# BOEING





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MODEL NO. CONTRACT NO. NASw-1650

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## ABSTRACT

This report describes the methods used to determine the vacuum specific impulse of the following primary propulsion system engines used in the Apollo Service and Lunar Modules:

- LM Descent Engine (TRW)
- SM Engine (Aerojet)
- LM Ascent Engine (Bell)
- LM Ascent Engine Injector Back-up Program (Rocketdyne)

A description of the acceptance test procedures, instrumentation and performance analysis methods used on the different engines is given, together with the manufacturers' estimate of the uncertainty in the quoted value of specific impulse.

CONTENTS

## INTRODUCTION

This interim report is in partial fulfillment of NASA Technical Direction NASw-1650 #28, "Apollo Rocket Engine Altitude Specific Impulse Ratings". (Reference 1) The report describes the methods used to determine vacuum specific impulse of the primary propulsion engines used in the Apollo Service and Lunar Modules. As directed by APO, no attempt has been made to assess the merits of the different methods. The information presented has been obtained from various reports and from a series of presentations given by the engine manufacturers at MSC-Houston on February 11 through 16. There has been no opportunity for a first hand inspection of the test facilities.

The following engines are discussed in this report:

LM Descent Engine (TRW)
SM Engine (Aerojet)
LM Ascent Engine (Bell)
LM Ascent Engine Injector Back-up Program (Rocketdyne)

## 1. Discussion of Problems in Measuring Vacuum Specific Impulse

All of the engines considered in this report are pressure fed, ablative cooled space engines of high area ratio, using nitrogen tetroxide and aerozine-50 propellants at a nominal mixture ratio of 1.6:1. By boost engine standards, they run at low chamber pressure. The use of ablative cooled chambers and high area ratio nozzles results in special problems in obtaining reliable values of vacuum specific impulse.

### 1.1 Effect of Ablati Cooled Chambers

The chambers are constructed of refrasil fiber and a plastic binder. They have a limited allowable ground test run life; hence only the final acceptance tests are performed using flight chambers. This limits the number of repeat acceptance tests which can be carried out on an engine. The ablation which occurs with run time changes the finish and contour of the combustion chamber and throat. The resulting variation in throat area and in flow losses can cause a change in specific impulse with run time. For example, with the TRW LM Descent Engine, this amounts to a degradation of some 3 seconds in $I_{sp}$ over the nominal duty cycle. During a firing, the effective throat area may increase, because of erosion, or it may decrease, due to thermal effects and glass flow through the throat. Because of a combination of these effects, the throat area may first decrease and then increase on a long duration firing. This change in throat area makes measurement of C* (see Note) difficult, because throat area must be known in order to calculate C*. If the throat erosion is asymmetric, a change in thrust vector will result. The reduction in chamber weight with run time can also cause an apparent change in thrust vector.

NOTE:

$$C^*, \text{ by definition} = \frac{\text{chamber pressure} \times \text{throat area} \times g}{\text{propellant flow rate}}$$

where g is gravitational ratio of lb force/lb mass. Since

$I_{sp} = \dfrac{\text{thrust}}{\text{flow rate}} = \dfrac{C^*, C_f}{g}$ , a value of $I_{sp}$ can be obtained

from C*, if $C_f$ Thrust coefficient $= \dfrac{\text{thrust}}{\text{chamber pressure} \times \text{throat area}}$

is known. However, though flow rates and chamber pressure can be measured during a test, throat area cannot be, so pre-test or post-test measurements must be used.

## 1.2 Effect of High Area Ratio Nozzles

If the spacecraft engines were run at sea-level ambient pressure, flow separation would occur in their nozzles. Flow separation can cause side loads & nozzle damage; it also makes the prediction of altitude thrust from test data difficult. If a nozzle is flowing full, vacuum thrust is simply the measured thrust added to the product of the nozzle exit area and ambient pressure. A nozzle will flow full so long as the static pressure at its exit plane is greater than approximately 40% of the local ambient pressure. If this static pressure is less than 40% of the local pressure, separation will occur, with re-attachment (which may be unstable) within the nozzle. Therefore, when these engines are tested with their ablative chambers, they have to either be fired in an altitude cell or else have their nozzle extensions removed.

## 1.3 Sources of Error in Obtaining Vacuum Specific Impulse ($I_{sp\ vac}$)

Vacuum specific impulse is simply the ratio of engine thrust, at an ambient pressure of 0 psia, to propellant mass flow rate. Corrected vacuum specific impulse is vacuum specific impulse at nominal operating conditions

### 1.3.3 Performance Variation with Run Time

Quantitative data on performance variation with run time can only be obtained by firing an engine under full duty cycle conditions. Therefore, for production engines*, information obtained on the engine qualification and development test programs must be used. An indication of engine-to-engine variability on production engines can be obtained from the results of the injector-ablative thrust chamber compatibility tests.

At the presentations at MSC in February, there was considerable discussion on measurement error, less on correction error and very little on performance variation with run time.

### 1.4 General References


1. Technical Direction NASw-1650 Serial #28. January 9, 1968. C. C. Gay, Jr. Reference MAT-2

2. Instrumentation Error Analysis, Report of observations at TRW and AGC, June 19 - 23, 1967, and BAC, July 6 - 7, 1967. July 19, 1967. By R. K. McSheehy.

3. Trip Report to MSC Houston 11 through 16 February 1968. J. P. B. Cuffe. 20 February 1968.


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* This applies to ablation-cooled engines, where the thrust chamber and nozzle service life is ordinarily not much greater than the design duration for mission application.

Notation

F = Thrust lb.

$\dot{M}_{total}$ = Total propellant flow rate lb/sec

$p_a$ = Pressure in altitude facility psia

$A_t$ = Nozzle throat area in$^2$

$A_e$ = Nozzle exit area in$^2$

$p_{cc}$ = Measured thrust chamber pressure psia

$P_T$ = Total pressure at nozzle throat psia = f. $p_{cc}$.

g = gravitational ratio lb force/lb mass = 32.172 ft/sec$^2$

$I_{sp}$ = Specific impulse sec

$C_f$ = Thrust coefficient = $\frac{F}{P_T . A_T}$.

C* = Characteristic velocity ft/sec =

$$\frac{P_T . A_T . g}{\dot{W}_{total}}.$$

$$I_{sp} = \frac{C_f . C^*}{g} \text{ sec.}$$

| ENGINE | TRW L |
|---|---|
| Special Problems in Obtaining Vacuum Specific Impulse | Wide |
| Acceptance Test Procedure | Head e<br>altitu<br>A<br>Engine<br>facili |
| Method of Obtaining Specific Impulse | A comb<br>$I_{sp_{H_1}}$<br>$I_{sp_{H_2}}$<br>where<br>engine<br>$I_{sp_{V_2}}$<br>where<br>head e |
| Claimed Uncertainty of $I_{sp}$ When Corrected to Nominal Operating Conditions | 3 σ J<br>Thrust<br>Level<br>$I_{sp}$<br>Uncert |

FOLDOUT FRAME 1

## 2. SUMMARY

<table>
<tr><th></th><th>TRW LM DESCENT ENGINE</th><th>AEROJET SPS ENGINE</th></tr>
<tr><td>ecific</td><td>Wide throttling range</td><td>No altitude test facility available for production engines</td></tr>
<tr><td></td><td>Head end assembly performance tests in altitude facility<br>↓<br>Ablative compatibility firings<br>↓<br>Engine performance tests in altitude facility</td><td>Injector performance tests (at ambient pressure)<br>↓<br>Ablative compatibility firings<br>↓<br>Engine firings, without nozzle exte[illegible] (at ambient pressure)</td></tr>
<tr><td></td><td>A combination of the following:<br><br>$I_{sp_{H_1}} = \frac{F + p_a A_e}{\dot{M}_{total}}$ (engine tests)<br><br>$I_{sp_{H_2}} = \frac{C^* \cdot C_f}{g}$ (engine tests)<br><br>where $C_f$ is empirically derived from engine tests.<br><br>$I_{sp_{V_2}} = \frac{C^*}{g} C_{f_v}$ (head end assembly tests)<br>where $C_{f_v}$ is empirically derived from head end assembly and engine tests.</td><td>$I_{sp} = 0.05338 \, C^*_{inj}$<br><br>where $C^*_{inj}$ = $C^*$ measured on inject[illegible] performance tests.<br><br>0.05338 = empirical correlation fac[illegible] obtained from engine qualification test program in J-3 altitude test ce[illegible] at AEDC, Tullahoma</td></tr>
<tr><td>of $I_{sp}$<br>minal</td><td>3σ $I_{sp}$ uncertainty (for LM-3 engine)<br><table><tr><th>Thrust Level</th><th>Full Thrust</th><th>50%</th><th>25%</th></tr><tr><td>$I_{sp}$ Uncertainty</td><td>±1.6 sec</td><td>±2.8 sec</td><td>±2.4 sec</td></tr></table></td><td>3σ $I_{sp}$ uncertainty on all production engines ± 1.1 seconds.</td></tr>
</table>



