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Mission B: Lunar Payload Delivery Vehicle

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Mission Overview

Mission Objective:

- Deliver a 50 kg science payload from 400 km Low Earth Orbit (LEO) to a 100 km circular Low Lunar Orbit (LLO), with a restartable final stage.

Mission Maneuvers

1. Trans-Lunar Injection (TLI)
2. Trans-Lunar Coast
3. Lunar Orbit Insertion (LOI)

Mission Starting Conditions	
Parameter	Value
LEO Altitude	400 km
Final Orbit	100 km LLO
Payload Mass	50 kg



Mission ΔV Budget

Mission Delta-V Budget

- The required mission ΔV includes the primary burns plus correction and reserve margins.
- Mid-course correction maneuvers account for trajectory dispersions and navigation uncertainties during translunar coast.

Itemized Mission Delta-V Budget		
Maneuver / Category	ΔV (km/s)	Source
Trans-Lunar Injection (TLI)	3.107	[1], [5]
Lunar Orbit Insertion (LOI)	0.819	[1], [5]
Mid-course correction maneuvers (TCMs)	0.070	[2]
Injection/navigation/operation Reserve	0.080	[1], [2]
Total	4.076	

Rocket Stage Analysis / Study



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Input Parameters

- Two Stage
 - $\epsilon_1 = 0.1$
 - $\epsilon_1 = 0.1$
 - $I_{sp1} = 331.52 \text{ s}$
 - $I_{sp2} = 304.15 \text{ s}$
 - $\Delta V_1 = 3.107 \text{ km/s}$
 - $\Delta V_2 = 0.969 \text{ km/s}$
- Single Stage
 - $\epsilon = 0.11$
 - $I_{sp} = 304.15 \text{ s}$
 - $\Delta V = 4.076 \text{ km/s}$

Two-Stage Rocket – Mass Analysis							
Stage	$\Delta V \left(\frac{km}{s}\right)$	$I_{sp} (s)$	ϵ	$m_s (kg)$	$m_p (kg)$	$m_f (kg)$	$m_0 (kg)$
1	3.107	331.52	0.1	15.71	141.40	88.35	229.75
2	0.969	304.15	0.11	2.49	20.15	52.49	72.64

Single-Stage Rocket – Mass Analysis							
Stage	$\Delta V \left(\frac{km}{s}\right)$	$I_{sp} (s)$	ϵ	$m_s (kg)$	$m_p (kg)$	$m_f (kg)$	$m_0 (kg)$
1	4.076	304.15	0.11	28.26	228.67	78.26	306.93

Trade study Result:

- Single Stage: Initial Mass in LEO $m_0 = 306.93 \text{ kg}$
- Two Stage: Initial Mass in LEO $m_0 = 229.75 \text{ kg}$
 - Mass savings from staging = 77.18 kg

$$\Delta V = I_{sp} g_0 \ln \frac{m_0}{m_f} \quad MR = \frac{m_0}{m_f} \quad \epsilon = \frac{m_s}{m_s + m_p}$$

The **two-stage architecture was selected** because it minimizes initial mass by discarding the empty TLI stage before lunar arrival.

* ϵ values from [3], [4]*

Trajectory Analysis – Burn 1



- **Trajectory computed using patched-conic orbital mechanics approach.** [5]
- Steps:
 - Begin in 400 km circular LEO
 - Apply full stage 1 ΔV_1
 - Determine the resulting Earth-centered transfer orbit [2]
 - Compute spacecraft velocity at lunar distance
 - Determine arrival hyperbolic excess speed v_∞

