A Case Study On Propulsion System for Single Stage to Orbit Vehicle

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Declaration

We, Team Phoenix, hereby declare that the project entitled *"**A Case Study On Propulsion System for Single Stage to Orbit Vehicle**"*, submitted to Society for Space Education Research and Development (SSERD), is the record of original work done by us under the guidance of Mr.Mahesh Paneerselvam, Head of R&D, IPD SSERD.

Extracts of any literature which has been used for this project have been duly acknowledged, providing details of such literature in the references and this work has not been submitted for any degree, diploma or other similar titles elsewhere.

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Abstract

Space launches have always been one of the trickiest challenges. Precise calculations and various design changes are to be dealt with and weight and cost reduction has been one of the primary targets. One way to address both of these problems is to reduce the number of stages of a rocket. This can be achieved by using the concept of Single Stage to Orbit Vehicles (SSTO). These vehicles, although posing many challenges, can prove to be one of the most efficient methods of launching payload into orbit. This paper represents the studies done on various SSTO vehicles to obtain a conceptual design of a SSTO vehicle and the selection of the most suitable propulsion system along with certain modifications.
Single Stage to Orbit (SSTO) vehicles reach orbit from the surface of a body using only propellants and fluids without expending tanks, engines and other major hardware. This leads to subsequent cost and weight reduction. In this paper, we have done a detailed case study on tested SSTO's including the McDonnell Douglas DC-X, Rotary Systems ROTON, ARCA Haas, Reaction Engines Ltd. Skylon and Lockheed Martin VentureStar with our primary focus being the propulsion system. With various engine parameters in mind such as I_{sp}, thrust generated etc., this paper deduces an efficient model of a propulsion system for an ideal SSTO.

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SSTOV Single Stage to Orbit Vehicles

SSTO Single Stage to Orbit

MSTO Multi Stage to Orbit

RLV Reusable Launch Vehicle

HOTOL Horizontal Take Off and Horizontal Landing

VTVL Vertical Takeoff and Vertical Landing

LEO Low Earth Orbit

MECO Main Engine Cutoff

LOX Liquid Oxygen

LH_2 Liquid Hydrogen

GH_2 Gaseous Hydrogen

GO_2 Gaseous Oxygen

I_{sp} Specific Impulse

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Introduction

Single Stage to Orbit Vehicles (SSTOV) is a launch vehicle that can reach the orbit with just one stage or in other words without any staging. These vehicles reaches orbit from the surface of a planet/moon using only propellants and fluids and without expending tanks, engines, or other major hardware.The main projected advantage of the SSTO concept is elimination of the hardware replacement inherent in expendable launch systems.

1.1 | Single Stage to Orbit Vehicle

Generally, SSTOV concepts are cheaper and an easy feat when launched from gravitational fields of the Moon, Mars etc. But when it comes to The Earth, SSTO's are not that efficient. Even though there was a lot of scope for new design ideas and technology, the development of expendable MSTO Vehicles have put a nail on the coffin for SSTO's.
Usually (*but not exclusively*), SSTO's are referred as Reusable launch Vehicles because reusability is one of it's main advantages. This is due to the build and design of SSTO's. They usually do not have expendable parts which makes it easier and cheaper to reuse. There's no perfect SSTO launch till date, i.e., no launch vehicle has perfectly reached orbit without ejecting any of its components. But there are few launch vehicles, which reached Orbit with *Stage and Half Configuration* -

ATLAS – B (Sustaining Engine + Strap on Boosters)

SPACE SHUTTLE STS (Orbiter + External Tank + Solid Rocket Boosters)

The only perfect SSTO vehicle that ever flew (but on Moon) successfully was the Lunar module. To be more precise, the Lunar module's Ascent Stage is the perfect SSTO vehicle that took off from the moon surface to the orbiter (Command Module + Service Module) in lunar orbit. Apart from these, there are no semi-perfect nor perfect SSTO vehicles which were operational.

1.2 | Single Staging vs Multi Staging

- SSTOV don't carry extra weight or expendable weight. So reusability is considered cheaper in SSTO compared to MSTO.
- Since SSTO's don't carry expendable weight, change in mass ratio in the flight is very less compared to MSTO's since the only mass changing is of onboard fuel. Whereas there is a significant change to mass ratio of MSTO's in flight because they detach useless weight and send only payload to orbit.
  SSTO's send the whole vehicle to orbit.
- Research and development costs of SSTO's are very high since there is comparatively very less technical data available. MSTO's were considered more viable and efficient.
- Even though R&D of SSTO's costs more, the manufacturing costs and reusability costs of SSTO's are cheaper than MSTO's since MSTO's have expendable parts and cost more.
- MSTO's are very well tested and proven successful whereas the SSTO's are still under testing phase and we don't have much technical data SSTO's which make them unreliable for commercial usage.

1.3 | Challenges of SSTO's

- **Design Complexity :**
  Design of Single Stage to Orbit Vehicles (SSTOV) are more complex than expendable MSTO's since there are no detachable weights in it, all the components are to be put in a single vehicle. The heavier the vehicle is the more fuel required for the mission unlike MSTO which can be managed with staging and detachable mass. This is a real problem for the design engineers of SSTO's because there is not much scope for changes.
- **Research and Development :**
  SSTO's are still in the early phases of research, so it has very little technical data available on it. So there has to be intensive research on the field of SSTO's which involves a lot of time and money put in, a luxury which current investors don't have in the field of aerospace. This was also hit by the development of Multi staging vehicles which were proven better for the orbital missions. But because

of the demand for reusability and interplanetary missions, SSTO's are a better choice since you only have to deal with a single vehicle which is used for all round transportation.

- **Development of Reusable MSTO's :**
  As discussed above, one of the main strengths of SSTO's are they are mostly reusable so their long term usage costs will be comparatively lower than the expendable MSTO's. But the development of partially or fully reusable MSTO's like Falcon 9, Super Heavy etc., is a real problem for the SSTO's. With already proven MSTO's, if the main issue in currently reaching space efficiently i.e., reusability is resolved, it is going to be a checkmate for SSTO technology because the minimum funding and the very few companies working on it will be stopped. But the actual use of SSTO's is in interplanetary missions, because we will be needing single vehicle for carrying all the required instruments and food especially for human flights on a single vehicle for which SSTO's serves the purpose. .

- **Exhaust velocities provided by current chemical propellants :**
  For chemical fuels, the combustion properties determine the exhaust velocity. Unfortunately, the maximum velocities of chemical fuels top off at level that does not make it easy to get a single stage vehicle to orbit. The propellant combo of LH_2/LOX offers the highest v_e (exhaust velocity).
  This leaves the mass ratio as a parameter the rocket designer can control. One can maximize the velocity the rocket gains by minimizing the mass of the rocket after the fuel is burned. An SSTO vehicle, by definition, retains all of its hardware while only expending fuel. It does not drop mass like staged rockets. This in turn increases the mass of the vehicle after all the fuel is burned. Without the ability to drop mass, the m_f/m_i ratio remains small, making it difficult to achieve enough delta-v (change in velocity) to get to orbit. The supporting equation for this is the *Tsiolkovsky Rocket Equation*:

$$\Delta v = v_e * ln(m_f/m_i)$$

2

Case Study

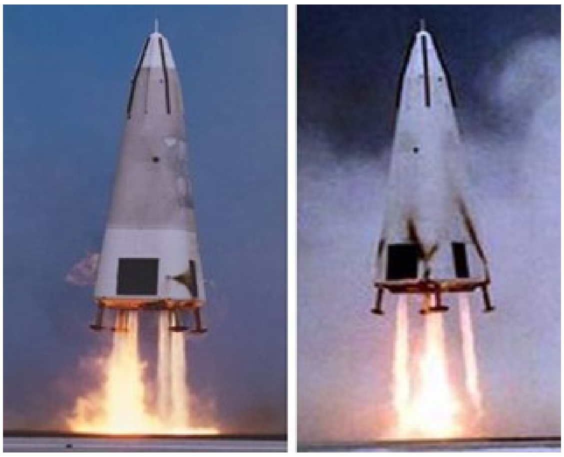
Many SSTO projects were proposed from the early 1960's to early 2000's. Although most of them were on paper, few managed to obtain funding and enter the testing phase. A list of the few popular and tested SSTO's were considered for this case study and are discussed below.

2.1 | DC - X (Delta Clipper Experimental)

The DC-X (Delta Clipper Experimental) was an unmanned prototype of a reusable SSTO launch vehicle built by McDonnell Douglass along with the Strategic Defense Initiative Organization (SDIO) of the US from 1991 to 1993.
Testing continued through funding of the US civil space agency NASA from 1993-1995. In 1996, the DC-X technology was completely transferred to NASA, which upgraded the design for improved performance to create the DC-XA.
The DC-X was never designed to achieve orbital velocity, but instead to demonstrate the concept of Vertical takeoff and landing. By the July of 1995 the DC-X had completed eight flights in two series, reaching 2500 m. On the eighth flight the aeroshell was cracked in a hard landing.

2.1.1 | Specifications

- Four RL10A-5 rocket engines, each generating 6,100 kgf thrust.
- Total Length: 14.00 m.
- Height: 12 m.
- Core Diameter: 3.05 m.



(a) DC-X

(b) DC-XA

Figure 2.1: Testing of DC-X & DC-XA

- Span: 4.10 m.
- Burnout mass: 9100 kg.
- Fuelled mass (with full load of propellants):18,900 kg.
- Liftoff Thrust: 223.00 kN.
- Propellants: Liquid oxygen (LOX) and Liquid hydrogen (LH_2).

Figure 2.2: LOX Tank

Figure 2.3: Thrust Structure

Figure 2.4: LH_2 Tank

2.1.2 | Propulsion System

The DC-X demonstrated the use of the first fully reusable LH_2/LOX propulsion system. The DC-X has successfully accomplished 11 hot firings and 3 tests in order to demonstrate technologies required for cryogenically fueled vertical landing vehicles.
The propellants used in DC-X are LOX and LH_2. The main LOX tank is placed forward which helps in improving vehicle stability and control during flight. The LH_2 tank is insulated with internal balsa wood insulation whereas the LOX tank is insulated with an external blanket.The propulsion system can be categorized into 3 sections which are

- Main Engines
- RCS (Reaction Control System)
- Vehicle Propulsion

The DC-X used four RL1OA-5 engines. These engines are the modified version of the existing RL-1O engine. The changes adapted in the RL1OA-5 engine include -

- Reduced nozzle expansion ratio for low altitude operation.
- Increased chamber length to regain some of the heat transfer surface lost.
- Modulating engine control valves for throttling.

