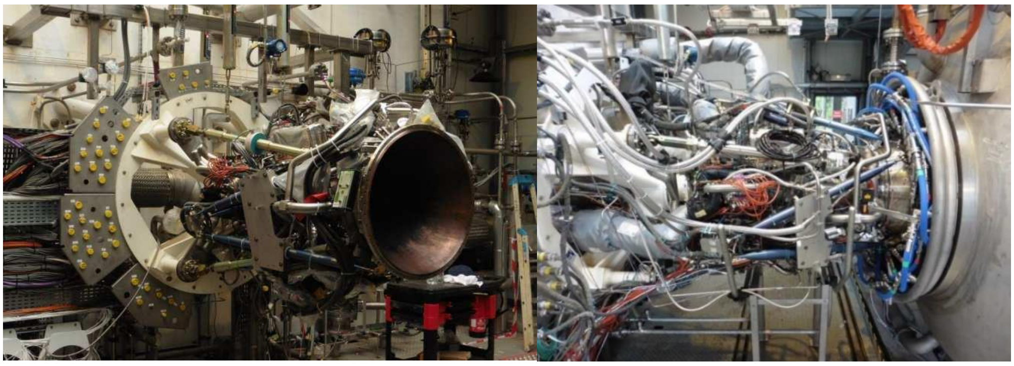


Figure 2.19: Altitude Test Facility [25]

Stage test facility is used for mission Concepts of Operations (CONOPS) for the related stage. Electrical and fluidic system configuration health is monitored that thermal behavior of the systems is observed during the firing. Upper stage configuration of the VINCI is seen in Figure 2.20 [26].

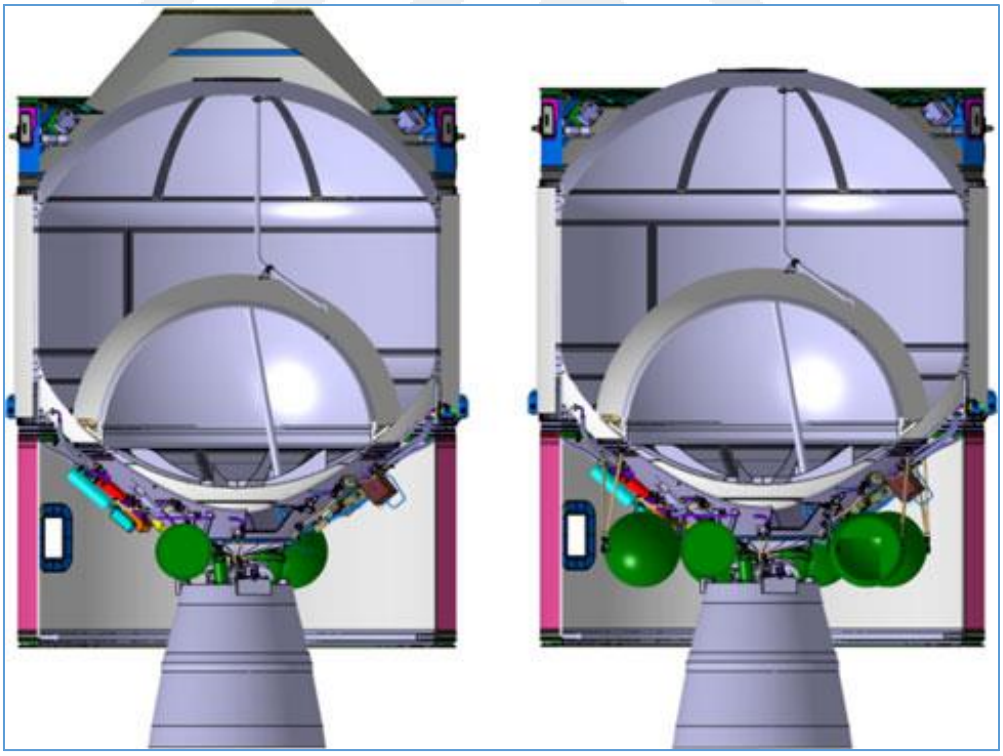


Figure 2.20: Stage Test Facility [26]

Turbine test rig is important test rig to observe generated torque, power and performance of the turbine. Simulant for the feeding is air in this test rig and similarity test conditions are 8 bar and 400-500K according to this test rig [27]. Turbine test rig of the VINCI can be seen in Figure 2.21.

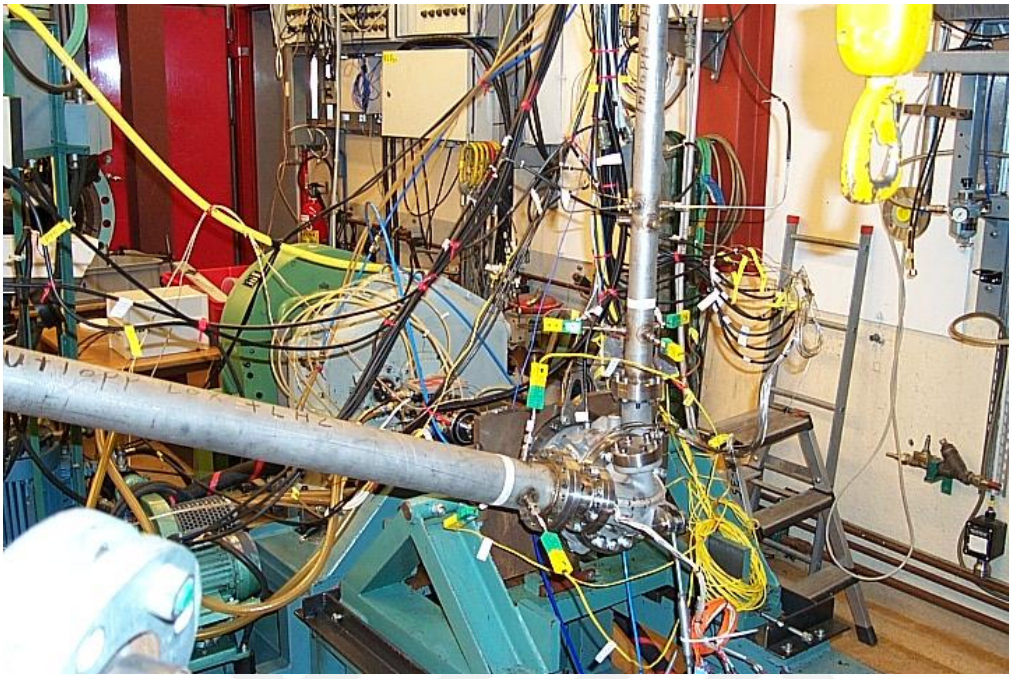


Figure 2.21: Turbine Test Rig [27]

3. 1D MODELLING OF AEROTHERMAL SYSTEMS

3.1 Simulation Program Library

In the Simcenter Amesim software, there a number of libraries available for different purposes. There are more than 35 different libraries included and, in this thesis, specific 6 libraries are used to model both VINCI engine and altitude test facility design which are signal-control, 1D mechanical, thermal, gas mixture, two-phase flow and liquid propulsion.

In signal and control library, most of control systems included PID control, transform function converters, constant signal producers, unit converters, variable signal producers and other important signal-control components are available as it is seen in Figure 3.1.

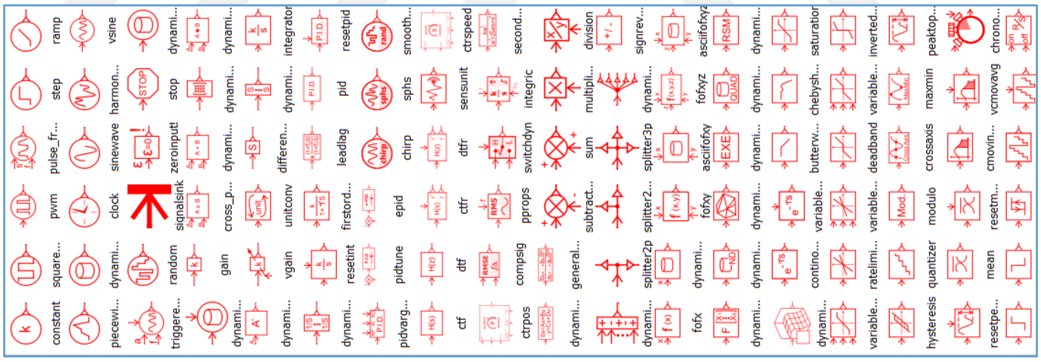


Figure 3.1: Signal and Control Library in Simcenter Amesim

In 1D mechanical library, there many components that are torque arms, springs, rotary parts, gearbox, dynamic load elements, dampers and screw-based components. For the liquid propellant rocket engines, pump and turbine connections are satisfied with reducer-connector shaft. Rotary power is maintained by the turbine and a rotary element is used. 1D mechanical library components can be seen in Figure 3.2.

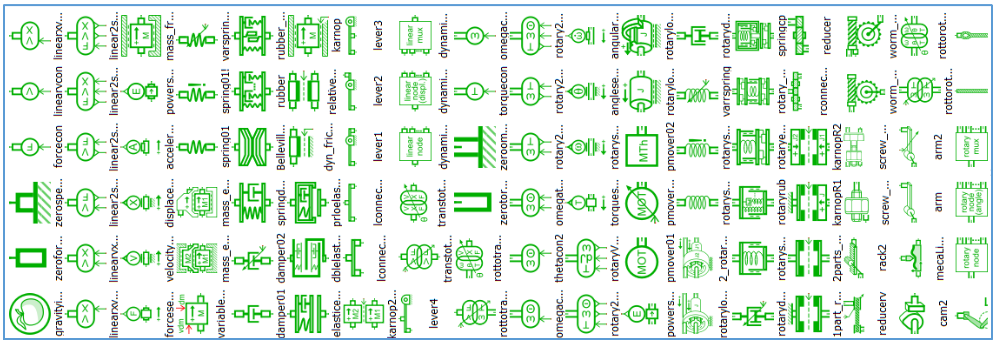


Figure 3.2: 1D Mechanical Library in Simcenter Amesim

In thermal library, heat transfer components and bulk sources and available. This library is used especially for the nozzle cooling system between fuel and nozzle. Thermal library components can be seen in Figure 3.3

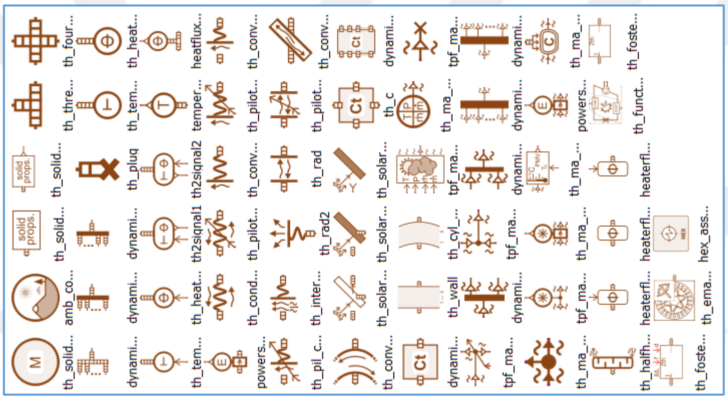


Figure 3.3: Thermal Library in Simcenter Amesim

Gas mixture library is the most used library in thesis study. Gas definition of combustion chamber products, convergent-divergent nozzles, pipes, ejectors, node connectors, gas mixture chamber and orifice components are used in created model. Components of the library can be seen in Figure 3.4.