$$1. \quad v_{leo} = \sqrt{\frac{\mu_E}{R_{leo}}}$$

$$2. \quad v^2 = \mu_E \left(\frac{2}{r} - \frac{1}{a} \right)$$

$$3. \quad v_\infty = |v_{transfer} - v_{moon}|$$

Burn 1: LEO Parking to TLI		
Parameter	Value	Equation
Initial LEO Parking Orbit	400 km	-
Circular LEO Velocity (v_{leo})	7.669 km/s	(1)
Velocity after stage 1 burn (v_{bo})	10.776 km/s	$v_{leo} + \Delta V_1$
Arrival Conditions (End of Coast)		
3 - Transfer speed at lunar distance (coast) ($v_{transfer}$)	0.756 km/s	(2)
Moon Orbital Velocity	1.018 km/s	
4 - Arrival Hyperbolic Excess Velocity (v_∞)	0.262 km/s	(3)

Trajectory Analysis – Burn 2



Hyperbolic Lunar Approach [5] Stage 2

Assuming no course corrections/maneuvers

1. Spacecraft enters the moons sphere of influence with velocity v_∞
2. Compute perilune velocity v_p
3. Determine LOI ΔV required
4. Compare required ΔV with stage-2 Capability

$$1. \quad v_p = \sqrt{v_\infty^2 + \frac{2\mu_M}{r_p}}$$

$$2. \quad v_{circ} = \sqrt{\frac{\mu_M}{r_p}}$$

$$3. \quad \Delta V_{LOI} = v_p - v_{circ}$$

Trajectory Analysis (Lunar Orbit Insertion)		
Parameter	Value	Equation
Perilune Velocity (v_p)	2.325 km/s	(1)
Circular Speed (100 km orbit) (v_{circ})	1.634	(2)
Required LOI burn ΔV_2	0.691 km/s	(3)
Remaining ΔV_2 after LOI Burn	0.278 km/s	-

LOI Burn Usage	
Parameter	Value
Propellant Used for LOI	15.03 kg
Propellant remaining	5.12 kg
LOI Burn Duration	186 s
Remaining maneuver capability:	
Remaining ΔV	0.278 km/s
Remaining Burn time	63 s

Propellant Candidates



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1st Stage

1. **LOX/LH₂**

- Highest Isp
- Exceptionally low density
- Large tanks and cryogenic complexity

2. **LOX/CH₄**

- Better density than hydrogen
- Cleaner than RP-1
- Lower heritage and still larger tanks than RP-1

3. **LOX/RP-1**

- Strong heritage
- High density, compact tanks
- Lower Isp than hydrolox and methalox

2nd Stage

1. **NTO/MMH**

- Hypergolic and restartable
- Storable for long coast phases
- Toxic handling concerns