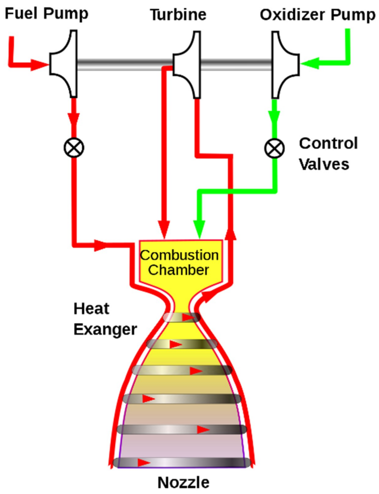


Figure 2.5: Expander Cycle used in RL10A-5 engine

The RCS system used in DC-X is an all gas hydrogen oxygen system.
GH_2 and GO_2 were used as the propellants for the RCS system. These propellants are stored in composite overwrapped bottles which are placed on the RCS pallet located in the center of the base region on the vehicle.
By using the GH_2 and GO_2 as propellants, the RCS could tap-off of liquid propellant tanks and convert propellant to high pressure gas storage on an operational vehicle. The four thrusters located at the base region of the DC-X produce approximately 450 pounds of thrust each.
LH_2 fill, feed, and vent systems, LOX fill, feed, and vent systems, and the helium pressurization and purge system constitute the propulsion system.
To load and to chill down the engines, free venting of LH_2 was base lined.
Capacitance point level sensors located along a mast inside the main tanks were used to achieve Propellant gagging. 8 sensors were placed in each tank and any four per tank could be activated for a flight test.
13 helium spherical storage bottles located in different areas of the vehicle which constitute the helium system.

- 3 bottles in the base of the vehicle.
- 8 in the inter-tank section.
- 2 in the avionics rack near the top of the vehicle.

The helium is stored in its ambient temperature with a volume per bottle of approximately 2.66 cubic feet. The high pressure GO_2 for the RCS and the high pressure GH_2 necessary for the hydraulic system accumulator are stored in similar spheres.
The lessons learned from health status and monitoring capability of the main engines demonstrated that there were other areas of the propulsion system which should have similar capability. One modification would be incorporating a real time on-board hydrogen detection system.

2.1.2.1 | RL - 10 Engine

Variant	Nozzles	Operational Vehicles
XRL-115 (RL-10)	1	Saturn - I
RL-10A-3-3A	2	Space Shuttle
RL-10A-5	4	Delta Clipper DC-X
RL-10C-1	1	Atlas V

Table 2.1: Various Models

Common elements of the RL-10 engine models are the use of liquid hydrogen and liquid oxygen propellants and also the expander cycle. In this cycle, the liquid hydrogen fuel is routed from the pump discharge to the combustion chamber, where it cools the thrust chamber jacket. The fuel is then directed to the turbine, where the heat absorbed from the jacket drives the turbine, which is then directly attached to the fuel pump rotor. The liquid oxygen pump is driven by a reduction gear arrangement.
In the RL-10 configuration, 360 thin-wall tubes are formed and flattened to form the primary nozzle. The 15,000 lb. thrust level of the RL-10A-1 was derived from the application of the centrifugal hydrogen turbopump. However, the lightweight turbopump created for the RL-10 accelerated at a much faster rate than anticipated. Chamber pressure was also increased, resulting in the first upgrading of engine thrust (to 16,500 lb.) since the RL-10A-l.

2.1.3 | Re-entry Configuration

The DC-X uses a nose first re-entry design with flat sides and large flaps. This configuration employs an attitude control thruster and retro rockets to control the descent. Finally, it rolls around and touches down on landing struts.

Figure 2.6: RL-10 Engine

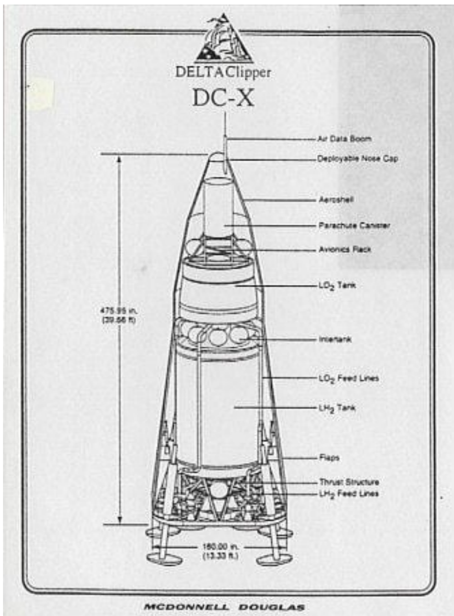


Figure 2.7: Components of DC-X

2.1.4 | Developmental Costs

Originally the DC-X was built in 21 months for a cost of 60 million dollars. This is equivalent to 102 million dollars in present-day terms. Comparing with Its counterpart LM X-33 which is 922 million dollars NASA + 357 million dollars Lockheed Martin.

2.1.5 | Testing Results

Flight	Launch Date	Duration (sec)	Altitude (m)	Description
1	August 18, 1993	59	46	Verified flight control systems and vertical landing capabilities.
2	September 11, 1993	66	92	Ascent and landing mode control and ground effects survey.
3	September 30, 1993	72	370	180 degree roll; aerostability data.
4	June 20, 1994	136	870	Full propellant load; radar altimeter in control loop.
5	June 27, 1994	78	790	In-flight abort after gaseous hydrogen explosion; vehicle demonstrated autoland capabilities.
6	May 16, 1995	124	1330	Continued expansion of flight envelope; constant angle of attack.
7	June 12, 1995	132	1740	First use of reaction control system thrusters; AOA from 0 to 70.
8	July 7, 1995	124	2500	Final flight; demonstrated turnaround maneuver; aeroshell cracked during 14 ft/s landing.

Table 2.2: Delta Clipper-Experimental (DC-X) Test Program

During testing, one of the *LOX* tanks had been cracked. When one of the landing struts failed to extend due to a disconnected hydraulic line, the DC-XA fell over and the tank leaked. The *LOX* from the leaking tank led to a fire which severely burned the DC-XA, causing extensive damage at such a level that repairs were impractical.

Flight	Launch Date	Duration (sec)	Altitude (m)	Description
1	May 18, 1996	62	244	First flight of the DC-XA; aeroshell caught fire during slow landing.
2	June 7, 1996	64	590	Maximum structural stresses with 50% full *LOX* tank.
3	June 8, 1996	142	3140	26-hour rapid turnaround demonstration; new altitude and duration record.
4	July 31, 1996	140	1250	Landing strut 2 failed to extend; vehicle tipped over and *LOX* tank exploded; vehicle destroyed.

Table 2.3: Delta Clipper-Experimental Advanced (DC-XA) Test Program

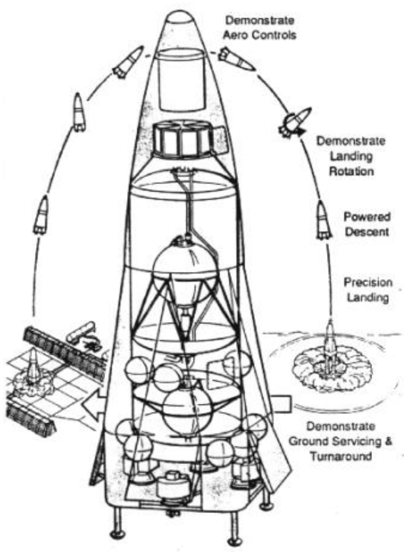


Figure 2.8: Expander Cycle used in RL10A-5 engine

2.1.6 | New technologies tested on the DC-X

NASA agreed to take over the program after the last DC-X flight in 1995. In particular, the oxygen tank was replaced by a lightweight aluminum-lithium alloy tank, and the fuel tank was replaced by a newer composite design. The upgraded vehicle DC-XA then resumed flight in 1996.

The DC-XA was operated by the NASA and the Department of Defense under the

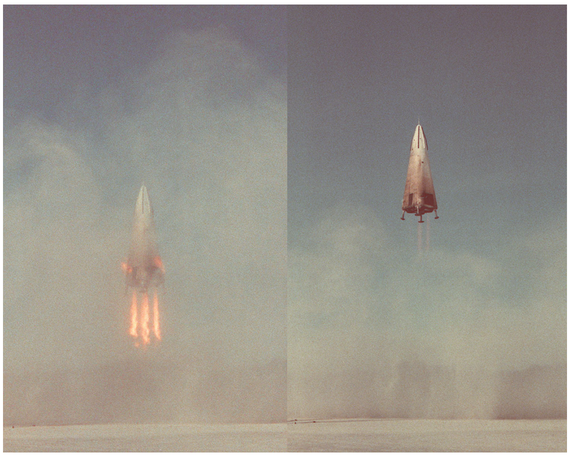


Figure 2.9: DC-X under action. Notice the yellow exhaust while its landing which is because of its low throttle settings, which burns at lower temperatures.

Reusable Launch Vehicle (RLV) program. This had a lightweight graphite-epoxy liquid hydrogen tank, an advanced graphite/aluminum honeycomb inter-tank, an aluminum-lithium liquid oxygen tank and an improved reaction control system from the aerojet. These improvements reduced the dry mass of the vehicle by 620 kilograms.

2.1.7 | Reasons for Cessation

Despite multiple successful flights and very less failures, it was under constant pressure from X-33 which was jointly performed by NASA and Lockheed Martin. Apart from that, there was intense political tussle between these two.
This was followed by NASA's Disapproval to fund for the upcoming DC-X's testing and research facility. Instead, NASA focused development on the Lockheed Martin VentureStar, which was homegrown project. This lead to a complete shutdown.

2.2 | ROTON ATV (Atmospheric Test Vehicle)

The Roton was a concept for a completely reusable SSTO launcher. The American Rotary Rocket Company developed this unique manned SSTO. The Roton was a space vehicle design intended to provide rapid access to orbit.
The Roton was a fully reusable SSTO. with Vertical take-off Vertical Landing (VTVL) configuration. It was designed to transport up to 3200kg to orbit.