FOLDOUT FRAME 2

| NGINE | BELL LM ASCENT ENGINE | LM ASC (ROCKETDYNE |
|---|---|---|
| ost facility available n engines | | Short time that operation. Altitude test fa completed Februa |
| ormance tests (at ure) ↓ compatibility firings ↓ s, without nozzle extension ressure) | Injector performance tests (at ambient pressure) ↓ Ablative compatibility firings ↓ Engine performance tests in altitude facility | Similar to Bell |
| $C^*_{inj}$ C* measured on injector ests. irical correlation factor engine qualification in J-3 altitude test cell ahoma | $I_{sp} = \frac{F + p_a A_e}{\dot{M}_{total}}$ (Engine tests) | $I_{sp} = \frac{F + p_a A_e}{\dot{M}_{total}}$ |
| tainty on all ines ± 1.1 seconds. | 3 σ $I_{sp}$ uncertainty (typical) ± 1.4 seconds | $I_{sp}$ uncertainty, ± 0.6 seconds |

FOLDOUT FRAME 3

| BELL LM ASCENT ENGINE | LM ASCENT ENGINE (ROCKETDYNE BACK-UP INJECTOR) |
|---|---|
| | Short time that program has been in operation. Altitude test facility modifications completed February 1968. |
| Injector performance tests (at ambient pressure) ↓ Ablative compatibility firings ↓ Engine performance tests in altitude facility | Similar to Bell LM Ascent Engine |
| $I_{sp} = \frac{F + p_a A_e}{\dot{M}_{total}}$ (Engine tests) | $I_{sp} = \frac{F + p_a A_e}{\dot{M}_{total}}$ (Engine tests) |
| 3 σ $I_{sp}$ uncertainty (typical) $\pm$ 1.4 seconds | $I_{sp}$ uncertainty, to 95% confidence, $\pm$ 0.6 seconds |

FOLDOUT FRAME 3

FOLDOUT FRAME 4

## 3. TRW LM Descent Engine

### 3.1 Introduction

The LMD gimballed thrust chamber is of ablative construction up to an area ratio of 16:1 and a crushable radiation cooled nozzle extension is fitted which brings the overall area ratio up to 47 1/2:1. The engine is restartable and can be continuously throttled over a range of 10 to approximately 70% full thrust, as well as being run at full thrust. The injector is a single variable area concentric (oxidizer inside fuel) pintle type. Variable area cavitating venturis control the propellant flow rates when the engine is throttled: at full thrust, flow control is by trim orifices and injector position. The wide thrust range (10:1) and low thrust chamber pressure at minimum thrust (approximately 12 psia) result in special difficulties in measuring the performance of this engine. The considerable mechanical complexity of this engine can also cause repeatibility problems.

#### 3.1.1 Nominal Operating Conditions

| | |
|---|---|
| Propellant temperature | 70° F |
| Propellant feed pressures at F.T.P.* | 220 psia |

Propellant feed pressures, under throttled conditions =

$$235 - 15 \left( \frac{\text{Flow rate}}{\text{Flow rate at F.T.P.}} \right)^2 \text{ psia}$$

| | |
|---|---|
| Ambient pressure | 0 psia |
| Mixture ratio | 1.6:1 |

#### 3.1.2 Nominal Performance (Under the Above Conditions)

At F.T.P.*

| | |
|---|---|
| Oxidizer flow rate | 19.886 lb/sec |
| Fuel flow rate | 12.429 lb/sec |
| Total flow rate | 32.315 lb/sec |
| Thrust | 9850 lb |
| $I_{sp}$ | 305 sec |
| Thrust chamber pressure | 104 psia |

*NOTE: F.T.P. = Full thrust position (of throttle). The actual thrust, with the throttle in this position, is dependent upon the amount of throat erosion which has occurred.

3.1.3 Nozzle Conditions

The pressure ratio across the 47 1/2:1 area ratio nozzle is approximately 700:1: that is, the chamber pressure is approximately 700 times the nozzle exit static pressure. To ensure that the nozzle is not over expanded, the ambient pressure at the nozzle exit has to be less than:

$$\frac{104}{700} = 0.15 \text{ psia at F.T.P.}$$

and

$$\frac{12}{700} = 0.017 \text{ psia at 10\% thrust.}$$

Assuming that separation occurs when the nozzle exit static pressure is less than 40% of the ambient pressure at the nozzle exit, the minimum pressure required to ensure that nozzle separation does not occur is:

0.37 psia at F.T.P. and 0.042 psia at 10% thrust

3.2 Acceptance Test Procedure

Calibration and acceptance testing is carried out in three phases. The head end assembly(H.E.A.) is first calibrated and accepted, using a slave water cooled thrust chamber, then the erosion characteristics of the injector are determined by two firings using a fiber glass chamber and finally the complete engine, with its flight ablative thrust chamber, but using a slave nozzle extension, is acceptance tested.

3.2.1 Head End Assembly Calibration and Acceptance Firings

The head end assembly is fitted to a water cooled steel chamber with a 2:1 expansion ratio. These tests are carried out in one of the Vertical Engine Test Stands (VETS). To enable the injector assembly to be run at low throttle setting, a diffuser/ejector exhaust system is used. (A-1 and A-2 stands only).

Test Conditions

Propellant temperatures will be 70 ± 5° F.
Propellant feed pressures, at engine interface, 220 ± 5 psia at F.T.P. and, under throttled conditions,

$$\approx 235 - 15 \left(\frac{\text{Flow rate}}{\text{Flow rate at F.T.P.}}\right)^2 \pm 5 \text{ psia}$$

On the calibration tests, the H.E.A. must be adjusted, where necessary, to ensure that the mixture ratio, when corrected to nominal inlet conditions, is within the following limits:

| Thrust Level | F.T.P. | 50% | 25% |
|---|---|---|---|
| Mixture Ratio | 1.6 ± 0.014 | 1.6 ± 0.025 | 1.6 ± 0.036 |

| 10% |
|---|
| 1:6 $\substack{+0.1 \\ -0.45}$ |

When satisfactorily calibrated, two H.E.A. acceptance tests have to be carried out over the full range of 10% thrust to F.T.P.

### 3.2.2 Ablative Throat Streak Tests

The calibrated H.E.A. is fitted to a slave combustion chamber with an ablative throat of fiber glass. Two tests are carried out at F.T.P., with propellants at the same conditions as required for the H.E.A. acceptance tests.

The injector erosion characteristics are measured by the time taken to erode the fiber glass throat area by 20%: this is referred to as the $T_{120}$ time. It is calculated as follows:

$$T_{120} = \frac{(0.20)(T_2 - T_1)}{\frac{C^*_1}{C^*_2} - 1.0}$$

where
- $T$ = time (seconds)
- $C^*$ = $C^*$ calculated from pre-run throat area
- Suffix 1 = conditions at 15 seconds
- Suffix 2 = conditions at last F.T.P. data point
- $T_{120}$ time must be greater than 75 seconds

This test is carried out after the H.E.A. has been adjusted to the correct mixture ratio at F.T.P., since the engine's erosion characteristics are greatly affected by mixture ratio: increasing the mixture ratio from 1.6 to 1.7 results in the erosion rate increasing by a factor of 4.

3.2.3 Engine Acceptance Test

The H.E.A. is now built onto its flight combustion chamber, fitted with a slave nozzle extension. Engine firings are carried out in the High Altitude Test Stand (HATS). The engine is mounted vertically in a capsule. The two stage steam ejector/exhaust diffuser system can maintain an ambient pressure of approximately 0.05 psia at F.T.P. and of 0.1 psia at 10% thrust. Flow separation therefore probably occurs at approximately 20% thrust. Because the engines' calibration may have been affected by the assembly of the H.E.A. to the flight chamber, an initial checkout firing may be carried out.

Two acceptance firings are required, with the engines' thrust varied between 15% and F.T.P. The conditions for the firings are the same as for the H.E.A. acceptance tests, except that the capsule pressure must be less than 0.2 psia.