2. **LOX/CH₄**

- Good performance
- Cryogenic storage is harder for long coast

3. **LOX/LH₂**

- Highest Isp
- Worst density and most difficult storage

Initial CEA Sweep



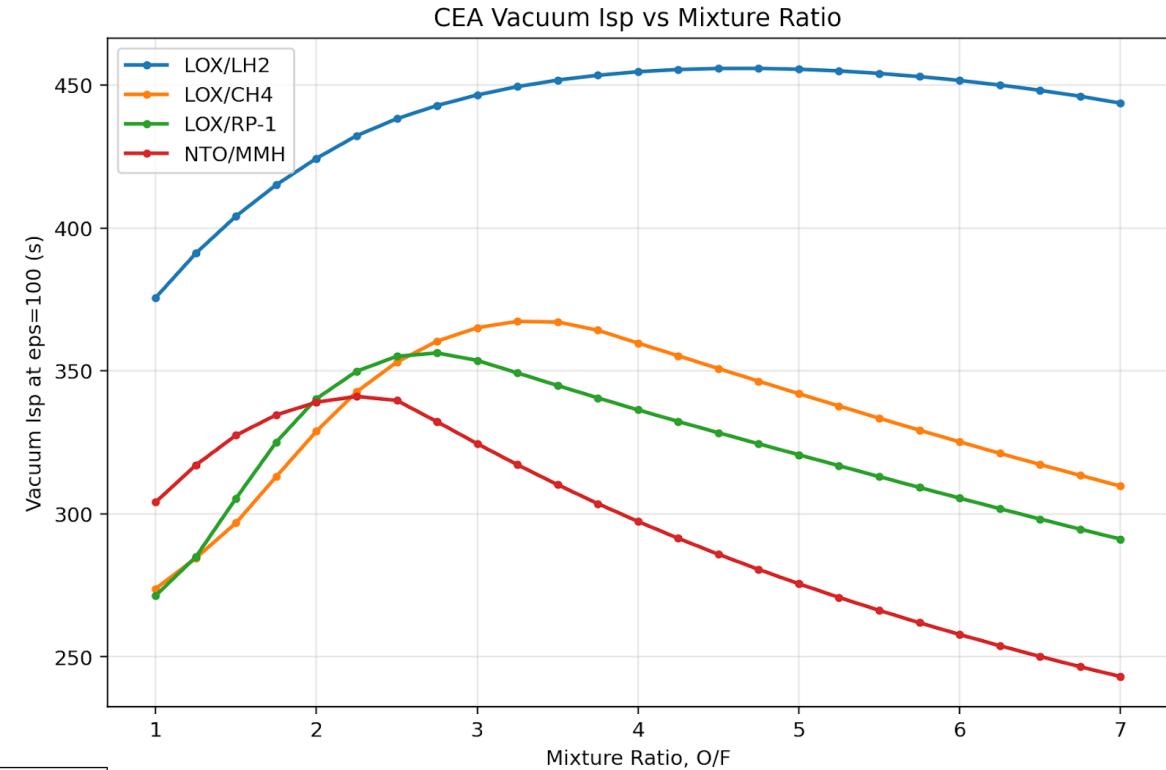
- O/F Ratio Sweep Range :1-7

LOX/LH2: peak at O/F = 4.75, Isp = 455.8 s, **Ivac = 468 s**

LOX/CH4: peak at O/F = 3.25, Isp = 367.3 s, **Ivac = 381 s**

LOX/RP-1: peak at O/F = 2.75, Isp = 356.3 s, **Ivac = 371 s**

NTO/MMH: peak at O/F = 2.25, Isp = 341.1 s , **Ivac = 353 s**



Stage	Candidate	Heritage	Vacuum performance	Storability	Restart ease	Toxicity	Relative tank size	Overall fit
1/2	LOX/LH2	High	Excellent	Poor	Moderate	Low	Large	Good
1/2	LOX/CH4	Moderate	Very good	Good	Moderate	Low	Medium	Good
1	LOX/RP-1	Very high	Good	Fair	Moderate	Moderate	Small	Very good
2	NTO/MMH	Very high	Good	Excellent	Excellent	High	Small	Excellent

Selected Propellants



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Stage 1 selected: LOX/RP-1

- Chosen for high density to keep the TLI stage smaller and more compact
- Methalox offered slightly better Isp, but not enough to offset the larger tank volume (25% larger tank)
- Strong heritage and low technical risk

Stage 2 selected: NTO/MMH

- Chosen for reliable restart
- Hypergolic ignition removes need for a complex ignition system
- Storable over the lunar transfer coast