Figure 2.10: Conceptual Design

2.2.1 | Specifications

Payload	3200 kg (7,000 lb)
Gross Mass	180,000 kg (390,000 lb)
Height	19.50 m (63.90 ft)
Diameter	6.70 m (21.90 ft)
Apogee	300 km (180 mi)
Cargo Diameter	3.66 m
Cargo Height	5.08 m
Fuel Capacity	372,500 lb (169,000Kg)
Specific impulse	340 sec (3.3 km/s)
Burn time	253 sec

Table 2.4: Roton Specifications

2.2.2 | Structure

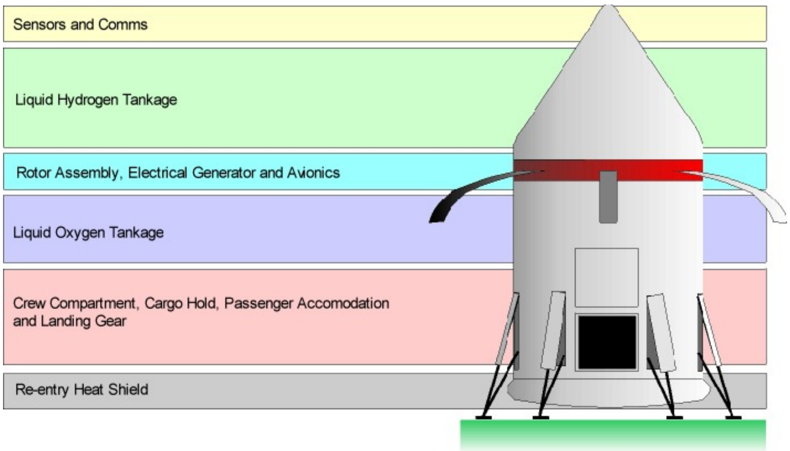


Figure 2.11: Structural Overview.

The Roton ATV structure mainly consists of five parts :

- Rotor System
- *LOX* Tank
- Cargo Compartment
- Crew Cabin
- Kerosene Tank

Water was used to cool the airframe. The present vehicle baseline uses composite skin for the fuselage structure. Composites have the advantage of high modulus, high tensile strength and a density lower than conventional aircraft aluminium.

2.2.3 | Mission Details

- During take-off, the blade is accelerated to control the speed, and collective pitch is added to the rotor blades. Initially, the vehicle is propelled almost entirely by the aerodynamic forces generated by the rotor.
- This is a very efficient operating mode for the Roton since the aerodynamic thrust delivered by the rotor generates about 5-10 times the thrust of the rockets which drive the rotor.

- As the vehicle accelerates blade pitch is increased, and because of this simultaneous pitch change of the rockets at the blade tips, the rockets begin to generate a larger amount of the overall axial thrust.
- At about 35,000 ft, the vertical speed rises to approximately 700 fps and the Roton makes its transition to pure rocket flight since the atmosphere has become too thin to provide useful assistance.
- At this point the blades are pitched almost vertically, much like a feathered propeller, and the rockets are throttled up. The vehicle continues in this manner until burn-out.
- As the vehicle reenters the atmosphere and aerodynamic forces increase, the rotor automatically rotates much like a helicopter during its vertical descent.
- Drag can be modulated by rotor coning, and can be controlled by a simple mechanical controller which adjusts blade pitch angle relative to the coning angle.
- This mechanism has been tested in operational systems and has shown itself capable of smooth rotor drag modulation from speeds of mach 3.5 to zero with no tendency for rotor over-speed.
- As the vehicle approaches the ground, the pilot or flight computer performs a collective flare to reduce vertical speed to near zero before touchdown.
- It is possible that the tip rockets will be restarted at low thrust so the Roton can be hovered or maneuvered before final touchdown.

2.2.4 | Propulsion System

Roton was to be powered by 8 fastrac derived engines. It will be quite different from the Space Shuttle Main Engine, which was designed at Marshall in the 1970s and is considered by many to be the world's most sophisticated reusable rocket engine.
While the concepts for the Fastrac engine have been around for decades the actual technological development and design began only in early 1996 and the engine's first flight was planned for late 1999. The simple, robust, easy-to-build engine is a component of the Low Cost Technologies effort, one element of NASA's Advanced Space Transportation Program managed at Marshall. This program is developing technologies which will dramatically reduce the value of going to space.

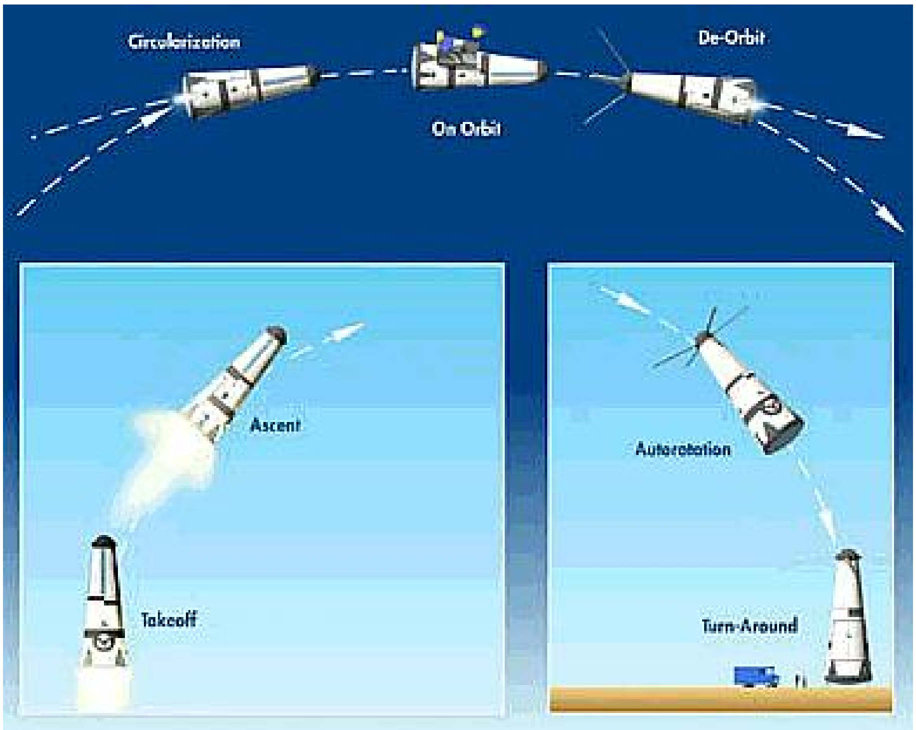


Figure 2.12: ROTON Mission Phases

Figure 2.13: Fastrac engine hot-fire test at Marshall Center (NASA/MSFC)

2.2.4.1 | Working of Fastrac Engine

- The Fastrac Engine is fueled by a mix of *LOX* and kerosene. Kerosene doesn't provide an equivalent impulse as hydrogen, which mixes with *LOX* to fuel the spacecraft - but is cheaper and easier to handle and store.

Figure 2.14: Fastrac Engine

- The engine is started with a hypergolic igniter. After the kerosene is injected, the propellants are supplied to the gas generator and thrust chamber assembly for mixing and burning.
- The Fastrac turbopump features two pumps - one for Fuel & other for Oxidiser. This turbopump also supplies the required chamber pressure.
- It uses the ablative cooling mechanism to cool the engine components.
- Nearly all of the engine's parts are reusable. Except the ablative chamber nozzle and thus the hypergolic ignition cartridge which may get replaced after each flight.

2.2.4.2 | Engine Specifications

No. of Chamber	1
Fuel	RP-1
Engine Height	1.21m (4 feet)
Engine Weight	2000 pound (910Kg)
Dry weight	910 kg
Oxidiser	Liquid Oxygen
Mixture ratio (O/F)	0.30
Fuel Feed System	Pump fed
Cycle	Gas generator cycle
Specific Impulse (in vacuum)	314 sec

Thrust (in vacuum)	270 KN
Chamber pressure	633 psi
Combustion temperature	1600R / 615.73C
Area ratio (Nozzle)	30:1
Area ratio	15:1
Expansion ratio	30:1

Table 2.5: FASTRAC Engine Description

2.2.5 | Testing

- The first test flight of the roton atv happened on july 23, 1999. During the test, the atv performed three takeoff and landing maneuvers to demonstrate the crew's ability to regulate the vehicle during the touchdown phase of the approach . During the 4 minutes and 40 seconds of the flight, the atv flew at an altitude of roughly 8 feet (2.4 meters).

- The second flight took place on september 16, 1999. The changes helped to increase thrust output of the blade mounted tip rocket and installation of an auto-throttle controller to maintain the rotor rpm at a set rate.

- The third and final flight of the roton atv took place on october 12, 2000. This was the test of the atv in transitional flight. The vehicle flew 4,300 feet (1,310 meters) down the line . And the top speed of 53 mph (85 km/h) was measured in the test.

- A planned fourth flight, during which the vehicle would fly to an altitude of 10,000 feet (3,050 meters) before throttling back and returning for a soft landing, was canceled due to a scarcity of funding and safety concerns.

2.2.6 | Reasons for Cessation

After the three successful tests of ROTON ATV with excellent results, the project was cancelled due to lack of funding.

2.3 | ARCA

ARCA space is an aerospace company located in the Valcea county, Romania which is also known as the Romanian Cosmonautics and Aeronautics Association. The company builds SSTO rockets, high-altitude balloons, and unmanned aerial vehicles. It was founded in 1999 as a non-Governmental and non profit organization by Dumitru Popescu and other like-minded aeronautics enthusiasts. ARCA focuses on developing aerospikes, water-based, electric rocket Launch Assist System (LAS) and also the aerospike Haas rocket. The ultimate aim of ARCA is to use clean and cost effective technologies for space travel and commercial launch applications.

2.3.1 | Launch assist system (LAS) system : Water based electric rocket

Polluting, explosive, corrosive, toxic, carcinogenic propellants are currently used in typical model rockets. A single launch of a rocket sends the same amount of toxic chemicals into the atmosphere as 1 million cars running at the same time. To avoid using polluting propellants, ARCA developed an electric, water-based rocket that serves as a first stage, using water as the propellant for launch vehicles, allowing for a 25% reduction in polluting propellants and a 30% increase in payload capability while remaining pollution-free. The LAS is not only clean, but is also safe and cost effective. It is built for two types of engines namely the classical bell nozzle engine and the linear aerospike engines.

2.3.2 | Tested Models of LAS

ARCA compared the two engine designs by testing the conventional engine first and then the aerospike engine for the LAS on the same stand, with the same tank and feed

(a) Cap1

(b) Cap2

Figure 2.15: ARCA's LAS

system, similar pressures, and same sensors.

Figure 2.16: Aerospike engine & the classic bell-shape nozzle engine during tests.