3.3 Instrumentation

3.3.1 Thrust

In the HATS facility, the engine is mounted vertically and thrust is measured on a single dual bridge strain gage load cell. Thrust vector is not measured, the engine having been optically aligned prior to installation. Calibration is carried out with the engine installed and with the feed lines pressurized before and after each test. Calibration loading is with dead weights applied on a beam balance, knife edge mounted, which has a mechanical advantage of 20:1. The load cell is calibrated over a range of 1 through 10 K lb. Hysteresis effects are ignored, since TRW considers that they are cancelled out by the dynamic effects of an engine firing.

TRW quotes the following levels of uncertainty for their thrust measuring systems:

| Nominal Load K Lb | 1 K | 2 K | 4 K | 6 K | 8 K | 10 K |
|---|---|---|---|---|---|---|
| $3\sigma$ random uncertainty % | 0.41 | 0.19 | 0.12 | 0.04 | 0.13 | 0.08 |
| Bias uncertainty % | 0.385 | 0.175 | -0.020 | -0.024 | -0.049 | +0.030 |
| Total uncertainty % | 0.795 | 0.365 | 0.140 | 0.064 | 0.179 | 0.110 |
| Total uncertainty lb | 7.95 | 7.30 | 5.60 | 3.84 | 14.32 | 11.0 |

3.3.2 Flow Rates

On both the HATS and VETS test stands, flow rates are measured by turbine flow meters, two being installed in series in each facility propellant feed line. Flow meters are calibrated with their working fluids at normal working pressure over a range of 60 to 600 CPS. Calibration is either done in place, using weight tanks or else using a ballistic calibrator. The ballistic calibrator is a piston displacement device, which forces propellant through the flow meter which is being calibrated. Flow rates and pressures are controlled by the pressure of the gas driving the piston and by the restriction down stream of the flow meter which is being calibrated. TRW has recommended that the ballistic calibrator be used to calibrate all of the 1 1/2 inch flow meters used on the LMD program, since there is less data scatter when flow meters are calibrated by this method than when weight tanks are used.

Since the flow meters are calibrated over the full range of engine flow rates and this information is used to curve fit a flow meter calibration factor which is not constant, but is dependent upon flow meter speed, TRW quotes a value of flow measurement uncertainty that is almost constant. Between 4 and 10 K, the $3\sigma$ flow uncertainty is given as 0.217 % of nominal flow rate at the thrust level being considered and between 1 and 2 K, the uncertainty is 0.22%.

### 3.3.3 Pressure

Pressure transducers are dead weight calibrated over their full operating range. This data is fitted to a least square linear calibration equation for each transducer. Hysterisis effects are ignored (see 3.3.1)

Cell pressure is measured by three 0 - 0.5 psia pressure transducers.

A single 0 - 10 psia hypergolic $\Delta p$ measurement is taken. TRW quotes a total $3\sigma$ uncertainty of 1.8% or 0.18 psia on hypergolic $\Delta p$ measurement which is the difference between the oxidizer & fuel pressures at the engine interface.

### 3.3.4 Uncertainty of Vacuum Specific Impulse Measurements

TRW quotes the following values for $3\sigma$ vacuum specific impulse uncertainties:

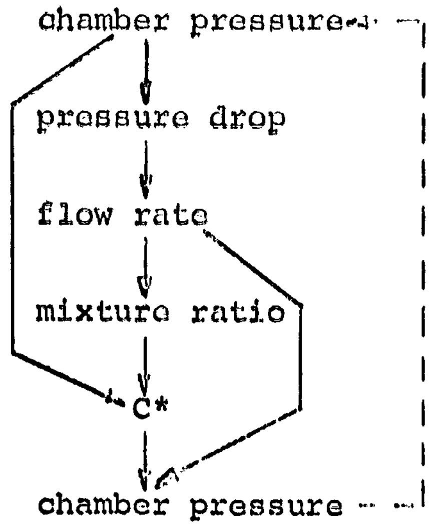
| Thrust Level | 1 K | 2 K | 4 K | 6 K | 8 K | 10 K |
|---|---|---|---|---|---|---|
| $I_{sp\ vac}$ uncertainty | 4.97 sec | 3.47 sec | 1.44 sec | 1.07 sec | 0.98 sec | 0.73 sec |

These values are for test vacuum specific impulse, obtained from measurements of thrust, cell pressure and flow rates.

They do not include any errors resulting from the correction of the data to nominal interface conditions.

### 3.4 Analysis

The calculations used to reduce test data to nominal conditions are dependent upon the engine's thrust setting. At F.T.P., flow is controlled by the hydraulic resistance of the total feed system (engine interface to thrust chamber pressure). Correction to nominal conditions is made by using the test values of hydraulic resistance and iterating around a chamber pressure loop, i.e.




The C* → mixture ratio → chamber pressure and $C_f$ → mixture ratio → chamber pressure information used by the program is partly theoretical and partly empirical.

When the engine is throttled and the flow control valves are cavitating, flow is simply determined by valve position, feed pressure and propellant saturation vapor pressure: there is therefore no need to iterate, since thrust chamber pressure variations do not affect flow rates.

The quoted "best estimate" of specific impulse given for each engine is obtained from both VETS (head end assembly) and HATS (complete engine) acceptance test data. The HATS specific impulse is calculated in two ways, by using measured thrust and flow rates ($I_{sp}\ H_1$) and by using measured C* and an input thrust coefficient ($I_{sp}\ H_2$). The VETS specific impulse is obtained by using measured

C* and an input thrust coefficient, which has been empirically determined from other VETS and HATS data ($I_{sp}$ $V_2$). At low thrust settings (10%), specific impulse is obtained solely from VETS data. The weighting factors used in calculating the best estimate of specific impulse from these three sources of data are calculated from the inverse of the observed variances of the different methods, thereby putting greater weight on the data with the lower variability.

An example of the reduction in variability in specific impulse, using the "best estimate", is shown in the following table of data from engine #1037.

| | Operating Point | | |
|---|---|---|---|
| | F.T.P. | 50% | 25% |
| $I_{sp}$ $H_1$ (sec) | 304.2 | 300.7 | 297.3 |
| $\sigma$ $I_{sp}$ $H_1$ (sec) | $\pm$ 0.82 | $\pm$ 1.16 | $\pm$ 1.00 |
| $I_{sp}$ (best estimate) sec | 303.8 | 301.0 | 298.2 |
| $\sigma$ $I_{sp}$ (best estimate) sec | $\pm$ 0.53 | $\pm$ 0.92 | $\pm$ 0.85 |

### 3.5 Operation

The dwell times on the H.E.A. tests at each throttle setting vary between 6 and 9 seconds, when throttled, and 14 or 20 seconds at F.T.P. It is doubtful that the engine's operation would be completely stabilized after only 5 seconds. However, the limited acceptance test run time available on the flight ablative thrust chamber and the requirement for data slices at six different thrust settings result in these short dwell times.

The close tolerances on feed system conditions result in only very small corrections being applied when the test data is normalized to standard conditions. As an example, on the acceptance tests of the LM-3 descent engine, the maximum variations on both VETS and HATS tests of inlet conditions from nominal were 2.9 psi in interface pressure and 2.8° F in propellant temperature. The maximum corrections, resulting from the adjustment of test data to nominal inlet conditions, were only 2.9 ft/sec in C* on the VETS tests and 0.1 second in vacuum $I_{sp}$ on the HATS firings. However, TRW indicated that there was need for more empirical data, preferably obtained from tests specifically designed for performance gains, to accurately predict engine performance over the full range of possible operating conditions.

There is an approximate 0.26 second reduction in specific impulse for every 1% increase in throat area. If the mixture ratio and propellant temperatures are known for a particular mission, then the throat erosion, and hence $I_{sp}$ degradation, can be calculated from the $T_{120}$ time and thrust program. For a lunar descent using the LM-3 engine, approximately 15% increase in throat area or 3.9 seconds decrease in specific impulse would be expected. This degradation in $I_{sp}$ is higher than average, due to the fact that LM-3 engine #1030 had a low $T_{120}$ time (83.0 and 78.9 seconds - see Section 3.2). Any increase in mixture ratio above the predicted level would increase the rate of erosion.

It is usual to see engine performance shifts when the H.E.A. is removed from the steel chamber and fitted to the fiber glass or flight ablative one, but these are mainly shifts in mixture ratio and not in specific impulse.

The magnitude of the possible error in thrust coefficient in using a slave nozzle extension for the engine acceptance tests, instead of the flight nozzle, is not known.