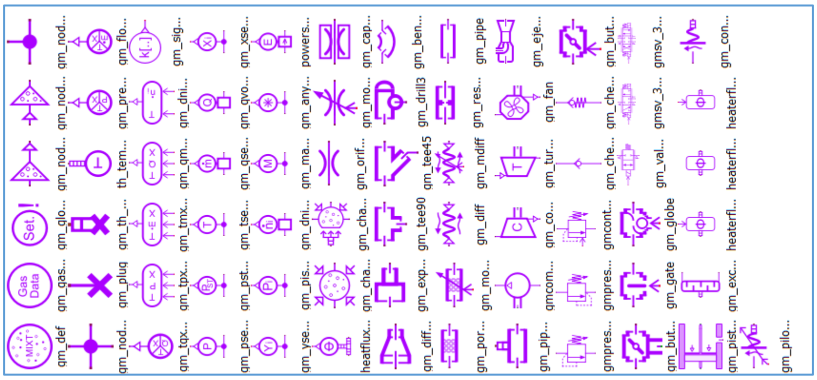


Figure 3.4: Gas Mixture Library in Simcenter Amesim

Two-phase flow library is used for LOX and LH2 feeding to the liquid propellant rocket engine. During the process, phase changes occur and this library-based defined sources are used. Library components resembles gas mixture library and can be seen in Figure 3.5

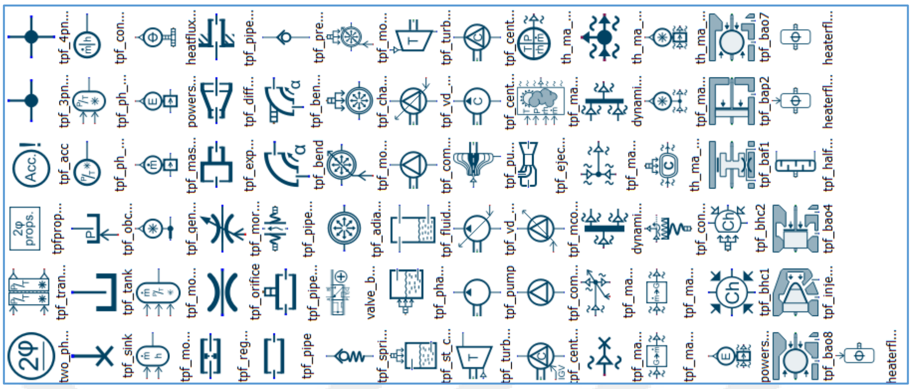


Figure 3.5: Two-Phase Flow Library in Simcenter Amesim

Liquid propulsion library is the most important library to model liquid propellant rocket engine. Components are limited in this library and combustion chamber, pumps, turbine and nozzle are modelled by this library that all components are seen in Figure 3.6

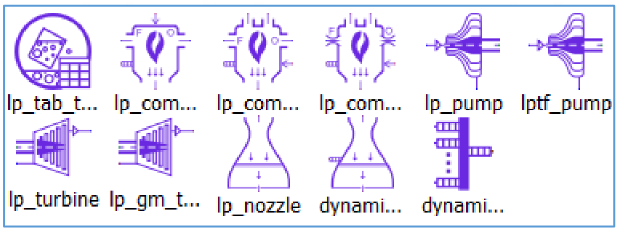


Figure 3.6: Liquid Propulsion Library in Simcenter Amesim

3.2 Simulation Components

Simulation program components are variable and each of the components are used for different purposes in simulation model of the VINCI engine and altitude test facility design. Simplified version of the model components can be seen in Table 3.1.

Table 3.1 : Component List and Specifications

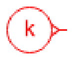
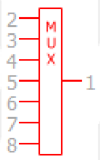
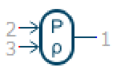
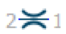
Component Name	**Representation**	**Background Information**	**Purpose of Usage**
Constant Signal		Components gives constant value to related systems	Defining boundary conditions for the temperature, pressure and exit products
Unit Converter		Unit of physical specification converting component	Unit conversion for the boundary conditions
Dynamic Multiplexer		From one input/output to separate input/output component	Outlet generated products for the outlet
Piecewise linear signal source		Component gives input to the control system with specific signal source	Combustion chamber ignition start and timing is specified
Generic modulated pressure and density source		Source for the feeding in specific pressure, density and temperature	To simplify turbopump-based feeding system for LOX/LH2 source
Orifice		Restriction for the fluid system to control pressure or mass flowrate	To determine and control mass flowrates of the LOX/LH2

Table 3.1 (continued) : Component List and Specifications

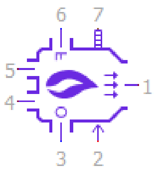
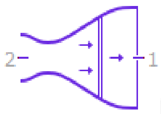
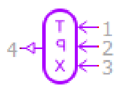
Component Name	**Representation**	**Background Information**	**Purpose of Usage**
Combustion Chamber		Combustion chamber component to start ignition for engine	VINCI's LOX/LH2 based fuel and oxidizer burning is satisfied
Nozzle		Acceleration of system and control thrust for the engines	VINCI's main thrust control nozzle in the system model
Zero flow source		Closing system input/output signal connection	Combustion chamber unused signal ports
Zero heat flow source		Closing system input/output signal connection	Combustion chamber unused signal ports
Gas mixture modulated pressure source		Source for the feeding in specific pressure, temperature and fractions	Outlet condition for the environment
Pipe		Connection for the fluid system components	Transferring exhaust gas to the environment
Progressive expansion/contraction		Velocity control of the fluid systems	Reducing nozzle outlet velocity in the pipelines.

Table 3.1 (continued) : Component List and Specifications

Component Name	Representation	Background Information	Purpose of Usage
Gas mixture chamber		Component's connectors and source for the systems	Connecting related components of the altitude test facility and creating vacuum chamber
Mass flow rate and temperature source		Mass flowrate and temperature input for the systems	Giving steam mass flow rate and temperature of steam to the system
Ejector pump		Ejecting system via driving flow from driven flow	Vacuum creation for the mixture chamber which is vacuum chamber in this study

3.2.1 Mathematical background of related components

For the orifice related definitions in formulas are defined in Simcenter Amesim as it follows: Cq maximum flow coefficient. Default set to 0.7 and is generally smaller than 1.0. ζ minimum singular pressure drop factor. Cv maximum flow capacity coefficient. Cv is numerically equal to the number of US gallons of water at 60 [degF] that flows through the element in one minute when the pressure differential across the element is 1 [psi]. Kv maximum flow capacity coefficient. Kv is numerically equal to the number of cubic meters of water at 60 [degF] that flows through the element in one hour when the pressure differential across the element is 1 [bar]. The following relations of orifice can be seen in Table 3.2. [29].

Table 3.2 : Orifice Mathematical Background

Component Name	Mathematical Background
Orifice	$\zeta = \frac{1}{C_q^2}$ $K_v = 0.865 \cdot C_v$ $A \cdot C_q = 1.6982 \cdot 10^{-5} \cdot C_v$

For the combustion chamber chemical reaction is defined in it for the reactions between fuel and oxidizer. In this thesis, LOX and LH2 based chemical reaction is solved in background. This component's equation definitions as it follows. U is the internal energy, $\dot{m}\ [j]$ is the mass flow rate in port, $h\ [j]$ is the specific enthalpy in port, $\delta\dot{W}$ is the work in unit time to the volume, $\Delta H(T)$ is heat of reaction, Ai general chemical species, n_i' is the mole of reactants, n_i'' is the mole of products, $\delta\dot{Q}$ is the heat exchange to the volume that is related with latent heat of vaporization, heat of combustion and heat exchange with the walls. Mass conservation of the combustion chamber is also satisfied as it follows. $\dot{m}_{in}$ and $\dot{m}_{out}$ are inlet-out mass flowrates, $\dot{x}_{in}$ and $\dot{x}_{out}$ are inlet-out mass fractions, M_j is the molar mass for the species, υ_j is the stoichiometric coefficient and R is the rate of the global reactions. Mathematical relations of the combustion chamber are seen in Table 3.3 [29].

Table 3.3 : Combustion Chamber Mathematical Background

Component Name	Mathematical Background
Combustion chamber	$\sum_j \dot{m}[j] \cdot h[j] + \delta\dot{W} + \delta\dot{Q} = \frac{dU}{dt} = \frac{d(m \cdot h - P \cdot V)}{dt}$ $dQ = \sum_j \dot{m}_j.\ (h_{j,inlet} - h_{j,Ref})$ $\sum_{i=1}^{N} n_i'.\ A_i \rightarrow \sum_{i=1}^{N} n_i''.\ A_i$ $\Delta H(T) = \sum_{i=1}^{N} n_i''.\ [\Delta H_{f,i}] - \sum_{i=1}^{N} n_i'.\ [\Delta H_{f,i}]$ $\frac{dm}{dt} = \sum_j \dot{m}[j] = x_{out,j}.\ \dot{m}_{out} - x_{in,j}.\ \dot{m}_{in} + M_j.\ (\nu_j.\ R)$

Nozzle mathematical background is based on isentropic equations and flow separation criteria like Stark Criteria, Summerfield Criteria and Schmucker Criteria. These criteria equation definitions as it follows. P_{sep} is the static pressure at flow separation point, P_{atm} is the atmospheric pressure and M_{sep} is the separation point Mach number. Critera mathematical background can be seen in Table 3.4 [29].

Table 3.4 : Separation Criteria Mathematical Background

Criteria	Mathematical Background
Stark Criteria	$\frac{P_{sep}}{P_{atm}} = \frac{\pi}{3.M_{sep}}$
Summerfield Criteria	$\frac{P_{sep}}{P_{atm}} = 0,35\ldots0,4$
Schmucker Criteria	$\frac{P_{sep}}{P_{atm}} = (1,88.M_{sep} - 1)^{-0,64}$

These criteria approximations are made according to the separation position Mach number and static pressure at separation point with iterative calculations. Mach number equation definitions as it follows. P_0 is the static pressure at extremity of the nozzle, T_0 is the static temperature at extremity of the nozzle, R is the gas constant per unit weight, A_1 is the cross-sectional area of the point of interest, M_1 is the Mach number of point of interest, γ is the specific heat ratio, A_T is the cross-sectional area of the throat, M_T is the Mach number of the throat. Iterative calculation of the separation point Mach number is used to control separation criteria and related mathematical background is seen in Table 3.5 [29].