Final CEA Values



KeroLox

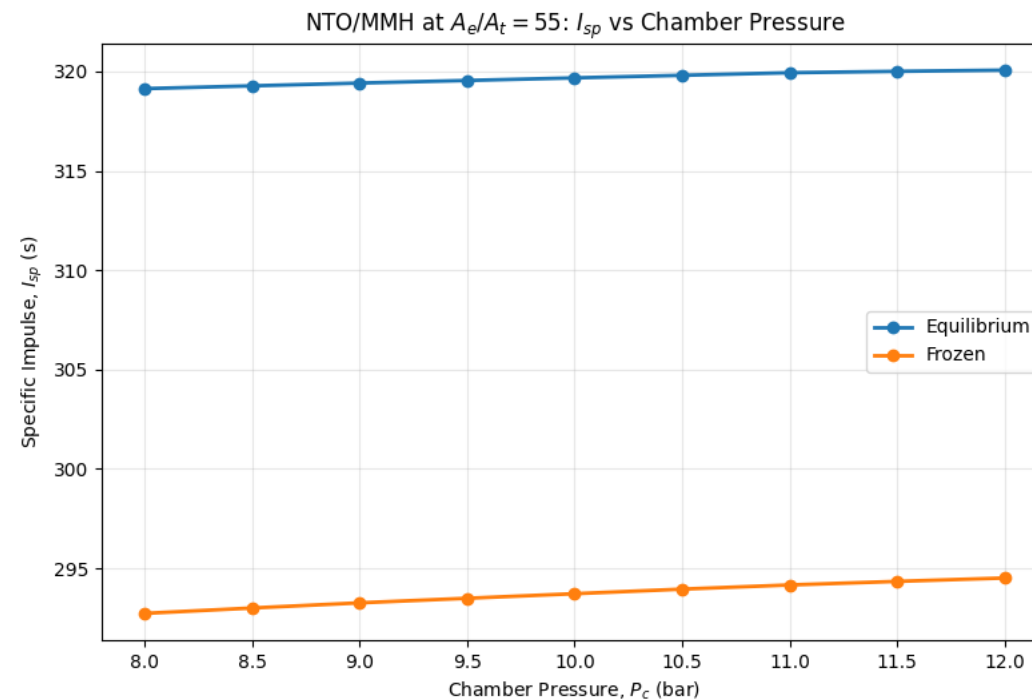
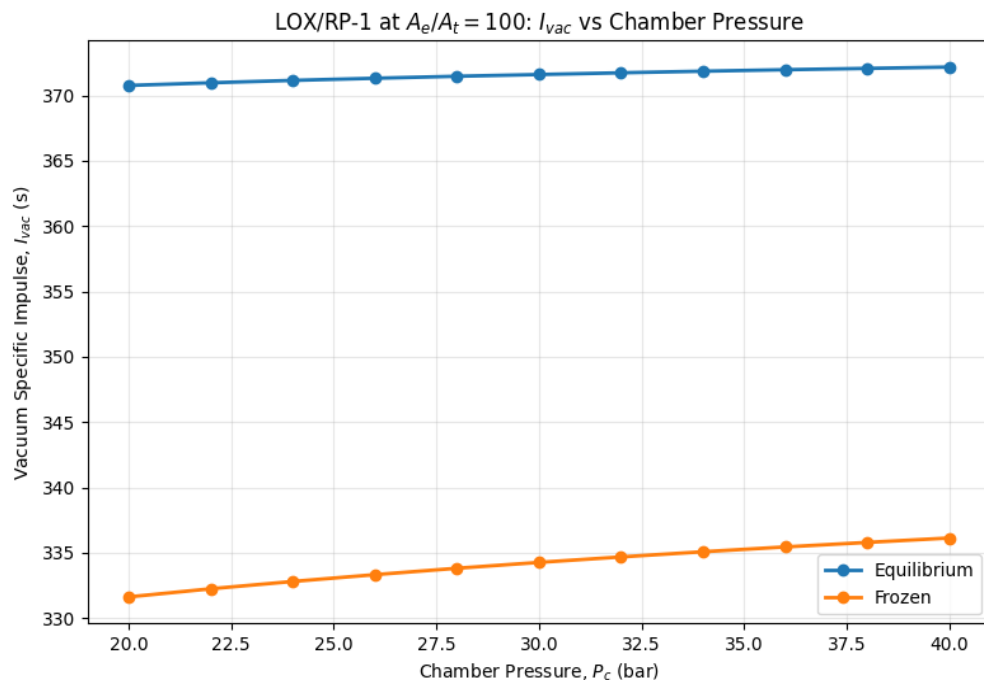
Frozen