## 3.6 References


1. Notes taken at TRW presentation at MSC, 12 February 1968.
2. TRW report 01827-6115-T000, 11 October 1967, "TRW LM Descent Engine Phase B Qualification Test Program Summary Report".
3. TRW report 01827-6122-T000, 11 December 1967, "TRW LM Descent Engine #1030 Acceptance Test Performance Report" (LM-3 Engine).
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# 4. Aerojet SPS Engine

## 4.1 Introduction

This report discusses the performance of the Block II engine, fitted with a Mod IV injector of 5-4-4 configuration (5 baffles, which are 4" long at the hub and 4" long at their tips). The engine has an overall area ratio of 62.5:1, a radiation cooled nozzle of columbium and titanium being fitted to the 6:1 area ratio ablative cooled chamber. This chamber has an injector area to throat area ratio of 2:1, with a straight taper of 8 1/2° half angle from the injector to the throat. The engine is non-throttleable, gimballed and has an active P.U. control valve in the oxidizer feed system: being upstream of the engine interface, the P.U. valve is not considered to be part of the engine system. Flow calibration is by orifices in the feed systems downstream of the interface connections. Aerojet has no test facilities at Sacramento for running the engine under simulated altitude conditions.

### 4.1.1 Nominal Operating Conditions

| | |
|---|---|
| Ambient pressure | 0 psia |
| Oxidizer inlet pressure | 160 psia |
| Oxidizer inlet temperature | 70° F |
| Fuel inlet pressure | 166 psia |
| Fuel inlet temperature | 70° F |
| Throat area | 121.680 inches$^2$ |

### 4.1.2 Nominal Performance (Under the Above Conditions)

| | |
|---|---|
| Mixture ratio | 1.6:1 |
| Thrust chamber pressure | 97 psia (now 99psia) |
| Specific impulse | 311 sec. (minimum) |
| Thrust | 20,200 lb. |
| Total Flow Rate | 65.0 lb/sec |
| Fuel Flow Rate | 25.0 lb/sec |
| Oxidizer Flow Rate | 40.0 lb/sec |

### 4.1.3 Nozzle Conditions

| Nozzle Area Ratio | 1.5:1 | 6:1 | 67.5:1 |
|---|---|---|---|
| Nozzle pressure ratio | 5.4 | 44 | 1100 |
| Static pressure at nozzle exit | 18 psia | 2.2 psia | 0.113 psia |
| Minimum ambient pressure, for nozzle operation without separation | | 5.5 psia | 0.28 psia |

The above calculations are only approximate, but indicate the reasons for the following:

4.1.3.1 A nozzle with a 1.5:1 area ratio will flow full without separation at sea level.

4.1.3.2 Separation will occur if the engine is run at sea level, even if the nozzle extension is removed, since the area ratio will be 6:1.

4.1.3.3 To obtain meaningful thrust data, the complete engine should be tested in an environment where the ambient pressure is less than 0.28 psia and preferably approximately 0.1 psia or lower. When the engine was tested in J-3 cell at AEDC, Tullahoma, the cell pressure, under steady state conditions, was maintained at a level of less than 0.1 psia.

## 4.2 Acceptance Test Procedures

Because Aerojet do not have the facilities for firing the SPS engine under simulated altitude conditions, engine performance must be derived indirectly from measured injector data. The correlation between injector performance and engine performance is obtained empirically from data obtained on the engine qualification test program at AEDC, Tullahoma.

### 4.2.1 Injector Test

The injector tests are carried out with the injector fitted to an uncooled steel thrust chamber of 1.5:1 area ratio. These tests are carried out under the following conditions:

| | |
|---|---|
| Test Duration | 5 seconds |
| Propellant Temperature | 70° F ± 30°, with fuel temperature within 10° F of oxidizer |
| Chamber pressure | 97 psia ± 3 psi (± 3%) |
| Mixture ratio | 1.6 ± 0.02 (± 1.25% ) |

The temperature of the steel thrust chamber must be less than 50° F above ambient.

Data is taken during the last two seconds of a test, up to one-half second before cut off. A minimum of three acceptable tests is required. These tests are primarily intended to give injector performance data: the engine mixture ratio and thrust calibration is not carried out at this time.

Throat area is measured prior to the first test. Thrust is measured, but is not now used for engine performance estimation, because this is now calculated from injector C* performance in the following manner:

$$I_{sp\ vacuum} = R \cdot C^*_{inj} = \frac{R \cdot P_c \cdot A_t \cdot g}{\dot{W}_{total}}$$

where:

$R$ = correlation factor, empirically determined from engine tests in the J-3 altitude facility at AEDC.

$C^*_{inj}$ = measured C* on injector tests using steel thrust chamber, ft/sec

$P_c$ = measured thrust chamber pressure on injector tests, psia.

$A_t$ = steel thrust chamber throat area, inches$^2$

$g$ = gravitational constant = 32.174 ft/sec$^2$

$\dot{W}_{total}$ = total propellant flow rate on injector tests, lb/sec

The possible errors resulting from the use of this equation will be discussed in Section 4.4.

### 4.2.2 Injector Ablative Compatibility

An ablative chamber of 1.5:1 or 6:1 area ratio is used. A single firing of 305 $\pm$ 5 second duration is required under the following conditions:

| | |
|---|---|
| Chamber pressure | 97 psia $\pm$ 5 psi ($\pm$ 5%) |
| Mixture ratio | 1.6:1 $\pm$ 0.1 ($\pm$ 6 1/4%) |

An injector's compatibility is acceptable, so long as there are no gouges or streaks of greater than 0.25 inch depth in the ablative material after this test.

### 4.2.3 Engine Test

The injector is fitted with its flight thrust chamber and ball valve assembly. The firing is carried out under local ambient pressure conditions, the nozzle extension not being fitted. The engine is tested in a simulated service module propulsion syst m test stand which allows the engine to be gimballed when firing. Tests are carried out under the following conditions:

| | |
|---|---|
| Propellant temperature | 70° F ± 30° F |
| Fuel temperature = Oxidizer temperature ± 10° F | |
| Propellant interface pressures | (at start) |
| Oxidizer interface pressure | 176 ± 4 psia |
| Fuel interface pressure | 178 ∓ 4 psia |
| Difference between interface pressures < 4psi | |
| Propellant interface pressures | (steady state) |
| Oxidizer interface pressure | 160 ± 4 psia |
| Fuel Interface pressure | 166 ∓ 4 psia and must also be in the range of oxidizer interface pressure + 6 ± 2 psi |

The orifices in the propellant feed lines are sized to ensure that the engine performance, when corrected to nominal operating conditions, is within the following limits:

Mixture ratio 1.60:1 ± 0.02 (± 1.25%)

Thrust (calculated from $I_{sp\ vac}$, which in turn has been calculated from $C^*_{inj}$) 20,300 ± 500 lb (±2 1/2%)

## 4.3 Instrumentation

The instrumentation discussed in this section is that used by Aerojet for measuring the performance of the SPS engine on injector tests. The measurement uncertainty quoted by Aerojet is of the following form:

uncertainty = bias error ± 2 $S_o$

where $S_o$ = standard deviation of the random error.

The uncertainties quoted by Aerojet will therefore be less than those which would have been obtained had the NASA recommended expression of uncertainty = Bias error ± 3 $S_o$ been used. Unless otherwise stated, these uncertainties are quoted in terms of percentage of measuring system full range.

### 4.3.1 Thrust

Thrust is measured on the injector performance tests, but is not used for engine performance predictions. A single load cell, with a dual bridge network is used.

### 4.3.2 Flow Rates

Two turbine type flow meters in series are used in each propellant feed line. There are at least thirty diameters of straight pipe upstream of the fuel flow meters, which are not in-place calibrated.

The fuel flow meters are calibrated in a flow rig, using water as the working fluid. Several flow meters may be calibrated at any time by mounting them in series in the test section. The flow meter pulse counters are operated by level switches in the water header tank. Knowing the volume of water between the level switches in the header tank, the calibration factors of the flow meters may be obtained. Since the engine is not throttled, a constant flow meter calibration factor is used. Only density corrections are made to convert flow meter water calibration data for use with Aerozine-50. Aerojet quote an overall fuel flow measurement uncertainty of ± 0.372%: this value includes the reduction in uncertainty due to using two flow meters in series.

The oxidizer flow meters are in-place calibrated with nitrogen tetroxide against standard flow meters. These standard flow meters have been calibrated with nitrogen tetroxide in a special facility, using a 'pressure intensifier'. This device consists of a cylinder with a gas actuated piston connected to another piston of one third its area in a cylinder containing nitrogen tetroxide. The oxidizer is then forced through the flow meters which are being calibrated as secondary standards: several flow meters in series may be calibrated at once. Calibration is carried out with the nitrogen tetroxide at its nominal working pressure and temperature. The volume of liquid passed through the flow meters is calculated from the linear movement of the pistons. As with the fuel flow meters, a constant calibration factor is used. Including the effect of using two flow meters, Aerojet quote an overall oxidizer flow measurement uncertainty of ± 0.234%.