Table 3.5 : Separation Point Mach Number Mathematical Background

Equations	Mathematical Background
M_{sep} iteration full equation	$\frac{P_0}{\sqrt{RT_0}} A_1 M_1 \sqrt{\gamma}\left(1+\frac{\gamma-1}{2}M_1^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}} = \frac{P_0}{\sqrt{RT_0}} A_T M_T \sqrt{\gamma}\left(1+\frac{\gamma-1}{2}M_T^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$

Table 3.5 (continued) : Separation Point Mach Number Mathematical Background

Equations	Mathematical Background
M_{sep} iteration simplified equation	$\frac{M_1}{\left(1+\frac{\gamma-1}{2}M_1^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}} - \frac{M_T}{\left(1+\frac{\gamma-1}{2}M_T^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}}\frac{A_T}{A_1} = 0$

Separation point pressure equation definitions as it follows. T_{sep} is the separation point temperature, *temp2* is the hot gases temperature at the nozzle inlet, v_{sep} is the velocity at separation point, ρ_{sep} is the separation point density, A_{sep} is the cross-sectional area of separation point, *dm1* is the mass flowrate, C_q is the flow coefficient. Iterative calculation of the separation point pressure is used to control separation criteria and related mathematical background is seen in Table 3.6 [29].

Table 3.6 : Separation Point Pressure Mathematical Background

Equations	Mathematical Background
P_{sep} iteration equations	$T_{sep} = \frac{temp2}{1+\frac{\gamma-1}{2}.M_{sep}^2}$ $v_{sep} = M_{sep}\cdot\sqrt{\gamma\cdot R\cdot T_{sep}}$ $\rho_{sep} = \frac{dm1}{A_{sep}\cdot C_q\cdot v_{sep}}$ $P_{sep} = \rho_{sep}\cdot R\cdot T_{sep}$

According to the nozzle design formulas for thrust and performance Sutton's book referenced formulas are same as Simcenter Amesim [4] [29]. For the pipes, general pipe friction loss formulas and general pipe flow equations are used for determining fluid transfer via pipelines [29]. For the convergent and divergent (expansion/contraction) elements classical Bernoulli approximations are used to control fluid velocities along the lines that is seen in equation 3.1 [29] [30] .

$$P + \frac{1}{2}\rho V^2 + \rho g h = constant\ along\ streamline \tag{3.1}$$

For the ejector pumps related diagram of the ejector pump representation is used as it is seen in Figure 3.7.

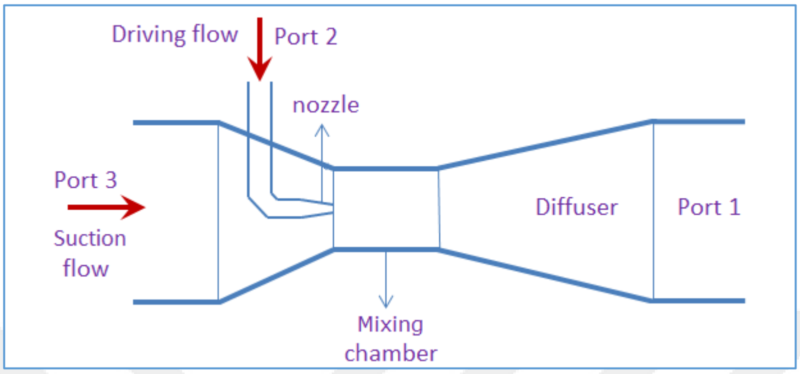


Figure 3.7: Ejector Pump Diagram in Simcenter Amesim [29]

Ejectors are used to generate vacuum along the vacuum chamber of engines. This component's equation definitions as it follows. P is the ratio in suction side, *w* is the pressure ratio in driving side, N is the pressure ratio, μ is the mass flowrate ratio [29]. The related formulas of the ejector pump are seen in Table 3.7.

Table 3.7 : Ejector Pump Mathematical Background

Component Name	Mathematical Background
Ejector pump	$P = \frac{p_3}{p_1}$ $w = \frac{p_2}{p_1}$ $N = \frac{p_1 - p_3}{p_2 - p_1}$ $\mu = \frac{dm_3}{dm_2}$

3.3 Assumptions

For the inlet boundary conditions for the LOX and LH2, pressurized LOX and LH2 values are used. LH2 temperature is given high for this boundary condition that some of the portion of the LH2 is used for surface cooling of nozzle. Hydrogen is normally stored in 20-21K according to Figure 2.9. Inlet conditions of the system is seen in Table 3.8.

Table 3.8 : Inlet Conditions

Element	Pressure (barA)	Temperature (K)
Liquid Oxygen	65	100
Liquid Hydrogen	190	200

For the outlet condition of the model, normal atmosphere conditions are used and it can be seen in Table 3.9.

Table 3.9 : Outlet Conditions

Element	Pressure (barA)	Temperature (K)
Dry Air	1	293

4. 1D MODELLING OF THE ROCKET ENGINE AND ALTITUDE TEST FACILITY

4.1 1D Models for Rocket Engines

1D modelling of the rocket engines are conducted via well-known programs such as EcosimPro and Simcenter Amesim. EcosimPro and Simcenter Amesim programs are widely used by rocket companies and agencies.

EcosimPro is a program that is supported by ESA and it has various libraries in it. Included important libraries for the rocket propulsion systems are Space Propulsion Simulation Toolkit (ESPSS), Dynamic Fluid Networks Simulation Toolkit (FLUIDAPRO), Liquid Propellant Rocket Engine Simulation (LPRES), Cryogenic System Simulation Toolkit (CRYOLIB), Control Library, Mechanic Library [31].

Simcenter Amesim is a program that is created by the Siemens company and it has a number of libraries in it. The main difference between Simcenter Amesim and EcosimPro is Simcenter Amesim program comprises hydraulic, pneumatic, gas mixtures, different types aerial vehicle design and various component designs of its structure. Included important libraries for the rocket propulsion systems are Pneumatic, Signal and Control, 1D Mechanical, Gas Mixture, Liquid Propulsion, Two-Phase Flow and Pneumatic Component Design libraries [32].

4.1.1 EcosimPro based 1D modelling of rocket engine

In literature, the common 1D models of the EcosimPro based rocket engines are RL10 and HM7. HM7 rocket engine is gas generator cycle liquid propellant rocket engine and it was used for Ariane launcher's upper stage. Related studies of the HM7 development are correlated via individual component designs then combination of the individual components is satisfied. The main components according to the related study is turbopump, thrust chamber and piping system between components. ESPSS library is used for designing all of the components of HM7 rocket engine. It is very important component that feeding lines of the oxidizer and fuel mass flowrates, pressure and temperature values affect engine performance considerably. The components of the turbopump in the model are boundary conditions, pipes, pumps, working fluids, nozzle, passage stages, valves, measurement sensors, controller units, gas generator, gearbox and thermal nodes. EcosimPro's ESPSS library includes a number of components that are related with space propulsion system modelling and components of ESPSS library. Components' brief descriptions and usage of the system can be seen in Table 4.1.

Table 4.1 : ESPSS Components and Usage

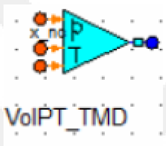
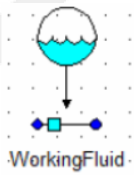
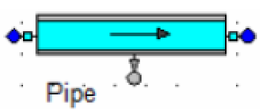
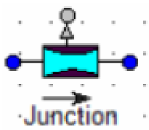
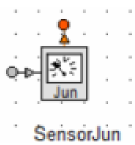
Component Name	**Representation**	**Background Information**	**Purpose of Usage**
VolPT_TMD		Components gives constant boundary conditions to related systems	Defining boundary conditions for the temperature and pressure
Working Fluid		Fluid type selection in related systems	Defining working fluid type in related fluid lines
Pipe		Pipe with insulated type according to material selection	Fluid line creation for fuel and oxidizers
Junction		Components connects resistive and capacitive ended components. Orifice definition can be made	Connecting fluid lines and injector definition for oxidizers and fuel lines
SensorJun		Components give information about physical values like pressure to control	Controlling the valve and other components according to signal

Table 4.1 (continued) : ESPSS Components and Usage

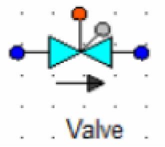
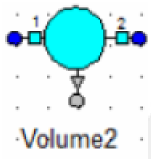
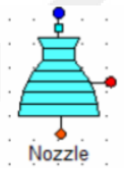
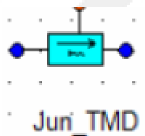
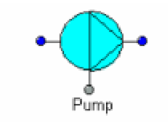
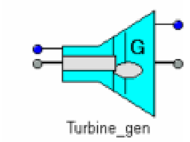
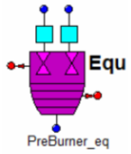
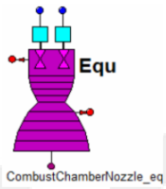
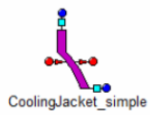
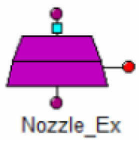
Component Name	Representation	Background Information	Purpose of Usage
Valve		Fluid line control valves for the processes	Controlling the pressure and mass flowrate of the oxidizer and fuel
Volume		Fluid line connecting with a volume definition	Connection between oxidizer, fuel and burned gasses
Nozzle		Rocket or other thrust generation system’s nozzle	Creating thrust for the propulsion system
Jun_TMD		Mass flowrate condition component to force related fluid line	Giving specific mass flowrate for the gases for engine
Pump		Pump definition for pressurize fluid lines	Turbopump’s pumping modelling for the oxidizer and fuel lines
Turbine_gen		Turbine definition to gain work in related fluid systems with turbine map values	Turbine and turbine stage definition of the gas generator cycle engine

Table 4.1 (continued) : ESPSS Components and Usage

Component Name	**Representation**	**Background Information**	**Purpose of Usage**
Preburner_eq		1D non-adiabatic combustion for related space system	Gas generator definition of the rocket engine
CombustChamberNozzle_eq		Combustion chamber with injectors, fluid volume and non-adiabatic nozzle	Controlling of the thrust for the rocket engine's burned products
CoolingJacket_simple		Getting heat from combustion chamber and nozzle	Heat absorption and hydrogen heating
Nozzle_Ex		Nozzle of the related propulsion system	Defined nozzle of the rocket engine

Related LH2 and LOX based gas generator cycle HM7 rocket engine's components can be created via using in Table 4.1. Turbopump model of the HM7 is seen in Figure 4.1 [33].