We were able to draw an initial conclusion that the aerospikes are better than the bell-shaped nozzle engines at sea level based on the first aerospike test, as follows :

Parameter for the test	**Classic Engine**	**Aerospike Engine**
LAS 25D dry weight (kg)	748	1025
Engine dry weight (kg)	48	184
Test pressure (bar)	6	5.2
Average I_{sp} (s)	17	20
Thrust (kN)	20.6	23.5

Table 2.6: Comparison between classical and aerospike engine for LAS.

2.3.3 | Problem faced by classical bell shaped nozzles

A traditional bell-shaped nozzle is effective at only one altitude, usually at or slightly above sea level. After the sea level, the engine starts losing its efficiency due to the decrease in atmospheric pressure. This problem arises due to the difference between exit pressure and atmospheric pressure, resulting either in under-expansion or over-expansion of the exhaust.
This led to the rise of aerospike engines as it unlocks virtually unlimited expansion ratios, thus significantly increasing the specific impulse of the engine at high altitudes.

2.3.4 | ARCA's Haas rocket

The Haas rocket series, named after Austrian-Romanian rocket scientist Conrad Haas, includes everything from a small orbital launcher capable of launching 40 kg of cargo to a hefty launcher capable of launching 60 tonnes of payload to LEO.
It is a vertical launch SSTO vehicle. There are two ways to use the Haas rocket. One option is as a standalone SSTO rocket, and the other is as a booster in conjunction with the Launch Assist System to dramatically increase the rocket's cargo capability and specific impulse. The advantage of the rocket is that it eliminates the need for additional upper stages as it can be refueled in orbit and re-utilize its aerospike engine.
This rocket is extremely simple and therefore affordable to construct and operate and also this rocket eliminates risk in staging since it is a SSTO. It is able to reach orbit in less than 24 hours from the moment of launch decision. Working in conjunction with the Launch Assist System, the rocket will boost its payload capabilities up to six times.Due to this conditional launch, these rockets can be used in emergency situations for crewed vehicles since they reach orbit in 24 hours from the launch.

Figure 2.17: Propellant Overview.

Propellant used in Haas

- Hydrogen peroxide (Oxidizer)
- Kerosene or RP-1 (Fuel)

2.3.5 | Haas engine

There are two types of engine which can be used by the HAAS rocket. They are:

- Executor Rocket Engine
- Venator Rocket Engine

Figure 2.18: Venator (left) and Executor (right) rocket engines nozzles and chambers.

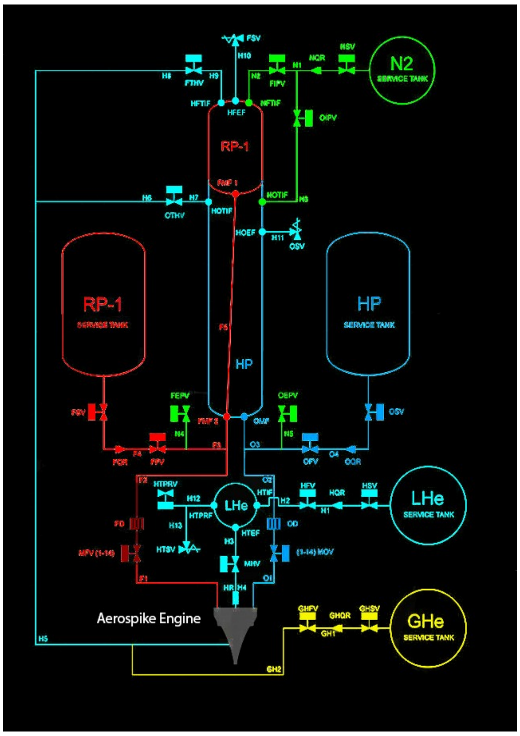


Figure 2.19: Process flow diagram of Haas.

2.3.5.1 | Executor rocket engine

Executor is a liquid fueled rocket engine. It has an open cycle gas generator combustion engine that uses liquid oxygen and kerosene as propellants.The maximum thrust produced is 24 tons. Most of the materials are made up of composite materials and aluminium alloys.The combustion chamber and the nozzle are made of two layers. The phenolic resin reinforced with silica fiber pyrolizes endothermically in the combustion chamber walls. This process will release gases like oxygen and hydrogen, leaving a local carbon matrix. The gases initially spread through the carbon matrix and finally reach the internal surface of the wall where they meet the hot combustion gases. It will then act as a cooling agent. The engine has a cooling system that injects coolant into the interior walls.Two hydraulic pistons gimbal the engine, which is powered by kerosene from the pump exhaust system.

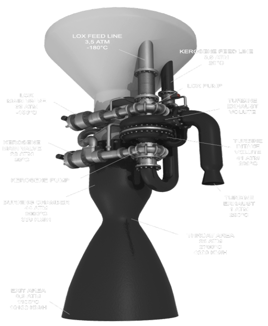


Figure 2.20: Executor Rocket Engine.

Technical Details of Executor Rocket Engine

Diameter	0.7 m
Length	2.2 m
Fuel	*LOX* + Kerosene
Burn Rate	85 kg/sec
Weight	250 kg
Ground Thrust	24 tons force
Specific Impulse (vacuum)	312 sec
Ground Thrust	20 tons force
Vacuum Thrust	24 tons force
Thrust Weight Ratio	110

Table 2.7: Executor Rocket Engine

2.3.5.2 | Venator Rocket Engine

Venator is a liquid-fueled, pressure-fed rocket engine. It is used to power the Haas rockets' second stage. The propellants used are Liquid oxygen and kerosene. It has the capacity to produce a maximum thrust of 2.5 tons of force. Majority of the engine components are made of composites. These materials are cost efficient and also reduce the weight of the overall component.The phenolic resin reinforced with silica fiber pyrolizes endothermically in the combustion chamber walls, releasing gases like oxygen and hydrogen, leaving a local carbon matrix.The gases spread through the carbon matrix and reach the inner surface of the wall where they meet the combustion gases and act as a cooling agent.Furthermore, the engine is provided with a cooling system that injects on the inner walls. The Venator rocket has no vanes on most of the pipes. It instead uses burst disks between the tanks and engine. The second stage is pressurized at 2 atm during lift-off and after the first stage burn-out, the second stage is going to be pressurized at 16 atm. At that pressure the disks will burst and therefore the fuel will flow through the engine. The turbine rotation speed is 20.000 rpm.The intake gas temperature is 6200 °C. The second stage is spin stabilized at 60 rpm, immediately after staging. This (spin) is done by using four helium reaction control systems.

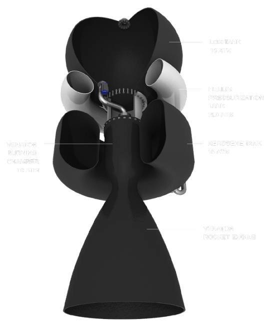


Figure 2.21: Venator Rocket Engine.

Technical details of Venator rocket engine

Diameter	0.8 m
Length	1.8 m
Fuel	*LOX* + Kerosene T1
Burn Rate	8.1 kg/sec
Chamber Pressure	10 atm
Weight	70 kg

Specific Impulse (Vacumm)	317 sec
Vacumm Thrust	2.5 tons force

Table 2.8: Venator Rocket Engine

2.3.6 | Advantage of using LAS in Haas 2CA

ARCA's Haas 2CA SSTO rocket has the capability to put 100 kg into low earth orbit in a single stage for a take-off mass of 16,000 kg. In order to accomplish this, 9000 kg of water for the LAS 25R booster would be required.This would offer a reduction of mass of the Haas 2CA rocket from 16t to 4t, without affecting payload capability. Another example is increasing the payload capabilities of the rocket from 100 kg to 700 kg by adding a 40t LAS 200E booster to the existing design.

2.3.6.1 | Specifications of Haas

Length	16 m
Diameter	1.5 m
Empty Mass	550 kg
Launch Mass	16,290 kg
Payload Mass	100 kg
Number of stages	1
Engine type	Linear aerospike
Engine feed	Pressure fed
Number of chambers	16
Spike cut	80%
Nozzle expansion ratio	80
Cooling type	Ablative + RP-1 film
Oxidizer	Hydrogen Peroxide
Fuel	RP-1
Burning time	272 s
Total thrust at sea level	22,920 kgf
Total thrust in vacuum	33,500 kgf
Sea level impulse	230 s
Vacuum impulse	314 s

Total HTP flow rate	88 kg/s
Total RP-1 flow rate	12 kg/s
Total propellant flow rate	100 kg/s
Mixture ratio including the film cooling	7.46:1
Propellant tanks pressure	20 barg
Chamber pressure	16 barg

Table 2.9: Haas Specifications

2.4 | SKYLON

The SKYLON spaceplane is a SSTO concept vehicle developed by the Reaction Engines Ltd (UK). The initial D-1 configuration was designed to take off and land on a runway delivering 15 MT of payload into LEO. SKYLON is an unmanned and uses innovative dual-mode SABRE engines. There were many designs in the SKYLON project including C2, A4, D1 and C1, with the C1 being the final version.

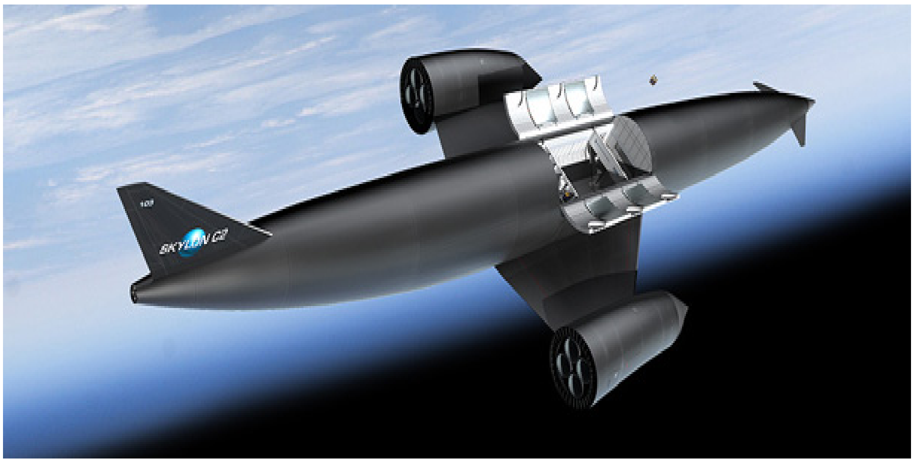


Figure 2.22: Concept Design.