### 4.3.3 Pressures

Pressure transducers are laboratory calibrated, using an oil dead weight tester for high pressure transducers and a manometer for low pressure ones. Except for the transducers which will be used for measuring thrust chamber pressure, the output of the transducers is adjusted to conform to a standard linear calibration curve. Individual calibration curves are used for the transducers which measure thrust chamber pressure. The quoted uncertainty of the measurements using these transducers, which have been individually calibrated, is ± 0.148% , whereas the uncertainty of the measurements obtained from the transducers with the standard calibration is 0.206%.

### 4.3.4 Area

Throat area is calculated from eight measurements taken by two individuals. The quoted uncertainty in throat area ($1\sigma$) is 0.065% of measured area.

### 4.3.5 Temperature

The platinum resistance thermometers, used to measure propellant temperature, have a quoted uncertainty of 0.408% of 70° F.

## 4.4 Analysis

### 4.4.1 General

Because Aerojet do not have the altitude test facilities which would allow them to carry out performance evaluation tests on production engines under simulated high altitude conditions, a method of predicting engine altitude performance on the basis of sea level test data had to be developed.

Four Mod IV injectors were tested in the J-3 cell at AEDC, Tullahoma: each injector was tested in two flight configuration ablative thrust chambers fitted with the full nozzle extension. Prior to being tested at AEDC, the performance characteristics of the injectors had been obtained from tests at Sacramento, where the injectors were fitted to a steel thrust chamber of 1.5:1 area ratio (Section 4.2.1). Thrust was measured on these tests, so both injector C* and specific impulse (in 1.5:1 area ratio chamber) were obtained. When these injectors were tested at AEDC, and their vacuum specific impulse measured, correlation was made with both the injector C* and $I_{sp\ (1.5:1)}$ calibration data in the following way:

$$R = \frac{I_{sp\ (62.5:1\ \text{vacuum})}}{C^*_{inj}}$$

or

$$K = \frac{I_{sp\ (62.5:1\ \text{vacuum})}}{I_{sp\ (1.5:1\ \text{vacuum})}}$$

where: $I_{sp\ (62.5:1\ \text{vacuum})}$ = mean value of specific impulse obtained from AEDC tests on one injector, corrected to nominal conditions.

$I_{sp}$ (1.5:1 vacuum) = mean value of specific impulse obtained on injector performance calibration tests at Sacramento, with injector fitted to 1.5:1 area ratio steel chamber. Data corrected to nominal conditions.

$C^*_{inj}$ = mean value of C* obtained on injector performance calibration tests at Sacramento (see Section 4.2.1) Data corrected to nominal conditions.

Because the variability of K was found to be more than three times that of R (Ref. 5) Aerojet decided to predict engine vacuum performance on the basis of injector C* and not on specific impulse measured on the 1.5:1 area ratio steel chamber.

4.4.2 Analysis of $C^*_{inj}$

The throat area of the steel chamber was found to change with run time, so a least squares curve fit equation was developed to fit this data. By use of this equation, and the measured value of throat area prior to the first injector calibration firing, the best estimate of throat area on successive tests is obtained. The measured $C^*_{inj}$ is corrected to nominal conditions by a covariant equation. This equation corrects for differences between test and nominal conditions of chamber pressure, mixture ratio and propellant temperature and was empirically derived from test data (86 tests) on a Mod 0 injector. Aerojet consider that the characteristics of a Mod 0 injector are sufficiently similar to those of a Mod IV injector for this data to be valid. The mean of the corrected values of $C^*_{inj}$ are then calculated: any test with a corrected $C^*_{inj}$ which differs from the mean by more than 19 ft/sec is not considered to be acceptable. The mean $C^*_{inj}$ must be obtained from at least 3 acceptable tests (Section 4.2.1). The predicted vacuum specific impulse of an engine is simply the mean $C^*_{inj}$ of the injector used on that engine, multiplied by 0.05338, the empirical value of R. (Reference 1).

In the Aerojet report on the engine qualification tests at AEDC, (Reference 5) the value of R was given as 0.05333, with a standard deviation of 0.1%. The values given in the MSC presentation (Reference 1) were 0.05338 for R and 0.04% for its deviation. Aerojet explained that these differences were due to the use of the normalized curve fit throat area equation and to the change of nominal thrust chamber pressure from 9. to 99 psia. The different values in $C^*_{inj}$ of the injectors used in the engines qualification program are as follows:

| Injector Number | Reference 5 Aerojet Qual. Report $C^*_{inj}$ ft/sec | Reference 1 Aerojet MSC Presentation $C^*_{inj}$ ft/sec | Delta ft/sec |
|---|---|---|---|
| 105 | 5869 | 5871 | +2 |
| 104 | 5877 | 5872 | -5 |
| 115 | 5855 | 5851 | -4 |
| 103 | 5916 | 5905 | -11 |

These changes in $C^*_{inj}$ resulted in the deviation of R being reduced by a factor of 2 1/2.

The 1 $\sigma$ uncertainty of measured $C^*_{inj}$ was given in both References 1 and 5 as 0.162% or 9.5 ft/sec: of this, 0.065% or 3.8 ft/sec was allocated to uncertainty in throat area measurement. The C* correction of injector #103, due to the improvement in throat measurement accuracy, is approximately three times the quoted 1$\sigma$ uncertainty in throat area measurement.

It should be noted that this value of $C^*_{inj}$ uncertainty refers to the measured value of $C^*_{inj}$, and does not take into account uncertainties in correcting measured $C^*_{inj}$ to nominal conditions.

### 4.4.3 Uncertainty in Predicted Vacuum Specific Impulse

The Aerojet calculation of $I_{sp\ vac}$ uncertainty is based on the assumption that this uncertainty is caused by the RSS (root sum square) of the uncertainties in -

$C^*_{inj}$ measurement

R factor value

AEDC $I_{sp}$ uncertainty

$C^*_{inj}$ measurement uncertainty (1$\sigma$) is given as 9.5 ft/sec. But since this value is obtained from the mean of at least three acceptable tests, this uncertainty is reduced by a factor of $\sqrt{1/3}$.

The 1$\sigma$ deviation in R factor is given in Ref. 1 as 0.04%. This deviation does not take into account the effect of different thrust chambers, since, for any given injector, R is calculated from the mean value of $I_{sp}$ obtained on all of the tests with that injector at AEDC and each injector was fired in two different thrust chambers. The maximum difference in mean $I_{sp}$, due to change in thrust chamber, was with injector #115, which had a mean value of $I_{sp}$ of 312.8 sec with one thrust chamber and 311.4 sec with another. If R for injector #115 was calculated using a value of $C^*_{inj}$ of 5851 ft/sec., then for each thrust chamber the following values would be obtained -

| | $I_{sp}$ vac | R |
|---|---|---|
| Thrust chamber #313 | 312.8 sec | 0.05346 |
| Thrust chamber #320 | 311.4 sec | 0.05322 |
| | Difference: | 0.00024 or 0.45% |

It will be seen that the percentage difference in R is eleven times the quoted $1\sigma$ deviation in R.

The $1\sigma$ measurement uncertainty in engine $I_{sp}$ vacuum data obtained from the tests in J-3 cell at AEDC is given as $\pm$ 0.19% by ARO (Reference 6) and $\pm$ 0.22% by Aerojet (Reference 5) or approximately 0.6 seconds in $I_{sp}$.

Because a large number of data points (approximately 100) were used in the derivation of R, Aerojet do not use these values of random measurement uncertainty, but allocate a best guess possible error in the AEDC data of 0.2 seconds in $I_{sp}$ vacuum.

The following expression for calculating the $3\sigma$ $I_{sp}$ vacuum uncertainty was given by Aerojet at their MSC presentation:

$$\text{uncertainty} = \pm 3 \sqrt{1/3\ (R \times B)^2 + (C^*_{inj} \times R \times A)^2 + C^2} \quad \text{sec}$$

where: R = 0.05338 $sec^2/ft$

B = $1\sigma$ $C^*_{inj}$ measurement uncertainty = 9.5 ft/sec

1/3 = effect of taking $C^*_{inj}$ from the mean of three acceptable tests.

$C^*_{inj}$ = average value of $C^*$ steel chamber = 5876 ft/sec

A = $1\sigma$ deviation of R = 0.0004

C = AEDC uncertainty = 0.[illegible] seconds

Using the above values, the uncertainty is $\pm$ 1.1 seconds.

The same numerical value of $I_{sp\ vacuum}$ uncertainty has been obtained by other methods (Reference 8). A $3\sigma$ $I_{sp\ vacuum}$ uncertainty of $\pm 1.1$ seconds is lower than the values given by both TRW and Bell: the Rocketdyne uncertainty value, being expressed as a 95% confidence limit, is not directly applicable.