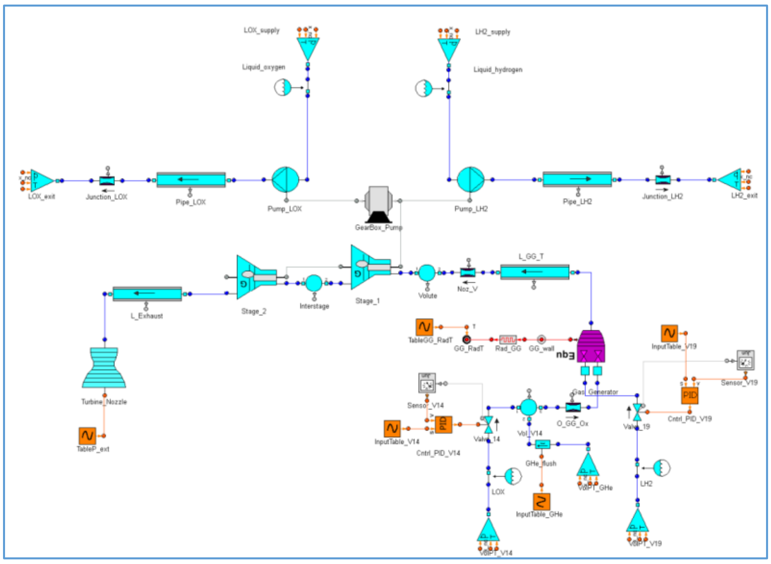


Figure 4.1: Turbopump Representation with ESPSS Library [33]

Other main component model is thrust chamber that includes combustion chamber. It is essential for generating thrust and performance of the rocket engine is affected hugely. Thrust chamber design and related components in model are boundary conditions, pipes, controller embedded tables, connector nodes, valves, thermal nodes, nozzle, combustion chamber and nozzle cooling skirt. Thrust chamber model of the HM7 engine is seen in Figure 4.2 [33].

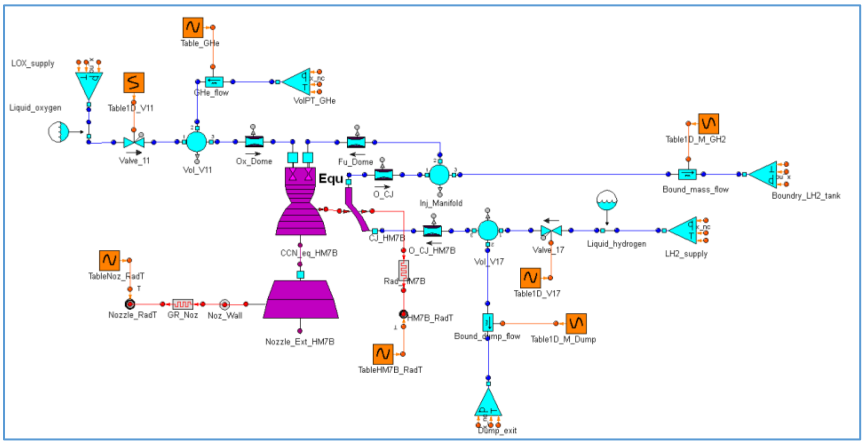


Figure 4.2: Thrust Chamber Representation with ESPSS Library [33]

The combination of the thrust chamber and turbopump is maintained by the feeding pipeline between them. In EcosimPro software, pipes can be designed as insulated pipes and thermal barrier adding port. Also in real model, there are cryogenic filters for the fuels and oxidizers. Complete model of the HM7 engine is seen in Figure 4.3 [33].

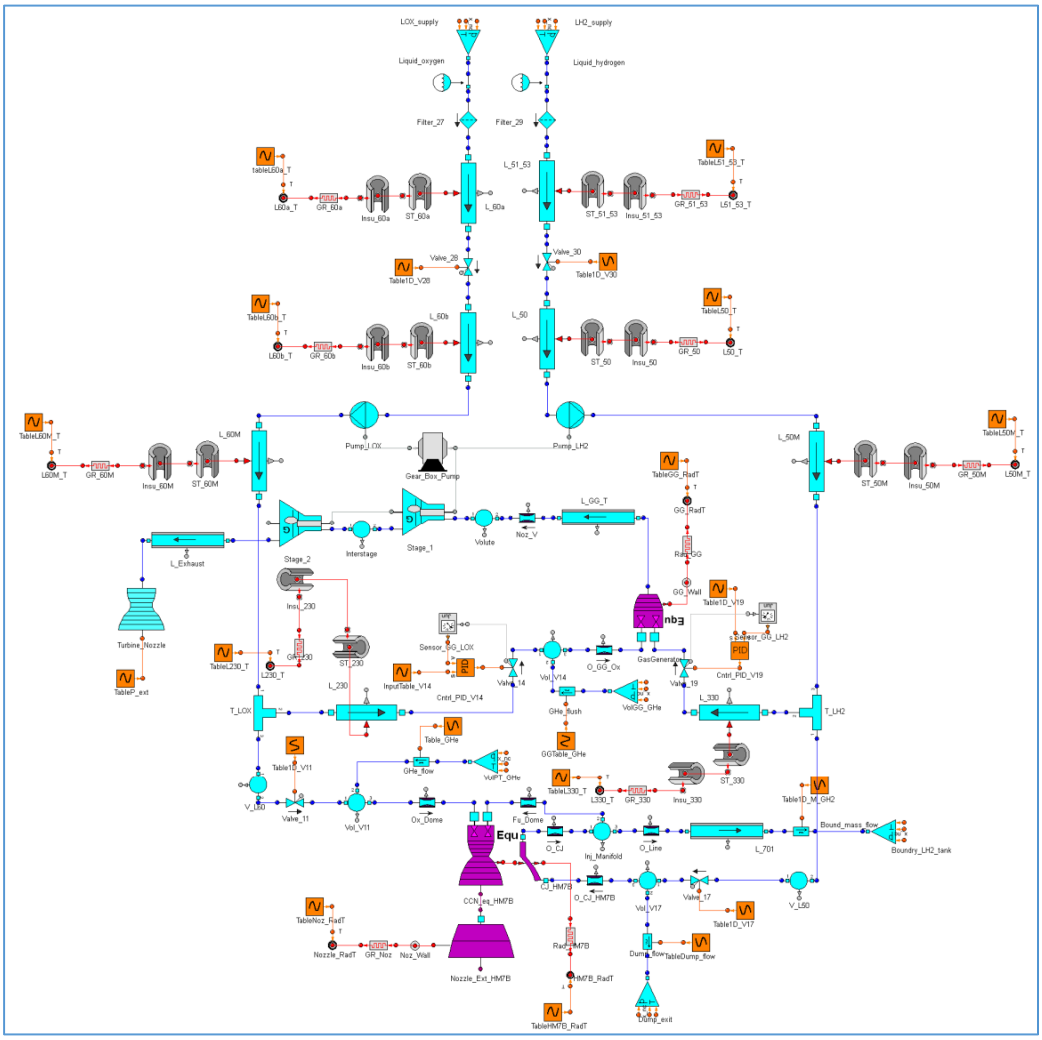


Figure 4.3: Thrust Chamber Representation with ESPSS Library [33]

Another 1D simulation and model is created for the RL10A-3-3A liquid propellant rocket engine. RL10A-3-3A is an expander cycle rocket engine and all design datasheets are available to model corrections for the 1D simulation programs. Details of the RL10A-3-3A are less than HM7 schematics that simplifications are made by assumptions. As it is seen in boundary conditions, working fluids, valves, pipelines, pumps, orifices, turbine and combustion chamber were placed in EcosimPro model schematic as it is seen in Figure 4.4 [33]

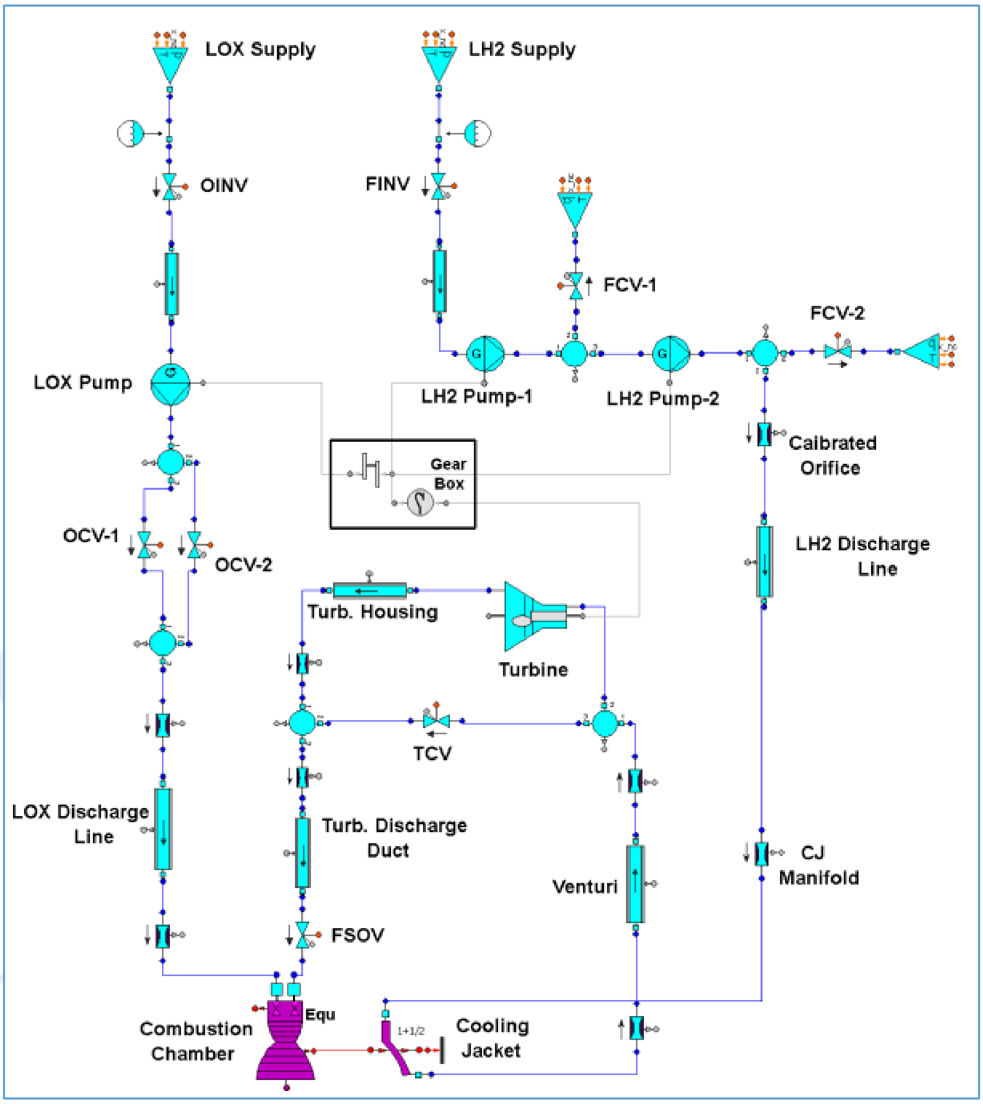


Figure 4.4: RL10A-3-3A Rocket Engine with ESPSS Library [33]

4.1.2 Simcenter Amesim based 1D modelling of rocket engine

In literature, RL10A-3-3A, Korean based liquid rocket engines and Chinese based liquid rocket engine studies can be seen. RL10A-3-3A is an expander cycle liquid rocket engine and its 1D design has also conducted in Simcenter Amesim program. As it is stated before, datasheets and performance values are shared in public for modelling and corrections of related programs. RL10A-3-3A simulation model is fed by the test results of the liquid rocket engine and this model is defined as digital twin. Digital twin brings about several benefits for design methodologies including rocket model performance without tests. Simcenter Amesim based model of the RL10A-3-3A can be seen in Figure 4.5. In this model boundary conditions, valves, state controllers, turbomachinery rotating elements, pumps, injectors, orifices, turbine, connector nodes, combustion chamber, nozzle, nozzle cooling skirt and heat absorbing elements are used. Nozzle cooling skirt is a super component which is a modelled component by the library elements to demonstrate heat transfer between fuel and nozzle [34].