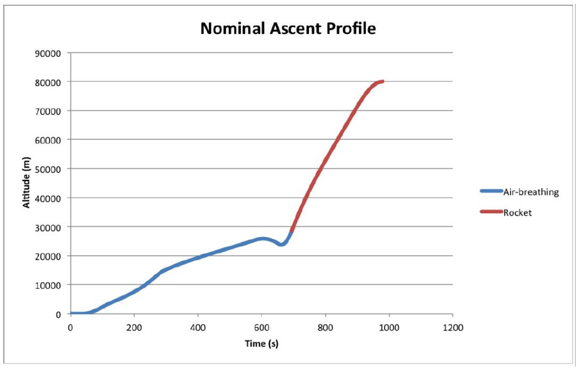
The conceptual idea behind SKYLON is that it is an aircraft like "spaceplane" that will take off from a runway, fly into orbit, perform missions such as launch satellites, or deliver crew and supplies to space stations, before re-entering the Earth's atmosphere. SKYLON will be a fully reusable space vehicle which will be capable of 200 operational flights. It is 84 m long, with 25 m wingspan and weighs 275 tons at take-off. The nominal payload that can be launched weighs approximately 12 tons.

2.4.1 | Targets & Mission Details

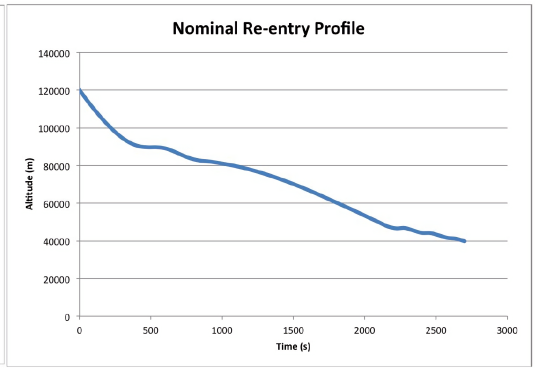
SKYLON targeted that C1 would meet some requirements, which are:

- 200 operational flights per vehicle
- 1 % abort rate per mission
- 1:20,000 loss rate per mission
- 48 hours turnaround time

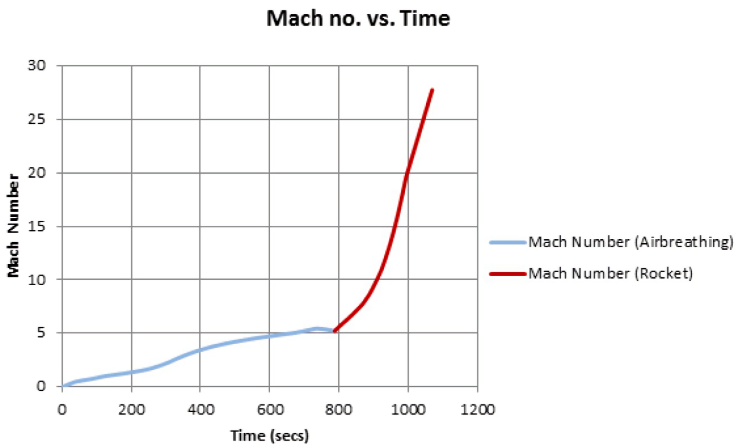
Flight Mission Details



(a) Ascent Profile



(b) Re-entry Profile



(c) Mach No. vs Time

Figure 2.23: Graphs Depicting the Flight Profile Of SKYLON

- Step 1 :
  After takeoff, the vehicle accelerates and climbs for 694 seconds on its designated path. (approx. 11½ minutes), by which time it has reached an altitude of 26 km and a speed of Mach 5.1.

- Step 2 :
  The engine then transitions to rocket mode to travel the last 80 kilometres to MECO, where it will be placed on a transfer orbit to attain the appropriate circularised orbit at apogee. The vehicle climbs quickly under rocket power, and the main engine shuts down after another 285 seconds (434 minutes), leaving SKYLON in an 80 by 300 kilometre transfer orbit.

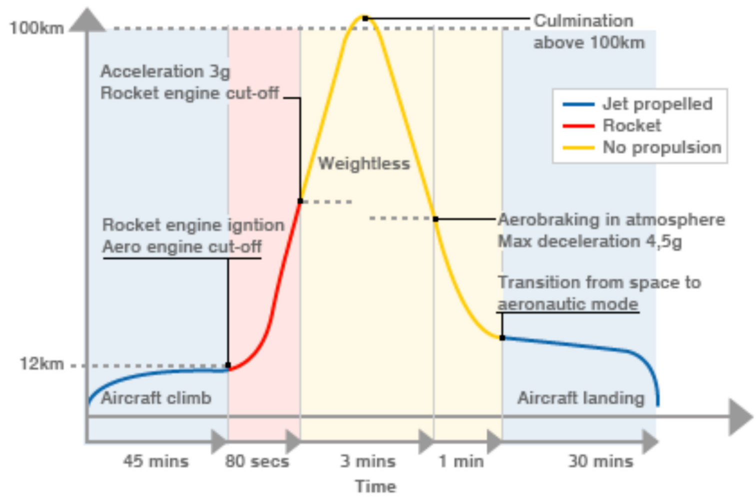


Figure 2.24: Typical Sub-Orbital Flight Profile of SKYLON.

- Step 3 :
  The SOMA thrusters will then be used to maneuver the vehicle into the appropriate orbit.

- Step 4 :
  The spacecraft will maneuver to reduce heat loads and follow strict glide range for descent and landing on a runway at the spaceport. The estimated re-entry interface is 120 kilometers.

2.4.2 | Design

The SKYLON is inspired from the design characteristics of the space shuttle and the HOTOL project. The design of SKYLON was done in a way to avoid all pitfalls of the HOTOL project.
The fuselage is long and narrow with the unique delta wings located roughly at the midpoint. The mainframe is constructed of Titanium reinforced with Silicon Carbide Fiber, due to relatively easy joining operation as well as high strength and operating temperature range. The propellant tanks are constructed of non-structural aluminum to save weight.
The Aeroshell, which is a rigid heat shielding shell that not only protects the vehicle from overheating during reentry but also helps decelerate the vehicle, was 0.5mm thick constructed of silicon carbide reinforced glass ceramic called System2.
Some areas of the SKYLON could get hotter than the others with temperature exceeding

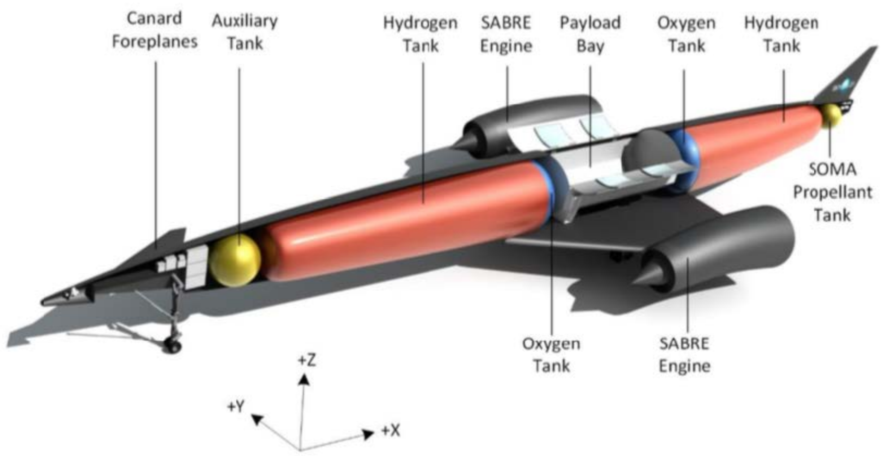


Figure 2.25: Internal Schematic.

over 1400 K. To counter this issue, water based cooling is provided to those areas.
The payload bay is situated right between the wings in the middle of the fuselage. The engines are placed in nacelles on the wingtips. The orbital positioning components along with the corresponding cryogenic propellant tanks are mounted on the frame.
As mentioned before, SKYLON was based on the HOTOL (Horizontal Take-Off and Landing) project. According to the earlier HOTOL project, the vehicle would have large delta wings starting from the middle of the fuselage extending to the aft section of the vehicle along with engines placed at the very end of the vehicle.
This revealed large issues with stability and the vehicle going in and out of trim condition as the speed increased. This was due to the center of pressure shifting ahead of center of gravity. SKYLON sought to reduce this by modifying the wing design and engine placement.

2.4.3 | Control Surfaces

In the atmosphere, the major control surfaces are canards for pitch, ailerons for roll, and a rudder for yaw. Deferential engine throttling and engine gimbaling take over during the rocket-powered flight until reaction control thrusters take over at MECO. The reentry will be primarily guided by the control surfaces, taking advantage of aerodynamic forces. Re-entry might be assisted by the RCS as well. The RCS was also called SKYLON Orbital Maneuvering Assembly (SOMA) had the following characteristics :

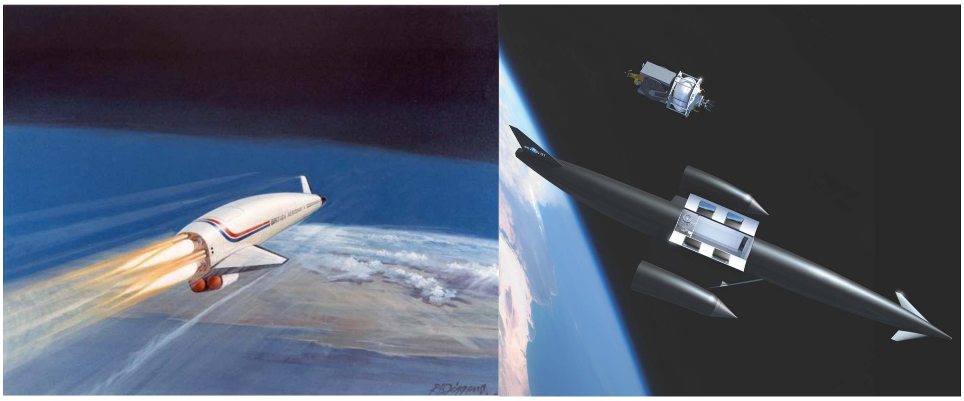


Figure 2.26: Comparison with HOTOL.

Thrust	40 kN
Chamber pressure	90 bar
Mass	102.5 kg
Specific Impulse (Isp)	465 s

Table 2.10: SKYLON Orbital Maneuvering Assembly (SOMA)

2.4.4 | Undercarriage & Braking

SKYLON would land and take-off from a specially built 5.5 to 5.9 km runway. As regular landing gears would fail at SKYLON's lading speed, SKYLON uses special landing gears that are sized and capable of rotation at 0.45 Mach.
In case of an aborted takeoff, the kinetic energy required to stop the plane, 3.24x109 J, would require a massive conventional disk braking system, calculated as 4000 kg. Rather than the conventional approach, the brakes have been undersized and use cooling water to remove the heat energy. The braking system carries 1200 kg of water to be blown through the brakes in the event of a runway abort and vented off as steam. After a successful takeoff, this extra water is dumped to reduce weight. This leaves the effective cooling and braking mass to be 515kg.