- $T_0 = 3506.20 \text{ K}$
- $M = 23.576$
- $\gamma = 1.2145$
- $I_{sp} = 3142.8 \text{ m/s} = 320.48 \text{ s}$
- $I_{vac} = 3252.2 \text{ m/s} = 331.63 \text{ s}$

MMH/NTO

Frozen

- $T_0 = 3167.90 \text{ K}$
- $M = 24.047$
- $\gamma = 1.2207$
- $I_{sp} = 2888.2 \text{ m/s} = 294.51 \text{ s}$
- $I_{vac} = 2982.8 \text{ m/s} = 304.15 \text{ s}$



- ❖ Stage 1: 20 bar was selected because increasing pressure gave only a marginal I_{sp} gain
- ❖ Stage 2: 12 bar was selected because lower pressures led to overly large ideal exit-to-throat area ratios

Propulsion System Design - Expansion Ratio



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- Using code, we determined Expansion Ratio where P_a is always 0
- I_{sp} , γ , T_0 and molecular weight already acquired – worked backwards with:

$$I_{sp} = \frac{c_F c^*}{g_0}$$

- Wrote a solver that iteratively adjusted A_e/A_t until the calculated specific impulse matched targeted CEA values

Stage 1 Expansion Ratio: 65.02

Stage 2 Expansion Ratio: 38.71

Stage 1 Chamber Pressure: 20 Bar

Stage 2 Expansion Ratio: 12 Bar

Stage 1 I_{sp} : 331.52 s

Stage 2 I_{sp} : 304.15 s

- Credibility: The AJ10-190 engine was used on the Space Shuttle Orbital Maneuvering System (OMS) 55:1 expansion ratio

Propulsion System Design – Thrust



- Clay provided me with thrust – Olly provided me with mass of propellant
Stage 1 Thrust: 1622 N Stage 2 Thrust: 240.8 N
Small – Starting in LEO Orbit – Last stage is restartable

$$\dot{m} = \frac{T}{g_0 I_{sp}} \quad t_b = \frac{m_b}{\dot{m}}$$

- Stage 1 mass flow rate: 0.498802 kg/s Stage 2 mass flow rate: 0.080695 kg/s
Stage 1 burn time: 283.47912 s Stage 2 burn time: 249.706304 s
- We can now find area at the throat

$$A_t = \frac{T}{C_F p_0}$$

- Stage 1 Area at the throat: 426 mm² Stage 2 Area at the throat: 108 mm²

Propulsion System Design – Nozzle



Nozzle Contour Selection

Both stages are using bell nozzles
Percentage Length is 80%

- Shorter, has lower mass
- Widely used in most rocket designs
- Maintains performance at high expansion ratios

$$L_c = \frac{L^* A_t}{A_c}$$

Parameter	Units	Stage 1	Stage 2
Thrust	N	1.62E+03	241
Chamber Pressure	bar	20	12
Burn Time	s	283	250
Mass Flow Rate	kg/s	0.499	0.0807
Characteristic Velocity c^*	m/s	1.71E+03	1.60E+03
Thrust Coefficient CF	-	1.9	1.86
Specific Impulse I_{sp}	s	332	304
Exit Area Ratio A_e/A_t	-	65	38.7
Exit Mach Number Me	-	4.68	4.34
Exit Pressure Pe	bar	0.0212	0.0241
Exit Pressure Ratio Pe/P_c	-	1.06E-03	2.01E-03
Throat Area A_t	mm ²	426	108
Exit Area A_e	mm ²	2.77E+04	4.18E+03
Characteristic Length L^*	m	1	0.6
Chamber Length L_c	m	0.25	0.15
Chamber Area A_c	mm ²	1.70E+03	431
Chamber Volume V_c	m ³	4.26E-04	6.47E-05
Chamber Diameter D_c	m	0.0466	0.0234



