NovaTech Space Systems™

Presenting: NOVA I

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Background

A group of venture capitalists are interested in funding a network of small satellites to provide satellite internet coverage across the globe. They have asked us to design a low-cost rocket with a reusable first stage that can place a single **1,000 kg satellite** into a **500 km altitude low-Earth orbit**.

Key design requirements:

- Deliver a 1000kg payload to 500km LEO
- 9km/s of ΔV required
- Rocket must be low cost
- First stage must be reusable

Design Thought Process

1. Determine number of stages
2. Type of rocket (SRM vs LRM)
3. Determine rocket fuel
   a. Selection based on I_{sp}/$ ratio
4. CEA to finalize ϕ, ε, P_c selection
   a. Range of values:
      i. $1 \leq \phi \leq 4$
      ii. $20 \leq \varepsilon \leq 50$
      iii. $150\ \text{atm} \leq P_c \leq 300\ \text{atm}$
5. Find optimum stage mass
   a. Used matlab script based on Orbital Mechanics ideal staging weight ratio examples
6. Selection of LRM engine configuration
   a. Determined cooling configuration
7. Reiterate and optimize to maximize I_{sp}
   a. Higher $I_{sp} \rightarrow$ lower cost

Design Overview - Staging Breakdown

- Two stage design selected
  - Based on common practice in the Space industry
- MATLAB script (derived from Lagrange multiplier method) generated following mass breakdown:
  - **Launch Total Mass = 15,266 kg**
  - Payload = 1,000 kg
  - First Stage:
    - m_o = 11,236 kg
    - m_e = 1,685 kg
    - m_p= 9,550 kg
    - **□ = 0.265**
    - **ε = 0.136**
  - Second Stage:
    - m_o = 3,030 kg
    - m_e = 455kg
    - m_p= 2,576 kg
    - **□ = 0.199**
    - **ε = 0.15**

Ideal ΔV Breakdown:

- Launch from French Guiana
  - ΔV_{surf} = 0.46 km/s
- Stage 1:
  - **ΔV = 4.1345 km/s**
- Stage 2:
  - **ΔV = 4.399 km/s**

Fuel Choice

- **Staged Combustion LMR** for both stages
  - High launch performance
  - Throttle control for 2nd stage
- Using $\mathbf{O_2(L)}$ and $\mathbf{H_2(L)}$ for both stages
  - Cheap, high I_{sp}, easily accessible
  - Challenging to manage cryogenic propellant storage
- Optimum mixing ratio of **2.1/1**

Fuel Performance Parameters:

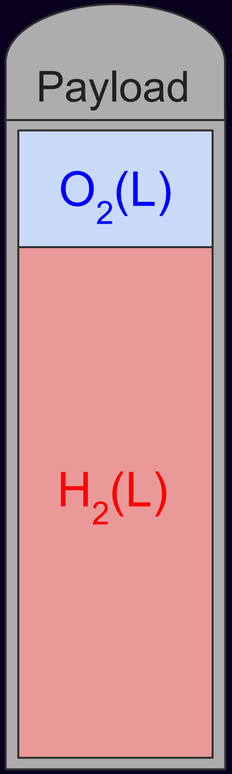
- $\mathbf{P_c = 30.398\ MPa}$
- $\mathbf{T_c = 2878.64\ °K}$
- $\gamma_c = 1.2206$
- $A_c = 0.202\ m^2$
- $L^* = 1\ m$
- $\mathfrak{M} = 9.607$ amu
- $C^* = 2416.9$ m/s

Cryogenic Fuel Storage: Design Attributes

- Storage temps:
  - H_2 = 90 K
  - O_2 = 20 k
- Double walled storage tanks
  - Inner wall SA240 Grade 304 stainless steel
  - Outer wall SA516 Grade 70 carbon steel
  - Evacuated glass bubbles held under vacuum
    - Reduce heat radiation between walls
    - NASA standard practice
- Additional design considerations:
  - Exterior of rocket painted white to reduce thermal radiation
  - Radiation shield composed of Polyethylene for 2nd stage

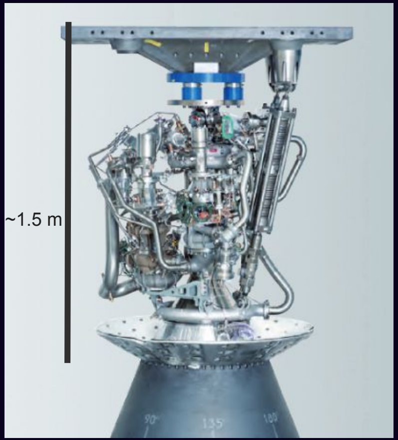
Storage Orientation

- lower center of gravity



Turbomachinery Overview

- Stage 1
  - Estimated mass of turbopump: 65 kg
  - P_i = 8.36 MPa
  - Estimated mass of all turbomachinery: ~160 kg
- Stage 2
  - Estimated mass of turbopump: 27 kg
  - P_i = 8.36 MPa
  - Estimated mass of all turbomachinery: ~70 kg



https://www.ariane.group/wp-content/uploads/2020/06/VINCI_2020_04_DS_EN_Eng_Web.pdf