2.4.5 | SABRE Engine

The SABRE engine system is the most significant and unique part of SKYLON. SABRE is the combination of both Air breathing Engine & Rocket Engine.
Using Traditional Engine for the entire flight goes in vain for a SSTO Vehicle because of

large weight of stocked oxidiser.
So, having the weight of the oxidiser reduced by using environmental Oxygen as oxidiser for combustion process like air breathing engines, we can evolve from Single use Multistage Launch Vehicle to Multi use Single Stage Launch Vehicle.
Other uses of SABRE involve reduction in cost because when in atmosphere, it is used as Air breathing Engine, which suck oxygen directly from atmosphere and ultimately, reliable & responsive space exploration with increased payload allows aerospace vehicle to cruise at low hypersonic speed (Mach 5) within the earth's atmosphere.Ltd (2013)

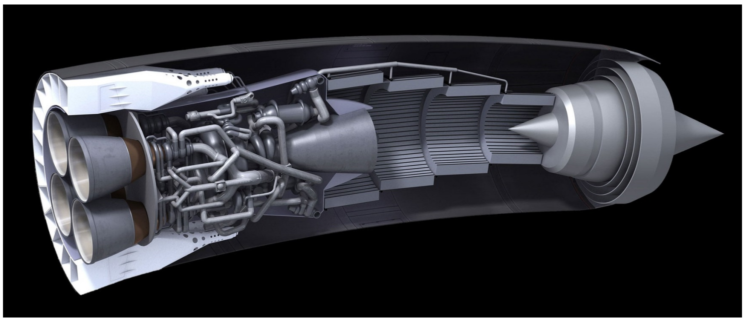


Figure 2.27: SABRE Engine Internals.

SABRE stands for Synergistic Air Breathing Rocket Engine and as the name suggests it operates in 2 modes, i.e., Air breathing mode and Traditional Rocket mode.
In the primary mode, it sucks Oxygen form Atmosphere to work as oxidiser to burn with stocked liquid hydrogen and once the Launch Vehicle achieves low hypersonic speed (Mach 5), it shifts to Traditional rocket mode by using pre-stocked oxidiser which is Liquid Oxygen on board.

2.4.6 | SABRE Engine Schematic

The same Combustion chamber & Nozzle are made to operate in both the modes by making it possible to synergize Turbo jet and Rocket Engines. Main components involved in SABRE are -
Translating Axisymmetric Shock cone, Pre-cooler, Compressor, Helium loop, Bypass thrusters and Nozzle.

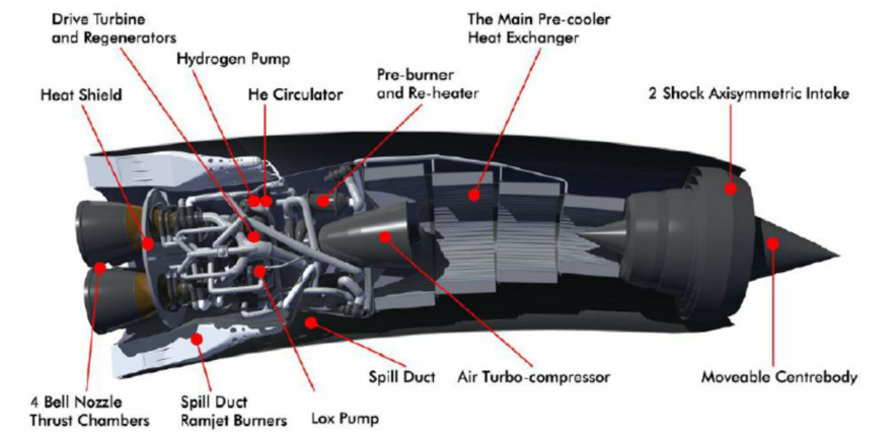


Figure 2.28: SABRE Engine Internals.

Components - Principles & Working -

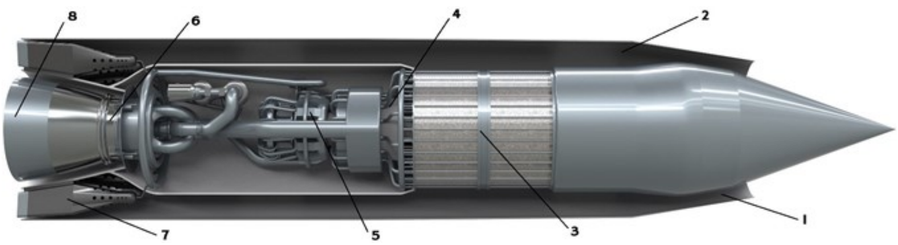


Figure 2.29: SABRE Engine Schematics.

1. Supersonic Intake :
   A simple translating axisymmetric shock cone inlet, at front of the engine, slows down the sucked environmental air to subsonic speeds by means of shock reflections and can operate at speeds above Mach 5. The intake is closed for rocket mode.

2. Nacelle :
   Heavy Ni and Cu based materials are generally used in jet engines to deal with extreme temperatures, whereas in SSTO's, usage of heavy materials increases the weight of the vehicle. So, the SABRE uses advanced lighter materials designed to withstand extreme temperature conditions.

3. Pre-Cooler :
   SABRE quickly cools the entering hot airstream (1000 °C) to ambient temperature

(using He loop) at a rate of 1000 °C to -150 °C in just 0.01 sec, allowing it to run at faster speeds than current engines. SABRE uses a methanol-injecting 3D-printed dicer to minimize liquefaction and the formation of ice during this procedure.

4. Compressor :
   It is a modified Turbo compressor, similar to those found in traditional jet engines, but it operates at an abnormally high-pressure ratio, aided by the precooled air's low temperature. The compressor compresses precooled air to a high pressure of 140 atm, which drives the rocket combustion chamber to ignite with stored liquid hydrogen (LH_2), allowing runway start and takeoff.

5. Engine Core :
   Powers SABRE during air-breathing flight. The hot Helium (He) from the pre-cooler is recycled by cooling it in a heat exchanger with LH_2, and the heat absorbed by He from the incoming air is used to power various components of the engine, resulting in significant efficiency gains by reducing fuel consumption.

6. Rocket Engine :
   Provides thrust to power the engine for space access during Traditional Rocket mode.

7. Ramjet Burners :
   Increases overall engine efficiency by generating extra thrust with excess air.

8. Nozzle :
   It operates both in the atmosphere and in space thereby reducing weight and complexity.

2.4.7 | Preliminary Heat Exchanger Testing :

Reaction Engines Ltd. was able to test an integrated, full-size pre-cooler with frost control in the mid-2000s. The pre-cooler was able to keep the temperature below 1000°C for more than 5 minutes. The test covered over 200 runs, and the pre-cooler performed admirably in terms of thermo-mechanical integrity throughout.

2.4.8 | Performance

- Reaction Engines Ltd. have designed SABRE ENGINE with Thrust to weight ratio (TWR) of 14, which is a far higher value than jet engines & Scramjet engines (whose TWR are 5 & 2 respectively).

Figure 2.30: Preliminary Heat Exchanger.

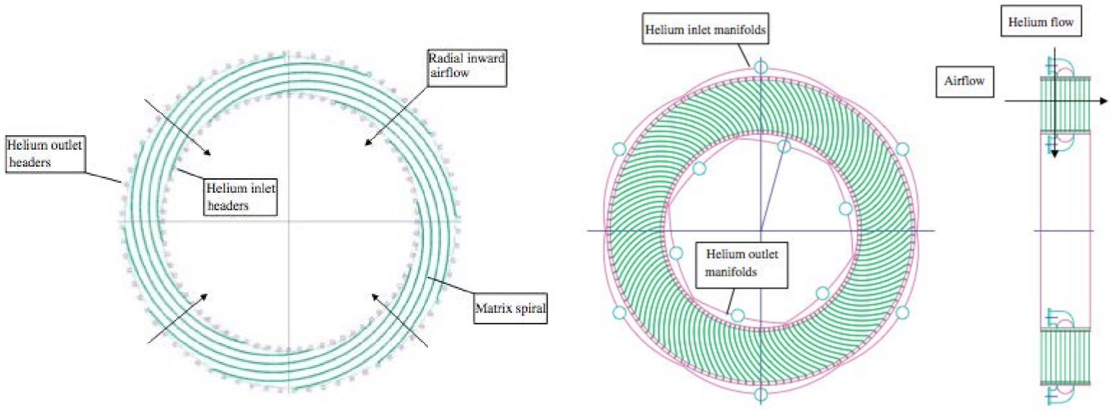


Figure 2.31: Working of Helium Loop inside the Heat Exchanger.

- This is by virtue of properties of SABRE engine – denser and cooled air from Pre-cooler require less compression, and this low air temperature of air allows the use of lighter alloys in engine.
- SABRE engine achieves Specific Impulse as high as 3500 seconds within earth's atmosphere, which again is far higher when compared to any other rocket propulsion system.
- The combination of the qualities, like fuel efficiency and a lightweight engine, allows an aerospace vehicle to perform an SSTO approach with air breathing mode up to a height of 17.709 miles and a speed of Mach 5.14.
- The greater weight of Skylon's wings negates the gains in overall efficiency and

intended flight plan due to additional weight of equipment that will stop working during rocket mode or close cycle mode.

- Furthermore, the ability to provide high thrust at speeds ranging from 0 to Mach 5.5, as well as high thrust across the entire flight from the ground to very high altitude, makes the SABRE engine a dominant alternative for launching vehicles above current launching vehicles.

2.5 | LOCKHEED X-33 and VentureStar

SSTO has always been a dream for mankind. The very first rocket developed by us humans the prestigious V-2 was also a single staged rocket. So why do not we have SSTO nowadays. What is stopping us to develop them. Let us have a look at one of the most nearly successful SSTO of all time the VentureStar.Before that let us first differentiate between the X-33 and the VentureStar. The X-33 was basically a test platform on which the materials and other components which would have been used on the VentureStar were tested. Let us have a look at the X-33 first as it was the one which was tested.