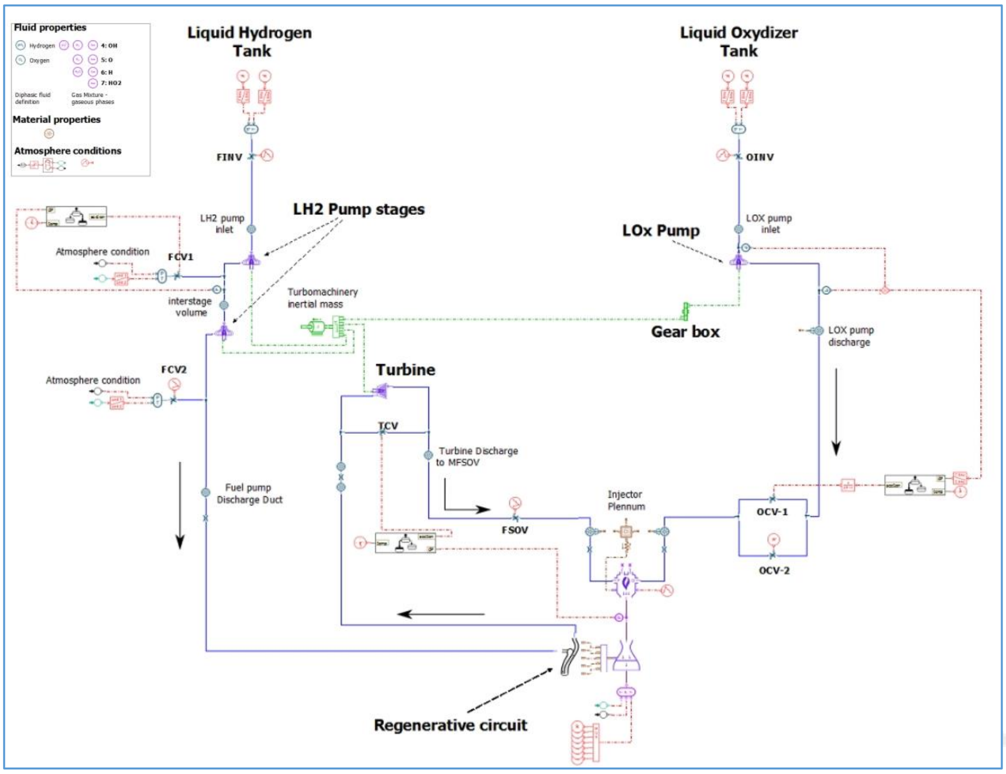


Figure 4.5: RL10A-3-3A Rocket Engine with Simcenter Amesim [34]

Another example of engine is orbital maneuvering engine model which is developed by Chinese academicians. Orbital maneuvering engines are used for reducing costs of the upper stage rocket engines costs and attitude determination around the orbit is controlled by high thruster rocket engine. Simcenter Amesim model of the engine resembles gas generator cycle but there is no certain statement that gas generator cycle is used. Rocket engine Simcenter Amesim model is seen in Figure 4.6 [35].

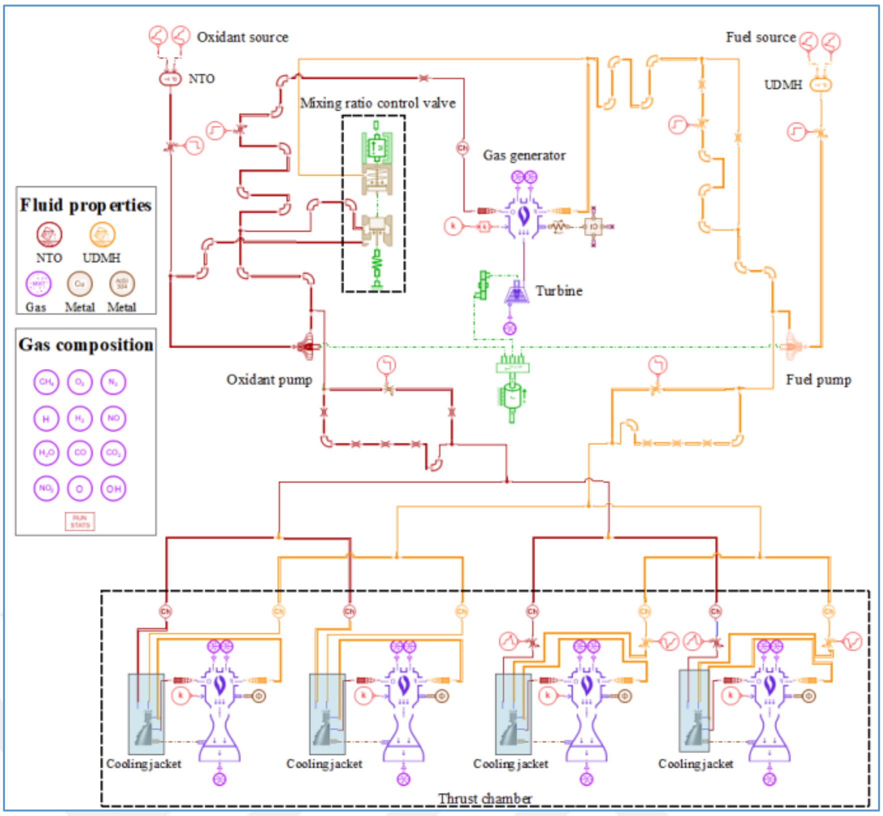


Figure 4.6: Orbital Maneuvering Engine with Simcenter Amesim [35]

Simcenter Amesim based another example is Korean academicians' study for the liquid propellant engine. In this study, start-up effects on turbopump based feeding system is modelled for the LOX/RP-1 liquid propellant rocket engine. Simplifications are applied to the model and modelled 1D simulation setup is seen in Figure 4.7 [36].

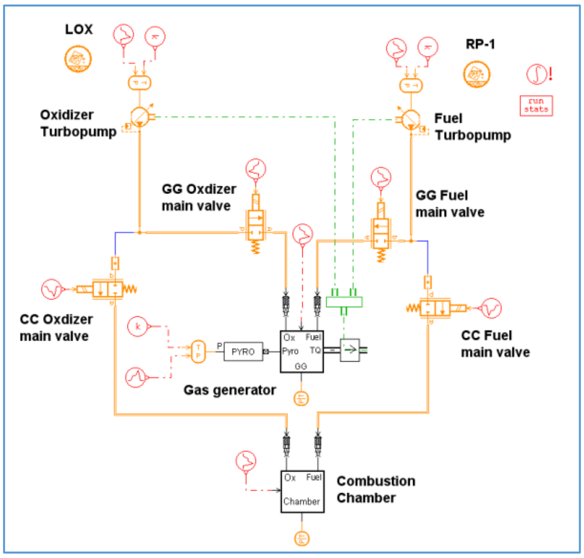


Figure 4.7: Korean Engine Model with Simcenter Amesim [36]

4.2 1D Modelling of the VINCI Engine

Simcenter Amesim software was used to create model of the expander cycle VINCI liquid propellant rocket engine. In the detailed model of the VINCI, turbopump modelling study was conducted. Turbopump elements which are pump and turbine maps are not shared by manufacturers according to the literature. Also valve characteristics and injector designs are not shared by the manufacturer of VINCI. Detailed model of the VINCI engine was created and simulated without validated values and simplification study was conducted to achieve feeding mass flowrate in precise pressure with orifice design. Verification and validation of the engine values have been checked with Figure 2.14. In this model, pipes are not considered due to verified pipe geometry information is not shared by the manufacturer.

The detailed version of the VINCI engine core study can be seen in Figure 4.8.

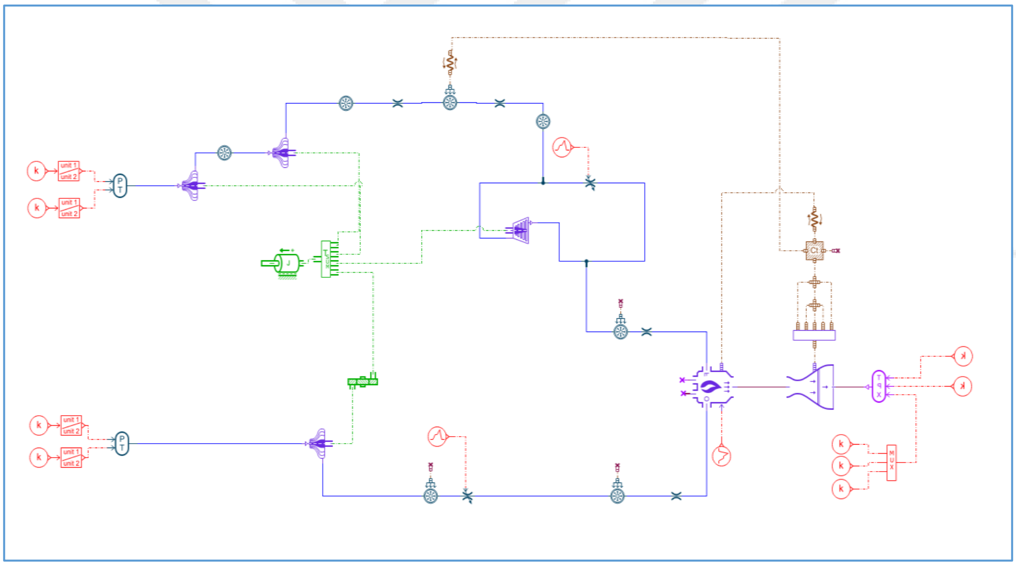


Figure 4.8: Detailed VINCI Model with Simcenter Amesim

Simplification of the VINCI Engine model was created and created model of the VINCI engine can be seen in Figure 4.9. Related components of the liquid propellant engine, assumptions, used libraries will be stated in related subjects. Energy budget of rocket engine is not taken into account due to nozzle skirt heat absorption is not shared by the manufacturer. This model could be created via Chemical Equilibrium with Applications (CEA) software but combining the components of the altitude test facility system connection in Simcenter Amesim is not validated case in this study.