## 4.5 Operation

No accurate data on $I_{sp}$ variation with engine run time is available from the AEDC tests, because the measured thrust became suspect after some 40 seconds of engine running, i.e., a change in thrust was indicated, with no corresponding changes in flow rates or chamber pressure. The apparent thrust shift is believed to be due to thermal effects. Despite the fact that there is no accurate measured data available, it is probable that the engine shows only slight variations in $I_{sp\ vacuum}$ with run time. The change in throat area on the ablative chambers used in the qualification test program, after having been fired for over 700 seconds, varied between 0.5% increase and 1.2% decrease. At one time, Aerojet allowed for a constant 0.2 second decrease in specific impulse over the engine's full mission duty cycle.

## 4.6 References


1. Notes taken at Aerojet presentation at MSC, 13 February 1968 and presentation material (3865-616).
2. Presentation to Dr. Rees , at Sacramento, 16 January 1968. (3865-608).
3. Aerojet specification #AGC - 46847A, 12 September 1967, Acceptance Test, Injector Assembly AJ10-137.
4. Aerojet specification #AGC - 46846B, 17 November 1967, AJ10-137 (Apollo) Acceptance Test Procedure for Engine Assembly - Service Propulsion System.
5. Aerojet report 3865-458, March 1967, "Apollo Service Module Engine Block II Performance Analysis Report".
6. AEDC report AEDC-TR-67-63, May 1967, "Qualification Tests on the Apollo Block II Service Module Engine (A310-137).
7. TRW report 05952-H220-R000, June 1967, "Final Performance Characterization for the SPS Block II Engine".
8. SPS Injector Performance. 1-11-68. By R.K. McSheehy.

## 5. Bell LM Ascent Engine

### 5.1 Introduction

The Bell LM Ascent Engine is not gimballed, is of fixed thrust and can be restarted. Calibration trim of thrust and mixture ratio is by orifices in the oxidizer and fuel feed lines. The ablative monolithic thrust chamber and nozzle are bonded together; there is no removable nozzle extension. The nozzle has an expansion area ratio of 45.6:1 and the chamber has an injector face area to throat area ratio of approximately 2.7:1. Nozzle exit area is approximately 750 $in^2$ and throat area 16.45 $in^2$.

#### 5.1.1 Nominal Operating Conditions

| | |
|---|---|
| Ambient pressure | 0 psia |
| Oxidizer interface pressure | 170 psia |
| Oxidizer temperature | 70° F |
| Oxidizer specific gravity | 1.446 60/60 (at 60° F, with respect to water at 60° F) |
| Fuel interface pressure | 170 psia |
| Fuel temperature | 70° F |
| Fuel specific gravity | 0.903 60/60 |
| Engine mixture ratio | 1.6:1 |

#### 5.1.2 Nominal Performance (Under the Above Conditions)

| | |
|---|---|
| Thrust | 3,500 lbs. |
| Oxidizer flow rate | 7.034 lbs/sec |
| Fuel flow rate | 4.396 lbs/sec |
| Total flow rate | 11.435 lbs/sec |
| Specific impulse | 306.1 sec |

### 5.2 Engine Acceptance Test Procedures

There are three stages in an engine's acceptance test firing sequence.

#### 5.2.1 Injector and Valve Assembly Calibration

Here, the injector and valve assembly are run in a water cooled slave thrust chamber, under local ambient pressure conditions. Mixture ratio and total propellant flow rates are adjusted, if necessary, and then six 15 second firings are carried out under the following conditions:

| | | |
|---|---|---|
| Oxidizer feed pressure | Pre-run | 205 ± 8 psia |
| | Steady State | 170 ± 5 psia |

| | | |
|---|---|---|
| Oxidizer temperature | | 70° F $\pm$ 10° F |
| Fuel feed pressure | Pre-run | 205 $\pm$ 8 psia |
| | Steady State | 170 $\pm$ 5 psia |
| Fuel temperature | | 70° F $\pm$ 10° F |

It will be seen that these tests have to be carried out with feed pressures within 5 psia, and propellant temperatures within 10° F of their nominal conditions.

Under these test conditions, the following requirements have also to be met:

Oxidizer assembly $\Delta p$, corrected to nominal temperature and flow rates: 49.0 $\pm$ 2.6 psi.

Fuel assembly $\Delta p$, corrected to nominal temperature and flow rates: Equal to oxidizer assembly $\Delta p$ $\pm$ 0.9 psi.

(Assembly $\Delta p$ is the pressure drop between engine feed pressure and thrust chamber pressure).

Mixture ratio, corrected to nominal temperatures and feed pressures: 1.600:1 $\pm$ 0.019 ($\pm$ 1.19%) with a maximum allowable variation on the six tests of 0.008 (1/2%).

Not Mandatory
(C* should be greater than 5380 ft/sec
(
(Total flow rate (measured) should be 11.430 $\pm$
(0.343 lbs/sec
(
(Mixture ratio (measured) should be 1.6 $\pm$ 0.06

5.2.2 Injector Ablative Compatibility

No performance data is obtained from this test. A full duration (460 sec) firing is carried out with the injector fitted to a slave chamber with an ablative liner. If, after this test, there are gouges in the ablative liner of greater than 3/8 in (within 4 inches of the injector face) or 1/8 in (between 4 and 8 inches from the injector face), the injector is rejected.

5.2.3 Engine Acceptance Firings

Two 15 second firings are carried out with the engine under simulated altitude conditions.

The required steady state feed system conditions are the same as for the injector and valve assembly calibration tests. The maximum allowable difference between the corrected altitude thrust on the two tests is 19 1/4 lb (0.55%) and between corrected $I_{sp}$ is 0.7 sec (0.23%). Engine specific impulse is measured directly from thrust and flow rates and is not calculated from measured C*.

## 5.3 Instrumentation

### 5.3.1 Thrust

Thrust is measured by 7 dual bridge bonded strain gauge load cells. The engine is mounted horizontally with four axial, two vertical and one lateral load cell. Thrust measuring system calibration is carried out with the engine installed, the feed lines pressurized with their working fluids and with the test capsule at altitude pressure, by a master load cell before and after a firing: during a firing, the master load cell is disconnected and its output monitored. Interaction effects are determined by using a special calibration loading program. The master load cell is laboratory calibrated with a N.B.S. certified proving ring. Resultant thrust and thrust alignment are obtained from the load cell data on a firing: thrust chamber weight loss is taken into account in the thrust alignment calculations, a constant rate of loss being assumed.

### 5.3.2 Flow Rate

Flow rates are measured by two turbine flow meters in series. Calibration is carried out in a calibration flow rig, using propellants conditioned to nominal temperature. The two flow meters are fitted in an assembly which has 40 inches of straight run pipe upstream of the first flow meter and a Rosemont temperature probe fitted between the flow meters. After calibration, the entire assembly is fitted in the engine test stand: the system is not broken into after calibration. Calibration is carried out at nominal flow rate and with the propellant at nominal working pressure. When flow conditions are steady, the flow is diverted into an unvented weigh tank for some 90 seconds. The diverter valve cavitates, hence change in pressure in the unvented weigh tank does not affect flow rate. The weigh tank load cells are dead weight calibrated. These weights are N.B.S. traceable. The flow meter frequency counters are also N.B.S. traceable. Propellant specific gravity is determined at ambient pressure: the same hydrometers are used to determine the specific gravity of the propellant used on engine firings, which is also measured at ambient pressure. On a firing, the flow rates, as measured by the two flow meters in series, must not differ by more than 0.35%. A constant calibration factor for each flow meter is used.

5.3.3 Pressure

The low range ($\pm$ 10 psia) hypergolic $\Delta p$ transducers are in place calibrated. The maximum allowable variation between the two hypergolic $\Delta p$ measurements on a test is only 0.07 psia (0.7%). Cell pressure is measured by four 0-2 psia pressure transducers: two of these are suspended inside the cell, with no tapping lines. Their temperature compensation is good for up to 120° F. Under normal operating conditions, the cell pressure is approximately 0.1 psia or only 10% of the cell pressure transducer full range. An error of 1% of the transducers full range in cell pressure measurement results in an error of $\frac{2 \times 750}{100}$ = 15 lbs in altitude thrust or 1.3 seconds in altitude $I_{sp}$.

5.3.4 Instrumentation Comments

At the Bell presentation at MSC on February 14, 1968, the two main doubts concerning their instrumentation were the range of the cell pressure transducers and the possible flow meter calibration errors, due to flow diverter valve characteristics.