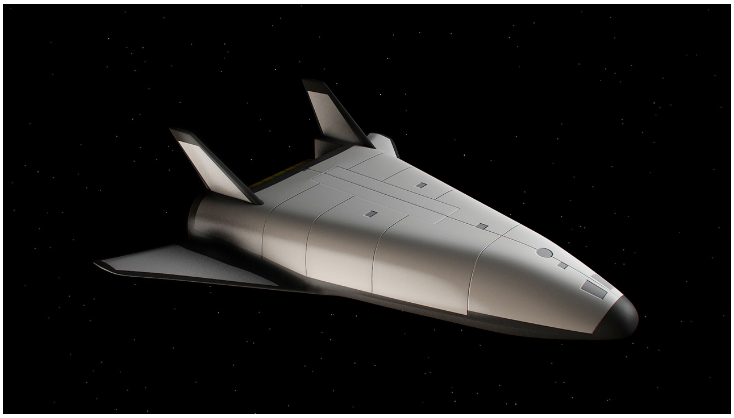


Figure 2.32: Conceptual Design.

2.5.1 | LOCKHEED X-33

- The X-33 is a lifting body shape, which provides advantageous L/D for cross range and maneuvering. During ascent X-33 is powered by two modified J-2S engines, called XRS-2200, delivering 410,000 lb thrust at liftoff. The engines are arranged as linear aerospikes, and differential throttling of each of the two banks of engines provides actuator-free thrust vectoring. The vehicle uses LH_2 and LOX propellant, with a fuel mass of 30,000 lb LH_2 and oxidizer mass of 180,000 lb LOX respectively. The vehicle is 69 ft. long and 77 ft wide. The X-33 has a gross mass of 285,000 lb, and can obtain a maximum speed of Mach 13.8.

- The integrated thrust structure, cryogenic tanks, and inter-tank structure make up the load-carrying structure. Secondary structures, such as control surfaces and the Thermal Protection System (TPS), are similarly linked to the core structure as the RLV. The majority of components are designed to perform many functions, which helps to reduce weight. The inter-tank and truss thrust structure is intended to serve as the main vehicle structure.

- X-33 utilizes two multi-lobe LH_2 tanks in the aft, and a single multi-lobe LOX tank in the forward section. Two vertical stabilizers provide yaw control, and two body flaps and two canted horizontal stabilizers with inboard and outboard elevons provide pitch, roll and supplemental yaw control. T-0 umbilical's provide ground interfaces for propellants, pressurants, command and electrical services.

- The X-33 is protected from high heating on its windward surface by 1241 durable, metallic Inconel TPS panels, which are mechanically attached to the vehicle using a standoff support structure. Flexible thermal blankets, similar to those used on the Space Shuttle, are used on leeward surfaces where heating rates are lower, and carbon-carbon is used on the nose cap and aerosurface leading edges.

- The TPS must keep the temperature of the TPS support shell (Gr-BMI) at or below 350°F. The carbon/carbon and metallic TPS is packaged into 18" square panels. The panels have been placed based on the X-33 using the maximum heating trajectory (Mach 15).

    1. Carbon/Carbon Temp > 2,000 °F
    2. PM2000 Temp < 2,000 °F
    3. Inconel 617 Temp < 1,700 °F
    4. Titanium Temp < 1,300 °F
    5. PBVAFRSI Blankets Temp < 900 °F

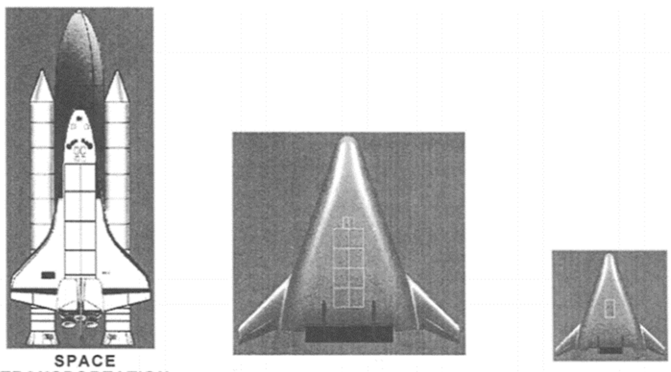
- The X-33 has two linear aerospike engines that use cryogenic LOX/LH_2 propellants. Boeing and Rocketdyne developed the engines, designated XRS-2200. The engines, used a series of 10 external combustor nozzles (thrust cells), arrayed on each side of two ramps to deliver thrust. The nozzles were truncated; turbine exhaust was delivered out of the base, producing additional thrust.

- The aerospike has the advantage of automatically compensating for altitude and delivering efficient thrust (i.e., the nozzles are not under or overexpanded like conventional bell nozzle engines). Another benefit is that differential throttling can be used to achieve pitch, roll, and yaw from the two engines. This eliminates the weight and complexity of gimbals, bellow feed lines, and actuators. The turbomachinery power packs can run in single-engine "engine-out" mode and feed both sets of ramps, maximising abort potential.

- The specific impulse of the engines is 340.3 seconds at sea level, and 429.3 seconds at vacuum.
- Two LOX tanks, were built, and were comprised of lightweight aluminium lithium material whereas because of the use of composites in the lower part of the aircraft the LH_2 tanks took more time to design. Two LH_2 tanks were constructed of composite graphite epoxy material. Each tank weighs 4,600 lb and is designed to carry 29,000 gallons of LH_2 at -423 °F.

2.5.2 | VentureStar

There is no difference in the basic vehicle structure between the X-33 and the VentureStar. VentureStar has the same composite cryogenic tanks, composite internal structure, LH_2/LOX Linear Aerospike Engine main propulsion, and metallic Thermal Protection System. Some other key points of the VentureStar are that it has a height of 38.7 m, diameter of 39 m and weighs 10,00,000 kg. It could carry a payload of 20,000 kg.
The VentureStar is powered by seven LH_2/LOX linear aerospike rocket engines. The combination of seven engines gives a thrust to weight ratio of 1.39 at lift-off, and a 105% emergency engine rating, allowing VentureStar to survive the worst-case engine loss at lift-off.



SYSTEM	SPACE TRANSPORTATION SYSTEM	VENTURESTAR™	X-33
LENGTH	56.1m	38.7m	20.4m
SPAN	23.8m	39.0m	20.7m
GLOW	2,041,000kg	991,500kg	123,800kg
FUEL	LH_2/LO_2 + SOLID	LH_2/LO_2	LH_2/LO_2
FUEL WEIGHT	1,725,000kg	874,500kg	95,700kg
EMPTY WEIGHT	269,400kg	89,300kg	28,500kg
MAIN PROPULSION	2 SOLIDS + 3 SSME BELLS	7 RS2200 LIN. AEROSPIKES	2 J-2S LIN. AEROSPIKES
LIFT OFF THRUST	28,468kN	13,390kN	1,823kN
MAXIMUM SPEED	ORBITAL	ORBITAL	MACH 15+
PAYLOAD			
(100 NM/28.5° ORBIT)	23,133kg +	22,680kg +	N/A
PAYLOAD BAY SIZE	4.57m x 18.3m	4.57m x 13.72m	1.52m x 3.05m

Figure 2.33: Comparision of STS, VentureStar & X-33.

The VentureStar is covered with a metallic Thermal Protection System (TPS). This TPS is composed of robust, damage resistant, 0.46m x 0.46m PM2000, Inconel and titanium panels housing encapsulated thermal insulation.

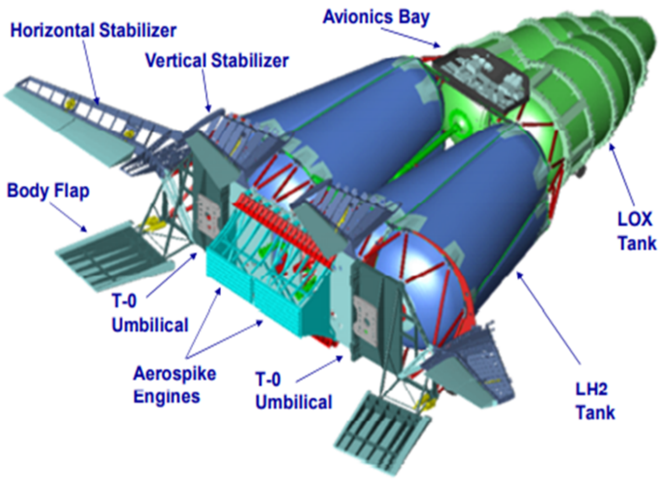


Figure 2.34: VentureStar Structural Schematics.

VentureStar nose cap and fin surface edges are oxidation resistant carbon-carbon. The fins are metallic hot structure while the lower body flaps are protected by carbon-carbon. The combination of robust design, cool re-entry temperatures afforded by the lifting body design and encapsulating the thermal insulation, eliminates costly repair, replacement, and waterproofing operations of the first generation ceramic TPS covering the Space Shuttle.

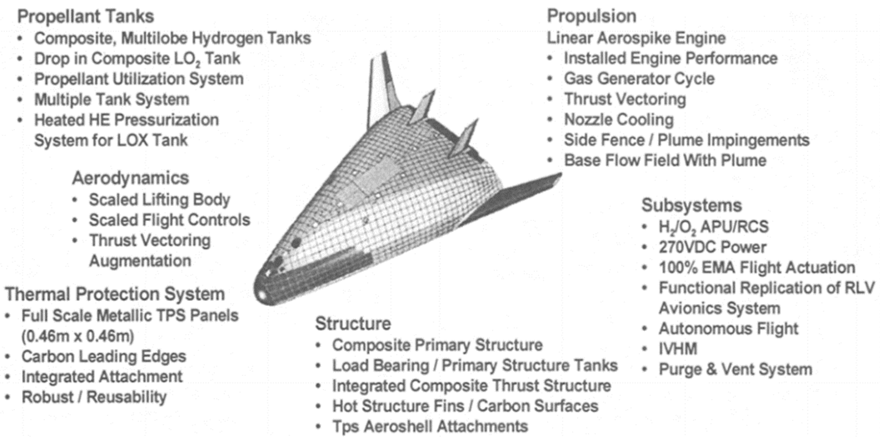


Figure 2.35: VentureStar.

2.5.3 | XRS-2200 ENGINE

The XRS-2200 Engine is a Linear Aerospike Engine that was incorporated/ developed from the Saturn-V's J-2 or J-2S engine hardware, to reduce the cost of development, schedule, and risk. The XRS-2200 engine was a fast-track program that integrated the existing J-2 and J-2S hardware assets and aerospike technology. This engine was developed and tested on the X-33 (predecessor of VentureStar).