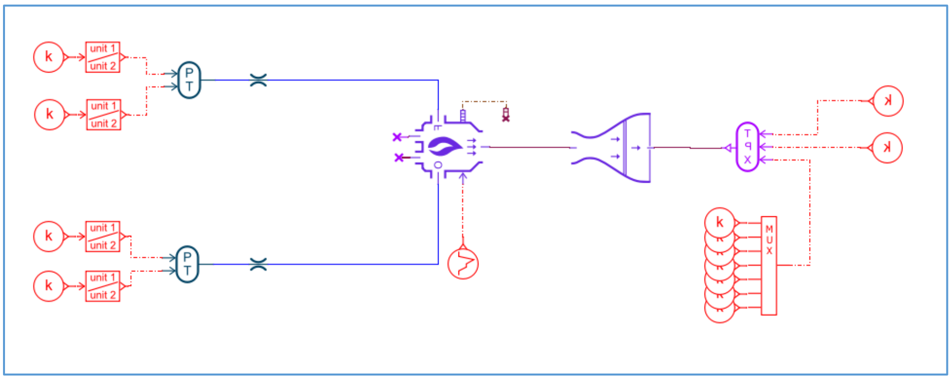


Figure 4.9: Simplified VINCI Model with Simcenter Amesim

Mass flow rates of the oxidizer and fuel are controlled using orifice designs before the combustion chamber. For the detailed VINCI model, mass flow rates of the oxidizer and fuel are managed through the relationship between pumps and control valves. The simplified version of the VINCI model maintains the same structure but without pumps and control valves for a steady-state system. The inlet boundary conditions for the oxidizer and fuel are shown in Table 3.8. These boundary conditions are given as pressure boundary conditions to simulate the pump head increase, where the pump outlet pressure is applied as the system inlet pressure. Orifices in the simplified design choke the system flow to achieve the required mass flow rate for the VINCI's oxidizer and fuel needs. Another choking condition is met by the nozzle of the VINCI rocket engine to generate thrust during flight. The outlet boundary condition of the system is set to standard day conditions, as seen in Table 3.9, which is valid for the altitude test system outlet during altitude tests. For the VINCI rocket engine model, vacuum condition thrust was observed for this simplified system according to the altitude test facility design outputs.

4.3 1D Modelling of the Altitude Test Facility

Altitude test facility design for the VINCI engine was created with Simcenter Amesim for modelled VINCI engine configuration. This altitude test facility system can also be used for other rocket engines to observe performance. In detailed version of the altitude test facility, steam ejector manifold usage studies were conducted. Steam ejectors are used to create vacuum condition for the vacuum chamber around engines. However big amount of ejector usage and node connection corrections detailed version of the altitude test facility was simplified. In VINCI’s real altitude test facility, there is also condenser component to regenerate water in system but in this design, methodology uses environment outlet that resembles to the NASA's altitude test facilities.

As it is stated before, simplification of the altitude test facility model was conducted and created model of the altitude test facility can be seen in Figure 4.10. Related components of the altitude test facility, assumptions, used libraries will be stated in related subjects.

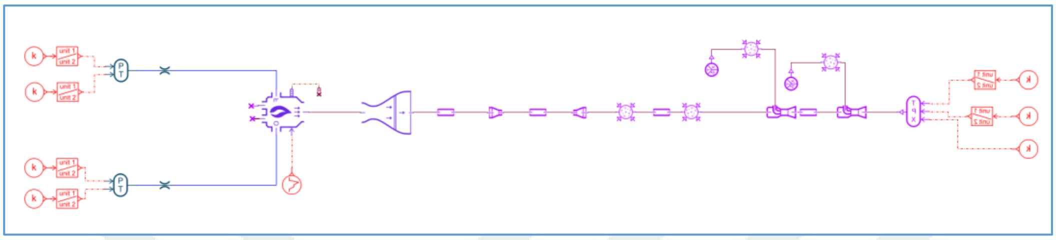


Figure 4.10: Simplified Altitude Test Facility model with Simcenter Amesim

5. MODELLING & RESULTS

5.1 1D Simulation Results of VINCI Engine

Oxidizer mass flowrate control has been controlled via orifice component. The inlet conditions of orifice have been given as is it seen in Table 3.8. Oxidizer orifice design specifications can be seen in Table 5.1.

Table 5.1 : Oxidizer Orifice Specifications

Title	Value
Pressure drop definition method	Cv
Singular pressure drop gain	0.5
Maximum flow coefficient Cv 2 -> 1	42
Maximum flow coefficient Cv 1 -> 2	0.8
Critical flow number	1000

1D simulation results of oxidizer mass flowrate has been found as 34.31 kg/s according to simulation result. Following performance figure can be seen in Figure 5.1.

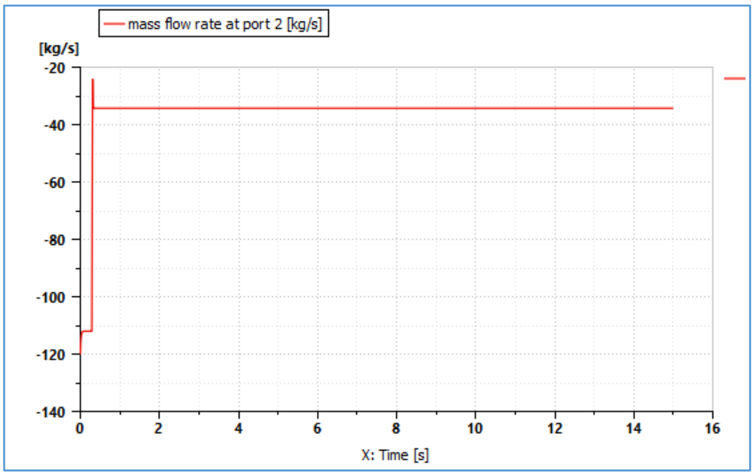


Figure 5.1: Oxidizer Mass Flowrate in kg/s

For the VINCI engine design specification, LOX mass flowrate is given as 34.11 kg/s according to the manufacturer [23]. Following error of the LOX mass flowrate is found as %0.59 for the simulation.

Oxidizer mass flowrate control has been controlled via orifice component. The inlet conditions of orifice have been given as is it seen in Table 3.8. Oxidizer orifice design specifications can be seen in Table 5.2.

Table 5.2 : Fuel Orifice Specifications

Title	Value
Pressure drop definition method	Cv
Singular pressure drop gain	1
Maximum flow coefficient Cv 2 -> 1	14.2
Maximum flow coefficient Cv 1 -> 2	0.8
Critical flow number	1000

1D simulation results of fuel mass flowrate has been found as 5.5 kg/s according to simulation result. Following performance figure can be seen in Figure 5.2.

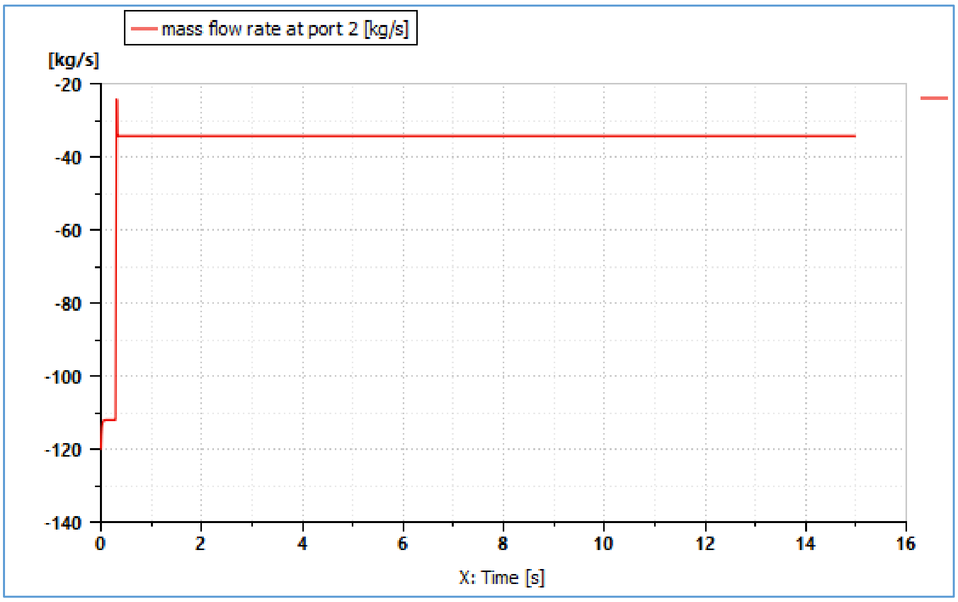


Figure 5.2: Fuel Mass Flowrate in kg/s

For the VINCI engine design specification, LH2 mass flowrate is given as 5.59 kg/s according to the manufacturer [23]. Following error of the LH2 mass flowrate is found as %1.61 for the simulation.

Volume of the combustion chamber has been accepted as 20L and ignition start was set to 0.3 with piecewise linear signal. 1D simulation results of combustion chamber pressure has been found as 59.78 barA according to simulation result. Following performance figure can be seen in Figure 5.3.

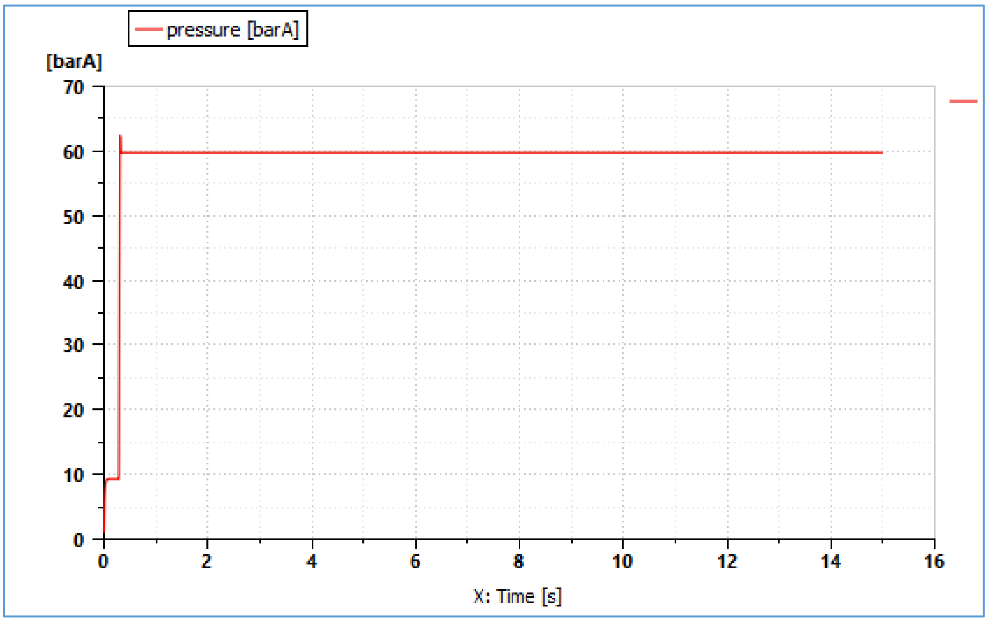


Figure 5.3: Combustion Chamber Pressure in barA

For the VINCI engine design specification, combustion chamber pressure is given as 60 barA according to the manufacturer [23]. Following error of the combustion chamber is found as %0.37 for the simulation.