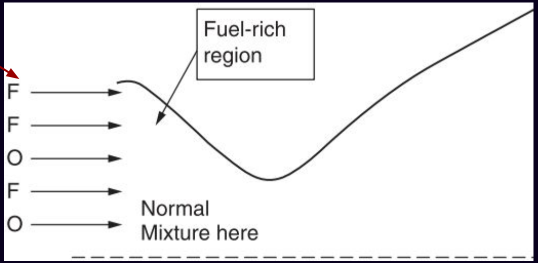
Turbomachinery of Vinci rocket with performance similar to stage 1

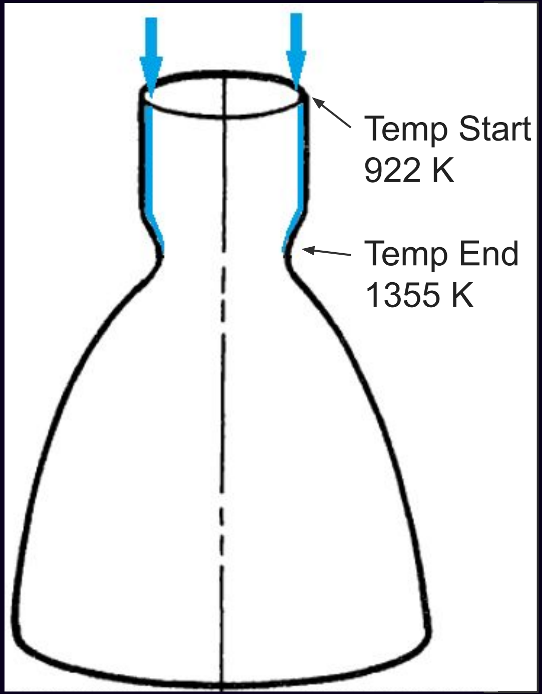
Combustion Chamber - Film Cooling

- Gaseous film cooling was necessary due to high combustion temperatures (2878 K)
  - Higher than melting point of stainless steel (1783 K)
- Pros
  - Little cost to overall weight, reduce temperature at walls in the hottest areas
- Cons
  - Manufacturing/Design/Testing cost increase
- Implementation
  - Embedded Hydrogen injectors providing a total fuel flow of 0.118 kg/s

$$\frac{T_{aw} - T_{wg}}{T_{aw} - T_{co}} = e^{-\left(\frac{h_g}{G_c C_{pvc} \eta_c}\right)} \qquad (4\text{-}34)$$

*Design of Liquid Propellant Rocket Engines, Huzel and Huang





Nozzle - Design Components

- Both stages use bell nozzle configuration
- 1st stage performance parameters
  - **P_e = 0.089487 MPa**
  - T_e = 855.71 K
  - **ε = 25**
  - A_t = 0.00373 m^2
  - A_e = 0.0933 m^2
  - V_e = 2305.6 m/s
  - C = 4163.6 m/s
  - C_F = 1.7277 m/s
  - **F = 187199 N**
  - $\dot{m}$ = 44.48 kg/s
  - **I_{sp} = 429 s**
- Expansion ratio of 25 compromise
  - Minimize I_{sp} loss
  - Minimize overexpanded nozzle loss
    - $P_A \cong$ 0.1MPa
- 2nd stage performance parameters
  - **P_e = 0.03394 MPa**
  - T_e = 673.62 K
  - **ε = 50**
  - A_t = 0.000929 m^2
  - A_e = 0.04645 m^2
  - V_e = 2374.5 m/s
  - C = 4314.5 m/s
  - C_F = 1.785 m/s
  - **F = 49418 N**
  - $\dot{m}$ = 11.724 kg/s
  - **I_{sp} = 440 s**
- Maximize expansion ratio
  - Stage 1 burnout at 150 km
    - $P_A \cong$ 0 Pa
  - No need to pressure match

Nozzle - Regenerative Cooling

- **Regenerative Cooling: Circulating coolant absorbs heat from the nozzle**
- Material selection: Stainless Steel
  - Unreactive
  - Melting point: 1783 K
- Benefits:
  - Efficient heat dissipation
  - Ensure structural integrity
- Final Pressure and Temperature
  - Pressure: 296.2 Bar
  - Temperature: 2301 K