5.4 Analysis

The flows as measured by the two series flow meters are calculated and averaged. Vacuum thrust is calculated by taking into account the load cell interaction effects to obtain a thrust resultant, and then adding the product of the nozzle exit area and cell pressure. Correction from test to nominal conditions is by linear gains, which have been empirically obtained from a large number (approximately 50 tests, using the same injector) of specifically designed performance tests, where $I_{sp}$ ~ chamber pressure ~ mixture ratio effects were determined over a small range. The uncertainty on corrected thrust was given as $\pm$ 0.703% or 24-1/2 lbs.

Measured mixture ratio is corrected directly from the measured hydraulic resistances of the two feed systems. Corrected mixture ratio is very sensitive to differences in inlet pressures (hypergolic $\Delta p$): a difference of 1 psi results in approximately a 1% shift in mixture ratio. Note that change in throat area, due to erosion, will not affect mixture ratio, only total flow rate. The uncertainty on corrected mixture ratio was given as $\pm$0.51%.

The corrected specific impulse quoted is, in fact, the same as the measured specific impulse, simply corrected to 0 psia ambient pressure. This partly accounts for the uncertainty in corrected $I_{sp}$ ($\pm$ 0.455% or 1.4 sec) being less than the uncertainty quoted for corrected thrust. Bell uses the following equation to calculate their total uncertainty:

$$\text{Total uncertainty} = \left[\text{Bias}_1^2 + \text{Bias}_2^2\right]^{1/2} + 3\left[\sigma_3^2 + \sigma_4^2\right]^{1/2}$$

where: $\text{Bias}_1$ = traceability to standard

$\text{Bias}_2$ = ~~zero~~ shift and system hysteresis (95% confidence)

$\sigma_3$ = non repeatability of calibration equipment (95% confidence)

$\sigma_4$ = non repeatability of instrument (95% confidence)

At 170 psia feed pressures and 70° F propellant temperatures, the following gain effects are used for calculating corrected thrust:

| A change of | in | will result in the following change in corrected thrust |
|---|---|---|
| +0.1 psia | Ambient pressure | -75 lb. |
| +1.0 psia | Fuel feed pressure | +6.15 lb. |
| +1.0 psia | Oxidizer feed pressure | +9.83 lb. |
| +1° F | Fuel temperature | -0.17 lb. |
| +1° F | Oxidizer temperature | -0.42 lb. |

The philosophy of using empirical data, obtained from specially planned performance tests, to correct data which are obtained from tests carried out under conditions which are very close to nominal, is a very good one. The use of linear gain factors, for corrections over a small range, is not only simpler, but may be more accurate than more complicated empirical methods. The fact that the engine acceptance tests are carried out under conditions which are very close to nominal presumably allows Bell to assume that altitude $I_{sp}$ and corrected $I_{sp}$ are the same.

5.5 Operation

After 15 seconds, conditions are reasonably stable, except for capsule pressure, which is still slowly decaying. Inspection of LM-3 engine acceptance data show very good run to run repeatability with both the injector and engine tests carried out under conditions which are very close to nominal. On the six injector and two engine acceptance tests carried out on the LM-3 engine, the maximum variation of inlet pressure from nominal was 3.6 psi and of inlet temperature 6.6° F.

The subject of performance degradation with run time was not discussed at MSC. As an example of this effect, the following data, extracted from a Bell program review (test 1BN-423 on 12-30-67) shows that over a 460 second firing, the $I_{sp}$ decreased by 3.1 seconds, the thrust increased by 60 lb and the effective throat area increased by 7.5%. These results, however, may not be typical.

## 5.6 References


1. Notes taken at Bell presentation at MSC, 14 February 1968.
2. Bell report #8258-927019, revision B, dated 15 January 1968, "Determination of Bell Model 8258 Altitude Performance".
3. Draft of LM updated uncertainty model report given by Bell at MSC presentation, 14 February 1968.
4. End item narrative report for engine assembly - LM ascent engine #P2C/102C, 15 February 1968, (this is the replacement engine for LM-3).
5. Bell LM ascent engine program reviews.

## 6. LM Ascent Engine (Rocketdyne Back-up Injector)

### 6.1 Introduction

Rocketdyne are presently developing a back-up injector for the LM Ascent Engine. Apart from the injector, the rest of the engine (thrust chamber, valves, vehicle interface plumbing, etc.) uses the same components as the Bell LM Ascent Engine. Rocketdyne received their go-ahead for the program from NASA on 3 August 1967. Modification of their test facilities at Santa Susana and Nevada were begun at this date. The initial tests in the Nevada altitude facility indicated that engine thrust measurements were anomalously high: the thrust measuring system was modified, and some of the data recording equipment was replaced. The first full duration firing in the facility, after these modifications had been completed, was on 10 February 1968.

Nominal Operating Conditions }
Nominal Performance } Same as for Bell LM Ascent Engine

### 6.2 Acceptance Test Procedures

No acceptance test procedures are available, but the performance testing follows a similar pattern to that used by Bell. Injector performance and erosion characteristics are obtained from testing in the BRAVO 3A test stand at Santa Susana and vacuum specific impulse is obtained from firing the complete engine in the NFL B-4 altitude test stand at the Nevada facility.

On the BRAVO 3A test stand, the engine is fired vertically and is tested at local ambient pressure. The injector is fitted to a chamber with an ablative liner and water cooled throat (performance data) or to a barrel thrust chamber with an uncooled ablative liner and throat (injector compatibility data).

In the NFL B-4 test stand, the engine and its thrust measuring are mounted horizontally in a capsule. Altitude pressure is obtained by a steam ejector and maintained during engine firing by an exhaust driven diffuser.

### 6.3 Instrumentation

#### 6.3.1 Thrust

The engine is mounted by six load cells to an isolation cradle, which is then suspended inside the test capsule. The load cells are dual bridge strain gage type. Three axial load cells measure main thrust, two having a range of 0-1400 pounds and the third one a range of 0-1700 pounds. The required system precision of these main load cells is ± 1/4%. Misalignment thrust is obtained from one horizontal load cell of ± 30 pounds range and two vertical load cells of ± 60 pounds range. The required system precision for the misalignment load cells is ±1%.

In-place calibration of the main thrust load cells is carried out at ambient pressure with a single standard load cell. A hydraulic ram is used to transmit the calibrating force through the standard load cell, to the main thrust load cells. The standard load cell is laboratory calibrated with a NBS certified proving ring.

Interaction effects were found to be very small: the maximum amount was 1.6 lb on one of the vertical load cells, when 80% load was applied to the main load cells and 50 lbs. to the horizontal load cell. There was no interaction of the horizontal and vertical load cells on the main load cells.

### 6.3.2 Flow Rates

At both the NFL and BRAVO test stands, two 1" turbine flow meters in series are fitted in each propellant feed line. At BRAVO, they are 10" apart, with their temperature probe 4 to 5 inches downstream of the second flow meter: at NFL, they are 12" apart, with their temperature probe between the flow meters.

The basic method used to calibrate the flow meters is similar to that used by Rocketdyne on their test stands using cryogenic propellants. The run tanks are fitted with a float, which, as the liquid level in the tanks change, can move and operate float switches which are fitted inside the tanks. By calibrating the run tanks with water, the volume of fluid between the different float switches is known and this is used to calibrate the flow meters in place on long duration runs. During tank calibration, water is run out of the tank into a standard volume measure: water level in the measure is read on a sight glass. The calibration procedure is repeated four times on each run tank, allowing the calibration precision to be obtained. The standard volume measure is laboratory calibrated against NBS certified weights. Tank calibration is carried out with water at ambient temperature and pressure: tank pressure, propellant temperature and water/propellant buoyancy corrections have to be calculated when this tank calibration data is used to calibrate the propellant flow meters.

The required system precision for the flow measurements at BRAVO and NFL is ± 1/4%.

### 6.3.3 Pressure

Cell pressure at NFL is measured by four Pace transducers of 0 to 0.2 psia range. The transducers are mounted on the outside of the pressure capsule and sense, through their tapping lines, the cell pressure at the following locations:

- Engine injector
- Engine throat
- Halfway between nozzle throat and exit
- Nozzle exit

These transducers have a required precision of ±1% and are calibrated using a manometer.

At the time of the Rocketdyne presentation (15-16 February), hypergolic Δp transducers were not used, though it was understood that they would shortly be fitted to the test facilities.

Feed system and injector pressure drops are directly measured with Δp transducers and also by taking the differences between the measured interface, injector and chamber pressures.

The method used to obtain the best estimate of pressure drop from these two sets of data is described in Section 6.4.1. Pressure transducers are laboratory calibrated with a dead weight calibrator, the working weights being calibrated against NBS certified standard weights.

