RPL

Apollo Lunar Module Ascent Engine

Historical reference comparison at the engine's intended vacuum operating condition

STATUS: REFERENCE-CASE AGREEMENT - NOT WHOLE-SOLVER VALIDATION

What this package is

This package compares one RPL Engine Workbench first-pass sizing result with authoritative public data for the Apollo Lunar Module Ascent Engine (LMAE). The LMAE is a useful reference because it is a documented pressure-fed, fixed-thrust, hypergolic vacuum engine. The historical engine used nitrogen tetroxide and Aerozine 50 and an ablatively cooled thrust chamber.

The primary case uses 0 Pa ambient pressure. It is not a sea-level LMAE case. The authentic application report is trial-watermarked because it was exported through the unchanged public trial build; the watermark was not removed or bypassed.

Package contents

File	Purpose
START_HERE_RPL_LMAE_VALIDATION.pdf	Historical basis, classification, comparison, and limitations
RPL_LMAE_ENGINE_REPORT.pdf	Authentic 11-page application-generated engine report
RPL_LMAE_NOZZLE_CONTOUR.dxf	Application-exported millimeter nozzle geometry for the same solved case
RPL_LMAE_REFERENCE_PROJECT.rplproj	Portable input-only project; reopen and solve to regenerate results

Bottom line

RPL predicts vacuum specific impulse of 307.97 s versus the NASA Table I value of 309.7 s: -1.73 s (-0.56%). This is useful reference-case agreement, but it is not formal validation: key engine conditions were entered as inputs and the calculation includes explicit efficiency assumptions.

The larger c* difference is reported rather than hidden, but the Boeing value is a production-injector acceptance-test reduction, not measured flight-engine c*. It is not an apples-to-apples validation metric.

Claim boundaries

- The comparison covers this versioned LMAE reference case only.
- Entered inputs are not counted as independent validation successes.
- No flight qualification, manufacturing authorization, or reacting-combustion CFD claim is made.
- RPL is not affiliated with or endorsed by NASA, Boeing, Rocketdyne, or the Apollo program.

Historical basis and actual operating condition

NASA TN D-7400 provides the most compact primary table of LMAE design characteristics and nominal performance. Its prose describes a nominal 3,500 lbf vacuum engine; Table I separately lists 3,450 lbf as typical of C-series Rocketdyne engines over a nominal 460-second duty cycle. These numbers are related but not interchangeable. NASA TN D-7082 identifies 3,500 lbf as the system requirement.

Historical values used

Quantity	Primary-source value	Use in this case
Operating environment	Vacuum	Entered: ambient pressure = 0 Pa
Propellants	NTO / 50% hydrazine + 50% UDMH (Aerozine 50)	Entered input
Mixture ratio	O/F = 1.6:1	Entered input
Chamber pressure	84.8 N/cm2 = 848,000 Pa = 0.848 MPa = 123 psia	Entered input
Expansion ratio	45.6:1	Entered input
Vacuum thrust	3,500 lbf design/nameplate = 15,568.8 N	Entered target: 15,568 N
Typical duty-cycle thrust	3,450 lbf = 15,346.4 N	Context only; not the target used
Vacuum specific impulse	309.7 s	Independent output comparison
Total propellant flow	5.08 kg/s (11.2 lb/s)	Qualified output comparison

Input-versus-output classification

Category	Items	Interpretation
Historical inputs	Propellants, Pc, O/F, expansion ratio, 0 Pa ambient, target thrust	Reproducing them shows case setup, not predictive success
Model assumptions	Combustion efficiency 0.96; nozzle efficiency 0.98; frozen-flow nozzle	User/default modeling choices; not NASA historical measurements
Independent outputs	Vacuum Isp, total mass flow, effective c*, throat and exit diameters	Calculated after inputs and assumptions are fixed
Target closure	Vacuum thrust = 15,568 N	Solver sizes the throat to hit the entered target; not validation

Historical comparison

Differences are RPL minus the cited reference. Percent differences use the cited reference as the denominator. Significant figures are kept consistent with the public source quality.

Quantity	Historical	RPL	Difference	Class
Vacuum thrust	15,568.8 N design target	15,568 N	-0.8 N (-0.005%)	Entered target closure
Vacuum Isp	309.7 s	307.9659 s	-1.7341 s (-0.5599%)	Independent output
Total flow	5.08 kg/s	5.15477 kg/s	+0.07477 kg/s (+1.472%)	Qualified output
Effective c*	1,724.7108 m/s injector test reduction	1,616.64 m/s	-108.0708 m/s (-6.2660%)	Qualified diagnostic only
Throat diameter	No directly comparable value used	111.86 mm	Not calculated	Derived geometry
Exit diameter	No directly comparable value used	755.35 mm	Not calculated	Derived geometry

Vacuum Isp interpretation

The -0.56% difference is small enough to be useful evidence that this bounded first-pass case is in the historical performance neighborhood. It does not isolate model accuracy because Pc, O/F, expansion ratio, propellants, and target thrust were supplied, while combustion and nozzle efficiencies were assumed. NASA Table I provides no uncertainty band for its 309.7 s value.

Mass-flow interpretation

The direct difference from the published 5.08 kg/s is +1.47%. This comparison is qualified because Table I pairs its flow with the separate 3,450 lbf typical duty-cycle value, while the RPL case is sized to the 3,500 lbf design target. The RPL mass flow is also conditioned on its predicted Isp.