Nozzle flow coefficient has been accepted as 0.3 and throat radius has been accepted as 0.13 meters for this simulation. 1D simulation results of engine thrust has been found as 178.01 kN according to simulation result. Following performance figure can be seen in Figure 5.4.

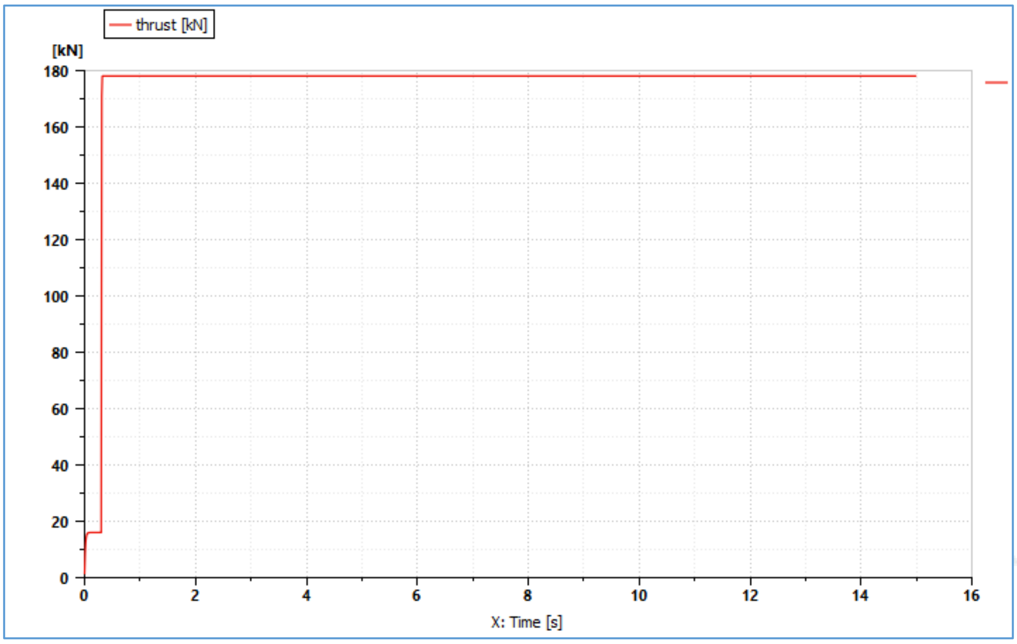


Figure 5.4: Engine Thrust in kN

For the VINCI engine design specification, engine thrust is given as 180 kN according to the manufacturer [23]. Following error of the engine thrust is found as %1.1 for the simulation.

1D simulation results of engine specific impulse has been found as 456.17 seconds according to simulation result. Following performance figure can be seen in Figure 5.5.

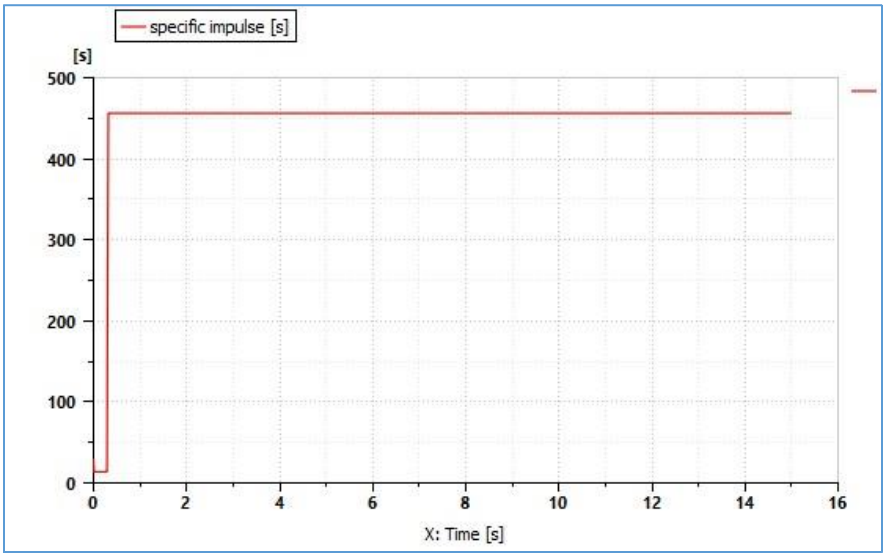


Figure 5.5: Engine Specific Impulse in s

For the VINCI engine design specification, engine specific impulse is given as 457.2 seconds according to the manufacturer [23]. Following error of the engine specific impulse is found as %0.22 for the simulation.

5.2 1D Simulation Results of VINCI Engine in High Altitude Test System

1D simulation results of engine thrust has been found as 178.15 kN according to simulation result. For the VINCI engine design specification, engine thrust is given as 180 kN according to the manufacturer [23]. Following error of the engine thrust is found as %1.03 for the simulation.

1D simulation results of engine specific impulse has been found as 456.3 seconds according to simulation result. For the VINCI engine design specification, engine specific impulse is given as 457.2 seconds according to the manufacturer. Following error of the engine specific impulse is found as %0.2 for the simulation.

1D simulation results of Altitude Test Facility vacuum pressure has been found as 42.93 mbarA according to simulation result. This value can be arranged via changing water mass flowrate and satisfaction condition accepted as below 50 mbarA. Following performance figure can be seen in Figure 5.6.

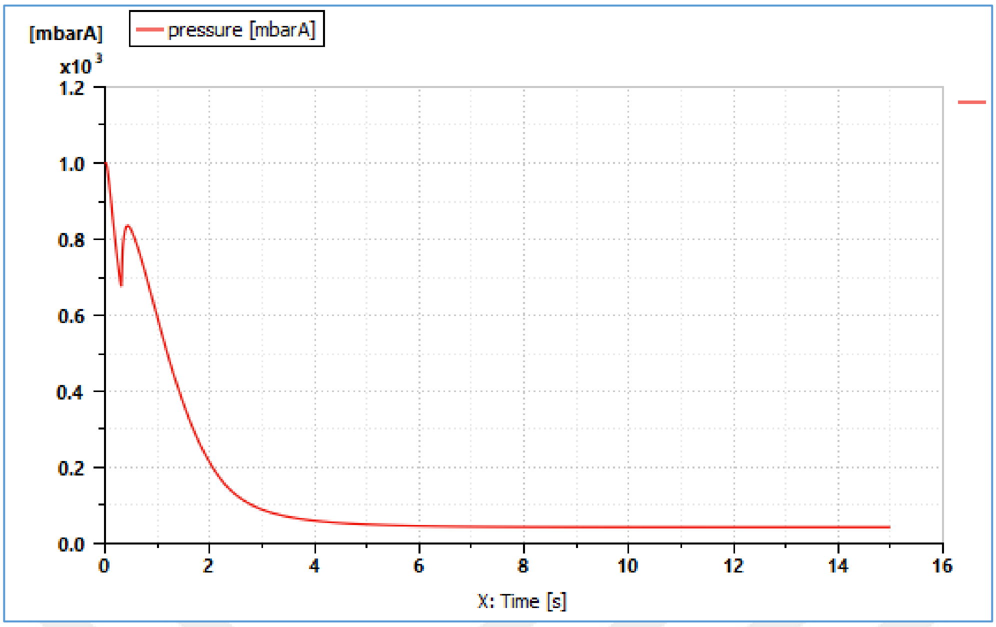


Figure 5.6: Created Vacuum Pressure in Vacuum Chamber in mbarA

For the VINCI engine experimental results below the 50 mbarA can be accepted as vacuum condition [12]. In simulation setup vacuum pressure has been find as 42.93 mbarA that improvement of the simulation of vacuum condition is %14.14 for this simulation.

For the stages of the steam ejectors water supply has been satisfied to the driving flow of the ejectors and simulation has been conducted. In altitude test facilities, there are two main stages for the ejectors and two main steam ejectors have been added to simulate steam mass flowrates. For the first stages of steam ejectors, 110 kg/s steam supplied as it is stated in literature and simulated mass flowrate figure can be seen in Figure 5.7 [12].

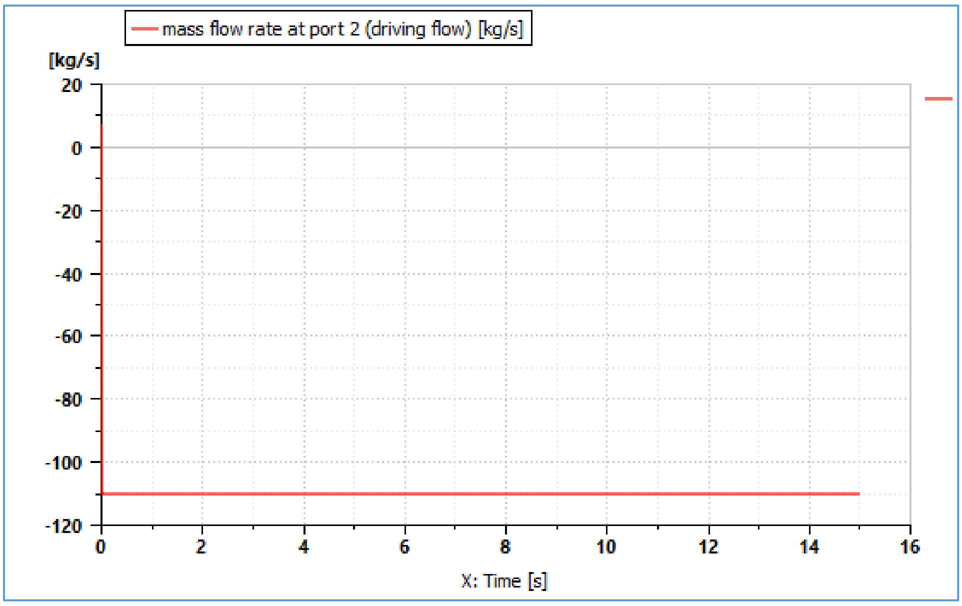


Figure 5.7: First stage steam ejector mass flowrate in kg/s

For the second stages of steam ejectors, 141 kg/s steam supplied and simulated mass flowrate figure can be seen in Figure 5.8. In related literature 118 kg/s is found and error band for 2nd stage mass flowrate is %19.5 [12]. Error band is relatively high because created vacuum value is better than literature study. Lower steam mass flowrate could satisfy the literature study steam mass flowrate and result in closer value for the literature. In this study, better vacuum condition desired according to literature.