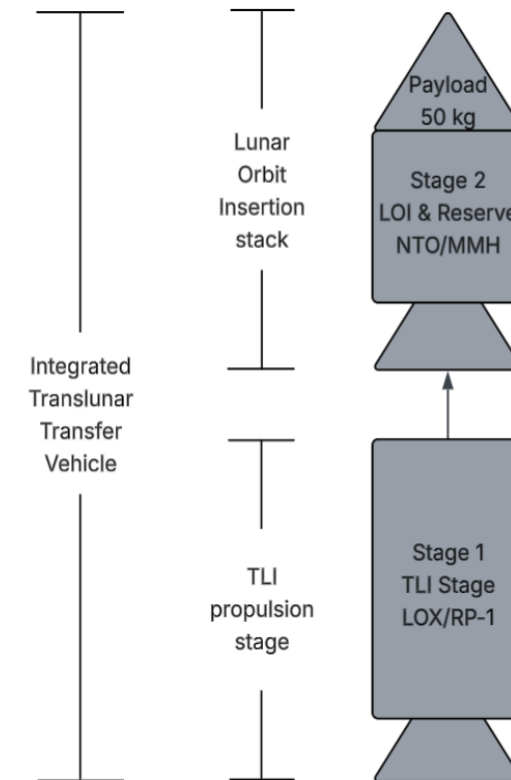
Vehicle Integration & Mass Closure

Component	Mass (kg)
Payload	50.00
Stage 2 propellant	20.15
Stage 2 structure	2.49
Stage 1 propellant	141.39
Stage 1 structure	15.71

$$m_0 = 50 + 20.15 + 2.49 + 141.39 + 15.71 = 229.75 \text{ kg}$$

Stage Interfaces

- Stage 1 performs Trans-Lunar Injection (TLI) and is jettisoned after burnout
- Stage 2 remains attached to the payload for Lunar Orbit Insertion (LOI)





Vehicle Performance & Apollo Comparison

Proposed Vehicle

Mission phase	Stage 1 (TLI)	Stage 2 (LOI)
Initial mass (kg)	229.75	72.64
Propellant mass (kg)	141.39	20.15
Structure mass (kg)	15.71	2.49
Thrust (N)	1622.15	240.78
T/W	0.72	0.338
Burn time (s)	283.50	249.70

Apollo

Mission phase	TLI	LOI
Stack mass (kg)	146,250	27,450
Thrust (N)	1,033,000	91,000
T/W	0.72	0.338
Burn time (s)	345.00	357.5

Performance Comparison

- Apollo TLI T/W = 0.72
- Apollo LOI T/W = 0.338
- Same mission phase T/W ratios

Result

- Comparable mission acceleration

Apollo mission-phase thrust-to-weight ratios were used as benchmarks to scale thrust for the translunar injection (TLI) and lunar orbit insertion (LOI) stages, since the Apollo performed the same mission phases.



Environmental & Societal Impacts

LOX/RP-1 exhaust composition (Stage 1)

Species	Mass %
CO ₂	49.891
H ₂ O	27.773
CO	21.690
H ₂	0.641
OH	0.004

NTO/MMH exhaust composition (Stage 2)

Species	Mass %
N ₂	39.78
H ₂ O	31.82
CO ₂	25.32
CO	2.58
H ₂	0.48

Scientific Benefits

- Enables low-cost lunar science missions
- Supports mapping and technology demonstrations

Commercial Opportunities

- Expands access to lunar missions for universities, startups, and private industry

Safety Considerations

- Hypergolic propellants require strict safety procedures



Sources

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- [5] Sharaf, M. A., Saad, A. N., and Nouh, M. I. *Transfers to Low Lunar Orbits*. JPL DESCANSO Monograph Series.
- [6] Binder, M. P., *RL10A-3-3A Rocket Engine Modeling Project*, NASA, 1997.