2.5.3.1 | Specifications

The X-33 is designed to house two XRS-2200 engines.These engines use LH_2 as the fuel & LOX as the oxidizer. This propellant combination is to provide the main propulsion for X-33 during vertical launch and ascend. The XRS-2200 is a simple gas generator cycle (open system with turbine gases expended overboard). A high-efficiency nozzle is created by combining turbine exhaust gases and thrust cells.

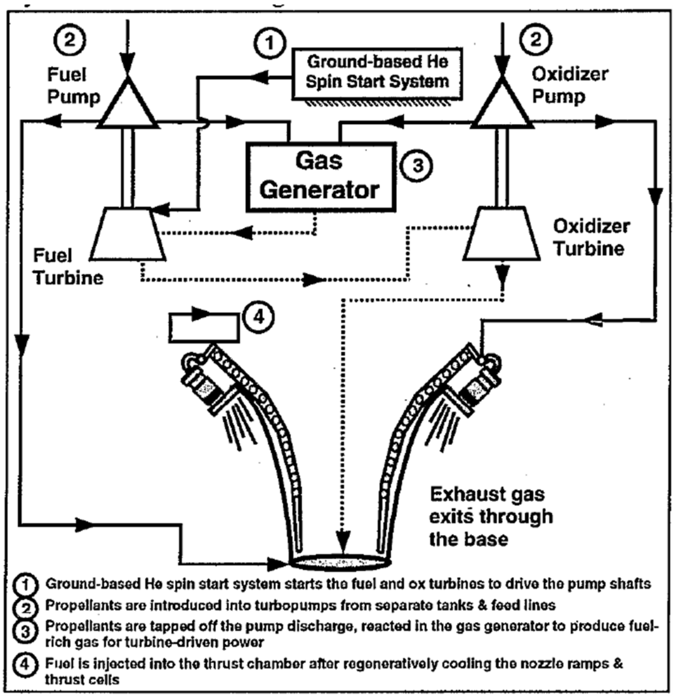


Figure 2.36: VentureStar Engine Schematics.

2.5.3.2 | Benefits Of An Aerospike

- Aerospike engines have altitude compensating nozzles. They can automatically adjust to the atmospheric conditions and enhance the nozzle efficiency.

- These engines can achieve Thrust Vector Control (TVC) without the usage of gimbals. It is done by diverting the flow from one side to other.

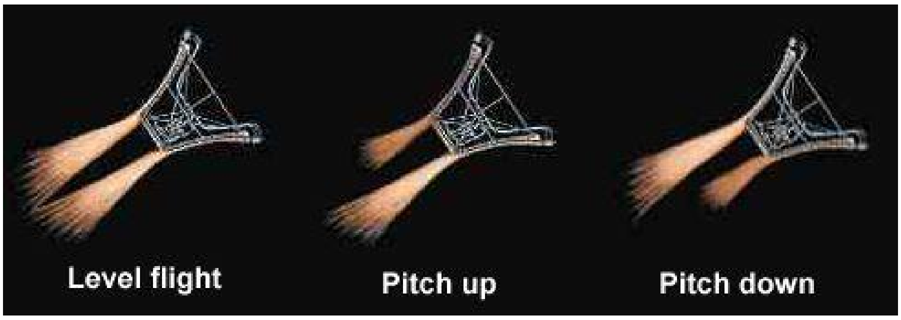


Figure 2.37: TVS (Thrust Vector Control) in Linear Aerospike Engines.

- Installed engine aids in weight reduction.
- Flexibility in engine packing.
- Shorter Engine length and altitude compensating nozzle.
- Modular inter-changeable hardware.

2.5.4 | Testing

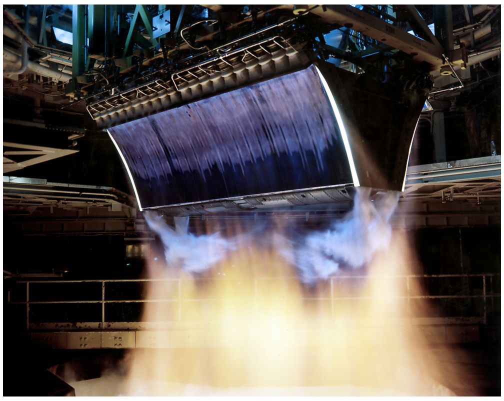
Early Testing:

- The first hot fire test of the linear aerospike engine took place in September 1971, with a combination of J-2S turbo-machinery and J-2 hardware.
- The key objective was to test thrust cell and nozzle performance. Twenty (20) thrust cells configured in a linear array (ten on each ramp/ side) produced 250,000 lb of thrust.
- A total of 44 hot-fire tests were completed with an accumulated ground test time of 3,114 seconds. The longest test was 592 seconds.
- Chamber pressures ranged from 680 psia - 1250 psia, with the mixture ratio between 3.2 - 5.6.

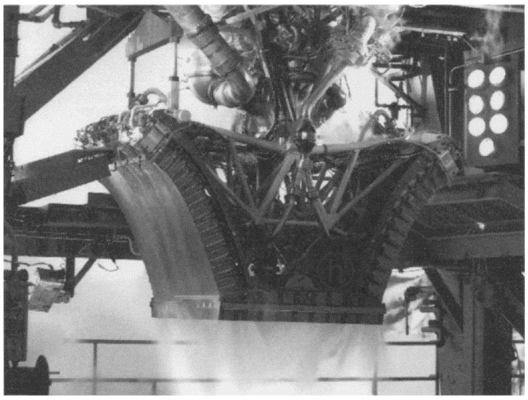
Second Testing:

- A second linear aerospike engine was tested in October 1972.

- The key objective was to test engine gimbaling to provide thrust vector control.
- Ten thrust cells (five on each ramp) produced 125,000 lb thrust.
- Thirty (30) hot fire tests were completed, 1000 seconds of operation.



(a)



(b)

Figure 2.38: XRS-2200 Engine undergoing second test.

SPECIFICATIONS OF **RS–2200** ENGINE		
Thrust (in lbf)	At Sea Level	431,000
	In Vacuum	495,000
Specific Impulse (in sec)	At Sea Level	347.0
	In Vacuum	455
Propellants	Oxygen & Hydrogen	
Mixture Ratio (O/F)	6.0	
Chamber Pressure (in psia)	2,250	
Cycle	Gas Generator	
Area Ratio	173	
Throttling, Percentage Thrust	50 - 100%	
Dimensions (in Inches)	Forward Aft	252 wide X 93 long
	Aft End	93 wide X 93 long
	Forward to Aft	170

Figure 2.39: Specifications of RS-2200 Engine.

2.5.5 | Reason For Failure

In November 1999, a joint NASA-Lockheed Martin investigation team reported that the damage or debonding, was caused by microcracks in the composite inner and outer skins. The cracks allowed pressurized hydrogen to seep into the core from inside the tank and caused the nitrogen gas maintained outside the tank as a safety measure to be "cryo pumped" in through the outer skin as the liquid hydrogen chilled it. That produced pressure that was higher than expected in the composite core, which in turn caused the separation. Due to the above incident and also due to financial woes the project was terminated.

3

Conclusions

By comparing the specifications of the propulsion systems of the various existing SSTO's, we have arrived at the conclusion that the SABRE engine of the SKYLON is the most efficient model and will help in the growth and development of SSTO's in the future.

Various engine parameters such as the Thrust to Weight ratio, Specific Impulse, Chamber Pressure and Thrust Generated were considered before arriving at this conclusion.

The detailed comparison of the propulsion systems is shown in the table below.

3.1 | Key Takeaways From Other SSTO's

After a detailed study on various SSTO's and their respective propulsion systems we have concluded that the SABRE engine of the SKYLON is the best propulsion system for an SSTOV at present. However, we cannot disregard the contributions made by the other SSTO's. Hence, here are a few major takeaways from each SSTO discussed in detail previously.

3.1.1 | DC-X (Delta Clipper Experimental)

The DC-X stands as a historical milestone in not only SSTO's but also RLV's. It was the first VTVL RLV which was completely tested. The major takeaway from this SSTO is the usage of LOX/LH_2 propellants which as we know, provides higher specific impulse when compared to all other chemical fuels.
However, the major drawback of the vehicle was that, the propulsion system was unable

to generate the required thrust to reach the orbit. Even after the modifications made to the RL-10 engine, the vehicle had only a range of over 3000m.

3.1.2 | ROTON ATV (Atmospheric Test Vehicle)

The Roton ATV was a SSTO vehicle which had a futuristic rotary wing design. However, this unique design was not able to meet up to the expectations. Also, the propulsion system which was used in the Roton ATV at that time was not completely developed and was later redesigned and used on the X-34 vehicle.

3.1.3 | ARCA

The major takeaway from this SSTOV was the usage of composites for majority of the engine components. These materials were cost efficient and also reduced the weight of the overall engine. Being an eco-friendly SSTO, further developments to this ongoing project can surely make it a strong contender and a successful SSTO in the future.

3.1.4 | Lockheed Martin X-33 And VentureStar

Though the X-33 and the VentureStar projects could not be mass produced, they showed the world that SSTO's could become a possibility in the near future. This project was also one of the major reasons why linear aerospike engines were so vastly researched and worked on.

3.2 | Reasons for Selecting the SABRE Engine

The major parameters that make the SABRE engine different from the other propulsion systems are :

- Unique Design :
  SABRE stands for Synergistic Air Breathing Rocket Engine and as the name suggests it operates in 2 modes, i.e., Air breathing mode and Traditional Rocket mode which helps in generating the sufficient thrust required for an SSTO.

- Horizontal Take Off and Horizontal Landing (HOTOL) :
  This configuration reduces the gross lift off thrust required for take-off significantly thereby reducing the total thrust required by the engine.

- High Specific Impulse (I_{sp}) :
  Due to its air breathing mode, the SABRE engine is able to produce a high specific impulse.

- Positive Test Results :
  Apart from its unique design and engine characteristics the SABRE engine has managed to produce many positive results when tested which increases its reliability when compared to other propulsion systems.

- Constant Funding and Support :
  Funding for SSTO's has always been an issue, however due to its positive results and highly promising design the SABRE engine or the SKYLON SSTO has managed to get constant funding and support from the European Space Agency (ESA).

3.3 | Final Remarks

The SABRE engine could become the future of the aviation and space industry, which may ease many missions from earth's surface to space. The potential of providing high thrust with speed from 0 to Mach 5.5 with outstanding thrust over the entire flight from ground to very high altitude efficiently, makes SABRE engine as a dominant solution for launching vehicles over the up to date launching vehicles.
Further modification in this engine may lead not only to orbit but also open up the possibility of interstellar travel. This is a revolution for the upcoming era.

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