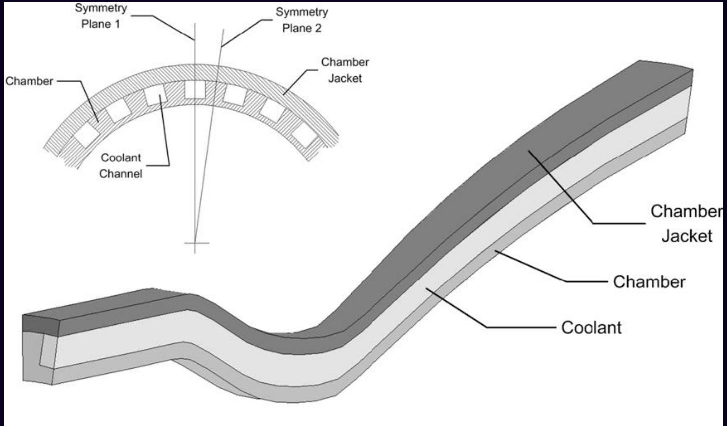


Fig. 1. Schematic view of solution domain.

Rocket Overview

- Rocket approx dimensions:
  - Width: 4 m
  - Height: 13.5 m
  - Cross Sectional Area: 12.566 m
- Stage burnout breakdown:
  - 1st stage burn time: 214.7 s
    - Alt = 155 km
  - 2nd stage burn time: 224.9 s
    - Alt = 545 km
- **Total ΔV : 9.175 km/s**
  - **ΔV_{excess} = 0.632 km/s**
- Total cost to refuel (10 launches): $300,000
- **Total cost: $13.5 million**

https://www.nasa.gov/image-article/spacex-falcon-9-rocket-vertical-position-launch-pad/

Future Work

Continued research provided adequate funding:

- Controlled reentry/landing of 1st stage using excess ΔV
- Optimize first stage
  - Pressure match to sea level conditions
  - Stability control (fins / gimbling nozzle / compressed air thrusters)
- Structural analysis of chamber and storage tanks
  - Determine wall thickness to contain high pressures
- Employ use of passive cooling where possible in nozzle

THANK YOU!!!

Reference

- https://www.semanticscholar.org/paper/Numerical-analysis-of-regenerative-cooling-in-Ula%C5%9F-Boysan/62da3334c460e708f17151fecdb60153783ff230/figure/0
- https://ntrs.nasa.gov/api/citations/20210018293/downloads/2021%20CEC%20Virtual%20Big%20Tank%20LH2%20DAA%20draft%2007JUl2021.docx.pdf
- https://ntrs.nasa.gov/api/citations/19710019929/downloads/19710019929.pdf

Variable	1st Stage	2nd Stage	Variable	1st Stage	2nd Stage
P_i	8.36 MPa	8.36 MPa	$\mathfrak{M}$	9.607 [amu]	9.607 [amu]
P_c	303.98 [bar]	303.98 [bar]	V_e	2305.6 [m/s]	2374.5 [m/s]
P_e	0.89487 [bar]	0.33994 [bar]	C	4163.6 [m/s[	4314.1 [m/s]
T_c	2878.64 [K]	2878.64 [K]	C*	2416.9	2416.9 [m/s]
T_c	855.71 [K]	673.62 [K]	C_F	1.7227	1.785 [m/s]
$gamma_c$	1.2206	1.2206	F	187199.315 [N]	49417.875 [N]
A_c	0.0212 [m^2]	0.00743 [m^2]	m_{dot}	44.48 [kg/s]	11.742 [kg/s]
A_t	0.00373 [m^2]	0.000929 [m^2]	Expansion ratio	25	50
A_e	0.0933 [m^2]	0.04645 [m^2]	Isp	429	440
L*	1 [m]	1 [m]	beta	0.265	0.199
L_c	0.1754 [m]	0.125 [m]	Structural factor	0.136	0.15

Presentation

from 8:30-10:20am. The presentation should summarize:

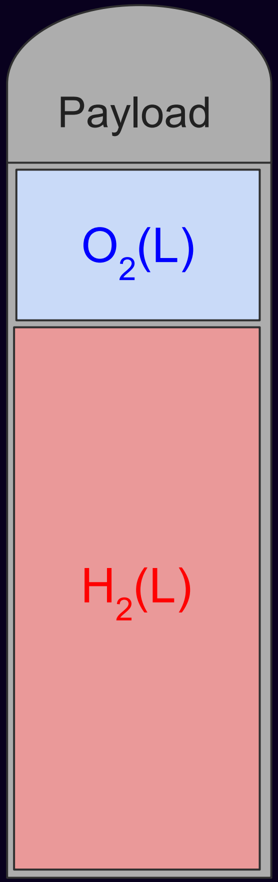
(1) your design approach,

(2) the main features of your design,

(3) the distinguishing aspects of the design

(4) the ability of the design to meet the stated objectives and cost per launch, and

(5) the next steps you would follow if it was chosen for funding



m0	1.5266e+04
m1	1.1236e+04
m1_e	1.6854e+03
m1_p	9.5503e+03
m2	3.0303e+03
m2_e	454.5382
m2_p	2.5757e+03
m_pl	1000