Thrust interpretation

The reported RPL thrust is not an independent prediction. Target-thrust sizing changes throat area until the entered 15,568 N target is met. The comparison verifies unit conversion and target closure only.

Geometry result

For the entered 45.6 area ratio, RPL derives a 111.86 mm throat and 755.35 mm exit. No independent historical diameter pair is used here, so these values are deliverable cross-checks between the PDF, DXF, CLI result, and project rather than historical validation points.

Why the c* comparison is not apples-to-apples

Boeing D2-117060-2 reports a mean production Rocketdyne LM-ascent injector c* of 5,658.5 ft/s (1,724.7108 m/s) across 20 production injectors, with injector-to-injector standard deviation of 7 ft/s. This is a test-reduced injector value, not measured in-flight or full-duty-cycle engine c*.

Important convention and hardware differences

- Tests used an ablative liner with a water-cooled throat and unsaturated propellants.
- Reduction used measured injector-end pressure, pre-run throat area, and measured flows.
- Boeing explicitly warns that different pressure-tap locations made even Bell and Rocketdyne c* data not directly comparable.
- NASA TN D-7400 describes an additional factor used to relate ground-test hardware to the flight ablative chamber.

RPL effective versus raw table anchor

Quantity	Value	Difference from Boeing injector mean
RPL effective c*	1,616.64 m/s	-108.07 m/s (-6.27%)
RPL embedded table anchor	1,684.00 m/s	-40.71 m/s (-2.36%)
Explicit combustion efficiency	0.96	Applied to the table anchor to obtain effective c*

The application report states that the embedded thermochemistry table is referenced at 6.895 MPa, whereas this historical case uses 0.848 MPa. The effective -6.27% difference therefore combines source-pressure mismatch, test-reduction conventions, hardware differences, and the explicit 0.96 efficiency assumption. It is presented as an unfavorable but honest diagnostic, not as a clean validation failure or a combustion-efficiency inference.

Other assumptions and limitations

- Frozen-composition nozzle expansion; reacting-flow species are not solved in this case.
- Embedded CEA-referenced thermochemistry tables, not a case-specific external CEA run.
- First-pass correlations for feed, thermal, injector, and mass-budget sections remain screening-level.
- Public historical data are rounded and do not provide a unified uncertainty covariance for this comparison.
- The authentic report includes conditional sea-level fields; they are not used as historical LMAE results.

Application identity and reproduction

Application/build identity

Item	Recorded identity
Application	RPL Engine Workbench v1.0.0
Run mode	First-pass sizing; target vacuum thrust; embedded CEA-referenced table
Operating condition	Absolute vacuum: ambient pressure 0 Pa
PDF provenance	Authentic GUI export; Qt 6.11.1 producer; 11 A4 pages; trial watermark retained
GUI report SHA-256	058051cfda6a95a4961461258fb09664 f8009519aba5a03776649ad126c74308
Production DMG	40,367,528 bytes; SHA-256 960958445dd7c1ed8cc951a35a49906a ffd608e92e7e37bdc009cb492a44cce9

Reproduce the reference case

1. Open RPL_LMAE_REFERENCE_PROJECT.rplproj in RPL Engine Workbench.

2. Confirm NTO/Aerozine 50, O/F 1.6, Pc 0.848 MPa, Pa 0 Pa, expansion ratio 45.6, and target thrust 15,568 N.

3. Run the first-pass solve and compare vacuum Isp, total mass flow, throat diameter, and exit diameter with the table in this document.

4. Trial users can reproduce the solve and export the watermarked PDF. Project saving and DXF export remain paid features; this package does not alter those gates.

Primary sources

1. J. C. Hooper III, *Performance Analysis of the Ascent Propulsion System of the Apollo Spacecraft*, NASA Lyndon B. Johnson Space Center, NASA TN D-7400 / JSC-S-354, December 1973. Design and nominal performance: printed p. 5, Table I (PDF p. 10); ground-to-flight c* discussion: printed pp. 11-12 (PDF pp. 16-17). Record: https://ntrs.nasa.gov/citations/19740005391
PDF: https://ntrs.nasa.gov/api/citations/19740005391/downloads/19740005391.pdf
Retrieved 2026-07-10.

2. C. E. Humphries and R. E. Taylor, *Apollo Experience Report - Ascent Propulsion System*, NASA Manned Spacecraft Center, NASA TN D-7082 / MSC-S-341, March 1973. 3,500 lbf requirement and propellant basis: printed pp. 1-2 (PDF pp. 5-6). Record: https://ntrs.nasa.gov/citations/19730010173
PDF: https://ntrs.nasa.gov/api/citations/19730010173/downloads/19730010173.pdf
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3. J. P. B. Cuffe, *Apollo Spacecraft Engine Specific Impulse, Part II*, Boeing Company, D2-117060-2-PT-2 / NASA-CR-112414, 28 October 1968, contract NASw-1650. Rocketdyne production-injector c*: Section 4.5.1, printed p. 70 (PDF p. 78), Appendix B printed pp. 88-90 (PDF pp. 99-101); comparability warning: Section 2.3.2.3, printed p. 14 (PDF p. 22). Record: https://ntrs.nasa.gov/citations/19700027045
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