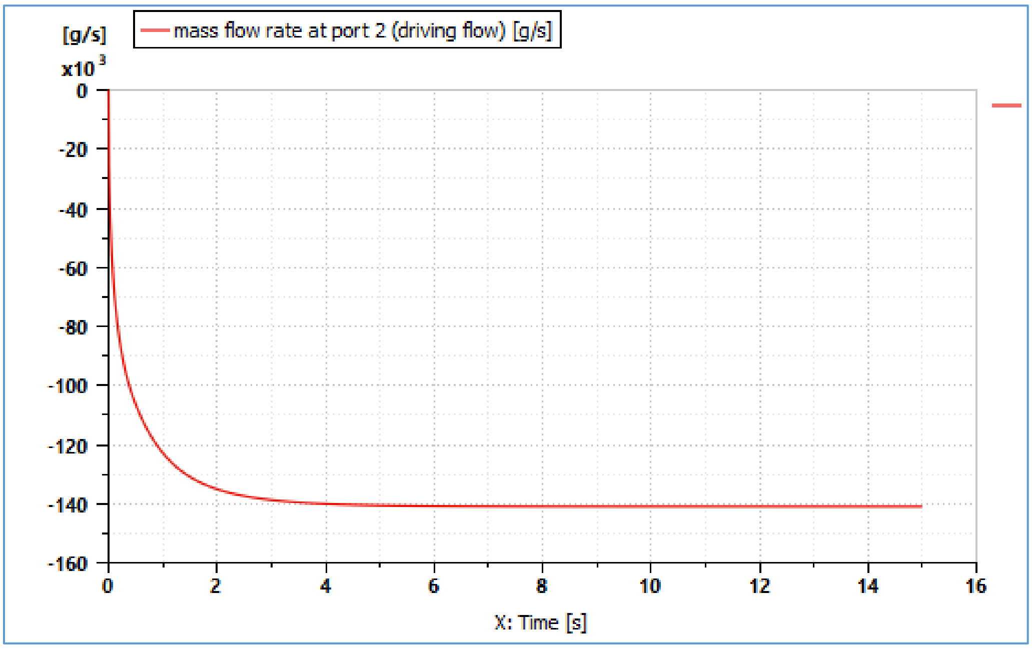


Figure 5.8: Second stage steam ejector mass flowrate in kg/s

6. CONCLUSIONS

In this thesis, high altitude test facility design and 1D liquid propellant rocket engine design was stated. Launch vehicle characteristics depend on generally rocket engine performance and test systems for sub components of rocket engine are in vital role. For the development process of the liquid propellant engines, NASA constructed the first altitude test facility and ESA constructed modified an altitude test facility for various rocket engines such as HM7 and VINCI. VINCI performance tests in altitude test facility were conducted in Germany and nowadays France's altitude test facility.

Results of the 1D simulated VINCI engine and performance in the 1D altitude test facility results compared with the experimental results and performance results according the related literature. Thrust performance, fuel mass flowrate performance, oxidizer mass flowrate performance, combustion chamber performance, vacuum chamber performance and specific impulse performance of the simulation results are lower than %5 error band. 2^{nd} stage ejector mass flowrate to create vacuum in desired error was found as %19.5 that system attitude is 3D can be different according to 1D 2^{nd} ejector stage. Simulated results are found to be in acceptable range that target error band was accepted as %10 for thrust, mass flowrates, specific impulse, vacuum and combustion chamber pressures. Modifications of the system components can be made to increase performance 2^{nd} stage ejector of the system according to the results.

In the future works, turbopump model of the engine can be created in detail with performance maps of pumps and turbine. 3D modelling and 3D CFD models can be created to examine all behavior of the components in detail. Construction of the altitude test facility and thermal considerations of the liquid propellant rocket engines can be added for the future works. Finally, digital twin model of the exact liquid propellant rocket engine model and 1D simulation models can be worked synchronously to increase real rocket engine performance.

REFERENCES

[1] **Turner, M. J. L.** (2009). *Rocket and Spacecraft Propulsion: Principles, Practice and New Developments / by Martin J. L. Turner*. Springer Berlin Heidelberg.

[2] **de Iaco Veris, A.** (2021). *Fundamental Concepts of Liquid-Propellant Rocket Engines / by Alessandro de Iaco Veris*. Springer International Publishing.

[3] **Dobrovolskiy, M. V.** (2016). *Sıvı yakıtlı roket motorları/ Mstislav Vladimiroviç Dobrovolskiy ; çeviren Gulzhanat Sakenova*. NDL Mühendislik Dış Ticaret Limited Şirketi.

[4] **Sutton, G. P.** (2017). *Rocket propulsion elements / George P. Sutton, Oscar Biblarz*. Wiley.

[5] **Edberg, D., & Costa, W.** (2022). *Design of Rockets and Space Launch Vehicles.* Reston, Virginia: American Institute of Aeronautics and Astronautics.

[6] **KELLER.** (2024, March). Series 21Y.

[7] **WIKA.** (2020, August). Cable thermocouple Mineral-insulated cable (MI cable) Model TC40.

[8] **KROHNE.** (2021, March). OPTIMASS 6000.

[9] **Anderson, J. D., & Cadou, C.** (2024). *Fundamentals of aerodynamics*. McGraw-Hill.

[10] **Brüel & Kjaer.** (2014, December). Miniature CCLD Accelerometer Types 4394 and 4397-A.

[11] **PCB PIEZOTRONICS.** (n.d.). SENSORS FOR LOAD & TORQUE. New York.

[12] **Schäfer, K., Zimmermann, H., & Kruhsel, G.** (2003). Altitude simulation bench for Vinci Engine. *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*. https://doi.org/10.2514/6.2003-5043

[13] **Schäfer, K., Zimmermann, H., Krühsel, G. & Kronmüller, H.** (n.d). P4.1 Test Facility for Altitude Simulation of VINCI Engine – Bench Development. *39th European Conference for Aero-Space Sciences (EUCASS)*

[14] **Raines, N., & Maynard, B.** (2008). The 26th AIAA Aerodynamic Measurement Technology and Ground Testing Conference. In *Altitude Testing of Large Liquid Propellant Engines*. Reston; NASA.

[15] **Ferrell, J.** (1967). Simulated Space Testing of Propulsion Units and Systems. *1967 (4th) Space Congress Proceedings*.

[16] **Dougherty, N.S.** (2010, July). *Liquid Rocket Engine Testing - Historical. Simulated Altitude Testing at AEDC*. Nashville; TN.

[17] **Moore, E., Waugh, I., Greig, A., Macfarlane, J., & Coxhill, I.** (2020). Overview of Rocket Testing at the Westcott Test Facility (2018-2020). *Space Propulsion 2020+1 Conference*.

[18] **Kim, T., Kim, W., Kim, S., Han, Y., & Ko, Y.** (2015). High altitude test facility for small scale liquid rocket engine. *Journal of the Korean Society of Propulsion Engineers*, *19*(3), 73–82.

[19] **Huzel, D. K., & Huang, D. H.** (1992). *Modern Engineering for Design of Liquid-Propellant Rocket Engines*. AIAA.

[20] **Alliot, P., Lassoudière, F., Fiorentino, C., & Ruault, J.-M.** (2006). Results of the VINCI Engine First Test Campaigns. AIAA 2006-4901. Proceedings of the 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Sacramento, California

[21] **Alliot, P., Delange, J.-F., De Korver, V., Sannino, J.-M., Lekeux, A., & Vieille, B.** (2019). Vinci, the European reference for Ariane 6 upper stage cryogenic propulsive system. *Progress in Propulsion Physics – Volume 11*. https://doi.org/10.1051/eucass/201911481

[22] **Durteste, S.** (2007). A Transient Model of the VINCI Cryogenic Upper Stage Rocket Engine. 43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Cincinnati, OH. AIAA 2007-5531. American Institute of Aeronautics and Astronautics.

[23] **Ariane Group.** (2020). *VINCI® ENGINE*. Paris; Ariane Group.

[24] **Goirand, B., Alliot, P., Barthoulot, J., & Bonhomme, C.** (2003). Testing the First Fuel Turbopump of the Vinci Engine. *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*. AIAA 2003-5069. American Institute of Aeronautics and Astronautics.

[25] **Lagnel, O.** (2018). Performance of vinci engine tests at PF52 test bench. *2018 SpaceOps Conference*. https://doi.org/10.2514/6.2018-2724

[26] **Greuel, D., Schäfer, K., & Schlechtriem, S.** (2013). Test facilities for Vinci®. *CEAS Space Journal*, *5*(1–2), 39–48. https://doi.org/10.1007/s12567-013-0043-8

[27] **Brodin, S., Steen, J., Ljungkrona, I., Edin, N., & Laumert, B.** (2005). Vinci engine development testing, a summary of turbine testing results. *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference &Amp; Exhibit*. https://doi.org/10.2514/6.2005-3949

[28] **Ellerbrock, H., & Ziegenhagen, S.** (2006). Liquid rocket engine test facility engineering challenges. *Acta Astronautica*, *59*(12), 1123–1134. https://doi.org/10.1016/j.actaastro.2006.04.013

[29] **Siemens Digital Industries Software.** (n.d.). *Help Guide*

[30] **Munson, B. R.** (2014). *Fundamentals of Fluid Mechanics*. John Wiley & Sons.

[31] **Internacional, E. A.** (2022, February 25). *Proosis modelling and Simulation Software*. EcosimPro. https://www.ecosimpro.com/

[32] **Siemens Digital Industries Software.** (n.d.). https://plm.sw.siemens.com/en-US/simcenter/systems-simulation/amesim/

[33] **Di Matteo, F.** (2010). *Modelling and Simulation of Liquid Rocket Engine Ignition Transients* (Doctoral dissertation, Sapienza University of Rome).

[34] **Jimenez Mena, D., Pluchart, S., Mouvand, S., & Broca, O.** (2019b). Rocket Engine Digital Twin – Modeling and Simulation Benefits. *AIAA Propulsion and Energy 2019 Forum*. https://doi.org/10.2514/6.2019-4114

[35] **Su, Q., Wang, J., Yan, M., Sun, Z., Zha, B., & Huang, W.** (2022). The dynamic model of the high-thrust orbital maneuvering engine with storable propellant. *Journal of Physics: Conference Series*, *2364*(1), 012014. https://doi.org/10.1088/1742-6596/2364/1/012014

[36] **Son, M., Kim, D., Koo J.** (2010). Analysis of Transient Characteristics for Turbopump-fed Liquid Propellant Rocket Engine in Start-up. *KSPE Spring Conference*

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