### 6.3.4 Temperature

The Rosemont resistance temperature bulbs, used for measuring propellant temperature, have a working range of 0 to 150° F, but are calibrated at cryogenic temperatures, liquid nitrogen and liquid helium being used as temperature standards. A 4th order polynomial calibration equation is developed to extrapolate the calibration data to the working temperature range: this is virtually a linear fit and shows good agreement with the manufacturers calibration data points, which are made using liquid nitrogen, water freezing and water boiling as temperature standards for use in the cryogenic range and water freezing and boiling for use in the 50 to 500° F range.

## 6.4 Analysis

The Rocketdyne data reduction program is similar to the ones used on their launch vehicle engines. Test data is converted into engineering unit data and various derived parameters and efficiencies are calculated. From measured flow rates, throat area, thrust and thrust chamber pressure, engine mixture ratio, specific impulse, C* and thrust coefficient are calculated. Theoretical values of C* and thrust coefficient are calculated and hence C* and thrust coefficient efficiencies are obtained.

The data is presented in the following five different forms:

6.4.1 Site Data

Parameters calculated directly from test data, at actual test conditions. No corrections are made. Flow rates are taken as the average of the readings of the two flow meters and thrust as the average of the readings of the two bridges in each load cell. Best estimates of feed system pressure drops are made by combining the data from the differential pressure transducers and the interface and chamber pressure transducers. The weighting factors used in combining these measurements are calculated from the uncertainty of the measurements. From the load cell data, the engine's thrust vector is determined, together with the required adjustments to the bushings in the three engine mounts needed to correct for misalignment.

6.4.2 Site Vacuum Data

Same as site data, but with thrust corrected to 0 psia ambient pressure. Flow rates and pressures are not changed and the only calculated parameters affected are thrust coefficient and specific impulse.

6.4.3 Site Vacuum (Standard Temperature)

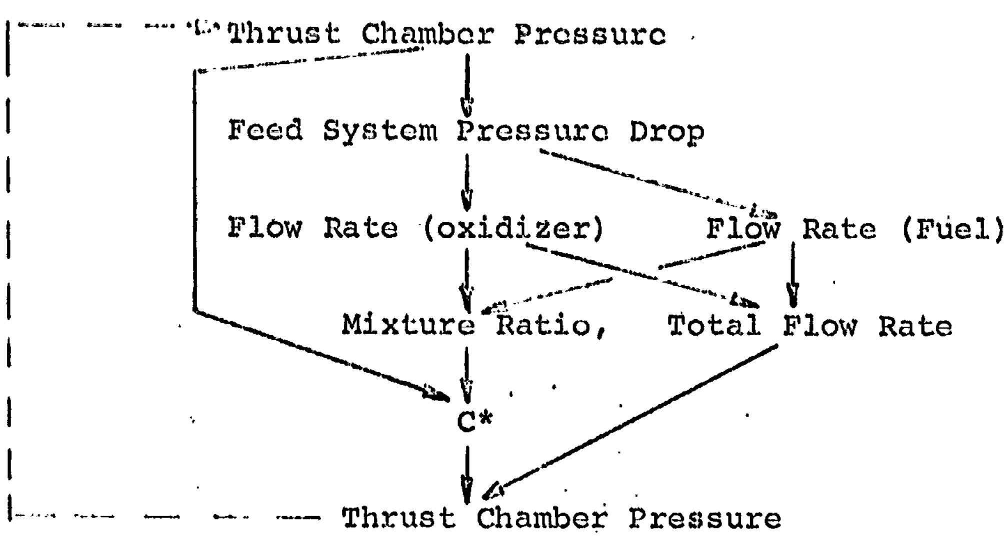
Site vacuum data, with the necessary corrections applied to bring the propellant temperatures and densities from their site to their nominal values. The resulting changes in density and viscosity are calculated from polynomial curve fit equations. As an example of the magnitude of these corrections upon engine performance, the following information is extracted from the analysis of an engine performance test at Reno (test 374-31 on 1-20-68: 15 second data slice).

| | Site Vacuum Performance | Site Vacuum (Standard Temperature) | Delta |
|---|---|---|---|
| Oxidizer temperature °F | 76.78° F | 70° F | +6.78° F |
| Oxidizer density $lb/ft^3$ | 89.74 $lb/ft^3$ | 90.21 $lb/ft^3$ | -0.47 $lb/ft^3$ |
| Fuel temperature °F | 76.02° F | 70° F | +6.02° F |
| Fuel density $lb/ft^3$ | 56.04 $lb/ft^3$ | 56.39 $lb/ft^3$ | -0.35 $lb/ft^3$ |
| Thrust lb. | 3263 lb. | 3258 lb. | +5 lb. |
| Mixture ratio | 1.505:1 | 1.495:1 | +0.01 |
| Specific impulse sec. | 306.8 sec. | 306.7 sec. | +0.1 sec |

6.4.4 Standard Performance

This is site vacuum (standard temperature) data, which has been corrected to bring the interface pressures to their nominal value of 170 psia.

The calculations for feed system corrections (in propellant temperature, density and interface pressure) are basically made by using site values of feed system hydraulic pressure losses and iterating around the following thrust chamber pressure loop:




The C* — Mixture Ratio — thrust chamber pressure information used by the program is based on empirical data obtained from injector tests at BRAVO. Test 374-31 was run with interface pressures which varied considerably from the nominal level of 170 psia and therefore some large corrections were made when the site vacuum (standard temperature) data was corrected to standard performance conditions.

| | Site Vacuum (Standard Temperature) | Standard Performance | Delta |
|---|---|---|---|
| Oxidizer interface pressure psia | 146.18 psia | 170 psia | -23.82 psi |
| Fuel interface pressure psia | 151.42 psia | 170 psia | -18.58 psi |
| Thrust lb. | 3258.0 lb. | 3649.0 lb. | -391.0 lb. |
| Mixture ratio | 1.495:1 | 1.610:1 | -0.115 |
| Specific impulse sec. | 306.7 sec | 307.2 sec | -0.5 sec |

### 6.4.5 Rated Performance

The engine's standard performance is adjusted to rated performance by mathematically changing the hydraulic resistance of the feed systems and calculating the necessary changes in trim orifice size. Rated performance of the engine, at standard conditions, is:

| | |
|---|---|
| Thrust | 3500 lb. |
| Mixture ratio | 1.6 |

Test 374-31 data gave the following orifice sizes required to bring the engine's performance, under standard conditions, to rated.

| | |
|---|---|
| Fuel orifice size: | From 0.901 inches to 0.652 inches |
| Oxidizer orifice size: | From 0.977 inches to 0.701 inches |

### 6.4.6 Uncertainty Analysis

Measurement uncertainties are all calculated on the basis of 95% confidence limits in instrumentation system precision. The NASA requested method of expressing error as the sum of bias error and three standard deviations of random error (to 95% confidence) is not used by Rocketdyne: it would, however, result in larger measurement uncertainties. Because of the different methods in specifying instrumentation uncertainties, it is difficult to make direct comparisons between the systems used by Rocketdyne and Bell.

The effects of the measurement uncertainties upon engine thrust, mixture ratio and specific impulse are calculated by perturbing each measurement parameter in turn, by an amount equal to its quoted 95% confidence measurement uncertainty, and obtaining, from the engine data reduction program, the effect upon engine thrust, mixture ratio and specific impulse under standard performance conditions. This process is repeated for each measurement parameter. The root sum square (RSS) of all the changes in thrust, mixture ratio and specific impulse are obtained and these values used to express engine performance uncertainties to 95% confidence. The error analysis of Test 374-31 data gave the following values of engine performance uncertainty:

| | | |
|---|---|---|
| Thrust | $\pm$ 8.83 lb. | ($\pm$ 0.33%) |
| Mixture ratio | $\mp$ 0.0096 | ($\mp$ 0.60%) |
| Specific impulse | $\pm$ 0.622 sec | ($\pm$ 0.20%) |

These figures are given as the confidence in the engine's performance under standard performance conditions. No estimate of the errors in the correction of the data from site vacuum to standard conditions are made: for test 374-31, these corrections changed site vacuum thrust by 11.8% and site mixture ratio by 7.3%.

## 6.5 Operation

Apart from the results of engine test 374-31 and injector test 636-206 in BRAVO 3A on 2-6-68, no test data are available.

## 6.6 References

1. Notes taken at Rocketdyne presentation at MSC, 15 - 16 February 1968, including copies of presentation material.
2. Draft description of REA steady-state data reduction program, including data reduction of injector test 636-206 (2-6-68) and engine test 374-31 (1-20-68).