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MODEL NO. CONTRACT NO. NASw-1650

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ABSTRACT

This report, in conjunction with D2-117060-1 "Apollo Spacecraft Engine Specific Impulse" May 6, 1968, describes the methods used to determine the vacuum specific impulse, discusses the various performance analysis procedures and summarizes the data obtained on production configuration Apollo Spacecraft engines. The following primary propulsion system engines, used in the Apollo Service and Lunar Modules, are included in this report:

LM Ascent Engines - Rocketdyne and Bell
LM Descent Engine - TRW
SM Engine - Aerojet

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INTRODUCTION

This report, in conjunction with D2-117060-1 "Apollo Spacecraft Engine Specific Impulse" May 6, 1968, is in partial fulfillment of NASA Technical Directive NASw-1650 Serial #28 "Apollo Rocket Engine Altitude Specific Impulse Ratings". The earlier report described the methods used to determine the vacuum specific impulse of the primary propulsion engines used in the Apollo spacecraft, i.e.

The LM Descent Engine (TRW)

The SM Engine (Aerojet)

The LM Ascent Engine (Bell)

The LM Ascent Engine Injector Back-up Program (Rocketdyne)

The same engines are discussed in this report, though the two LM ascent engines are covered in the same section. Where necessary, the information given in the earlier report has been amplified or updated. As requested by APO, quantitative information on the specific impulse of individual production configuration spacecraft engines is included in this document. Also, no recommendations are made or conclusions drawn.

Section 2 deals with the Bell and Rocketdyne LM Ascent engines and uses information available in June 1968. This section was reviewed in draft form by MSC (EP) and Rocketdyne. The TRW LM Descent engine is discussed in Section 3 and is also dated June 1968. The draft of this section was reviewed by MSC and TRW. Due to delay in obtaining the required data, Section 4, which covers the Aerojet SPS engine, is dated September 1968, and has not been reviewed by MSC or Aerojet.

1.0 "Minimum Required" Vacuum Specific Impulse

There are at least three different definitions of the "minimum required" vacuum specific impulse of the spacecraft main propulsion engines, namely:

A "minimum required" value of specific impulse obtained on engine acceptance test.

A "minimum required" value of average specific impulse obtained during a mission duty cycle.

A "minimum required" value of specific impulse obtained at any time during a mission duty cycle (M.D.C.).

The "minimum allowable" specific impulse on acceptance test will be the "minimum required" value plus the uncertainty. The uncertainties in each of these values of specific impulse will increase in the same order as they have been listed above. The most obvious reason for this is that whereas the specific impulse on acceptance test is directly measured for every production engine, the average and minimum values of specific impulse during M.D.C. have to be predicted on the basis of information gained on other (i.e. qualification and design verification test) engines.

1.1 Acceptance Test Specific Impulse

Except in the case of the Aerojet SM engine, specific impulse on acceptance tests is determined for each production engine from direct measurements of thrust, flow rates and test cell ambient pressure. These tests are carried out with engine interface temperatures and pressures close to the nominal levels. An analysis of the instrumentation system calibration data allows an estimate of the specific impulse measurement uncertainty to be calculated. This uncertainty is usually quoted as the engines' acceptance test specific impulse uncertainty, errors introduced in correcting the test data to standard conditions being ignored. Repeated tests allow the run-to-run variability to be obtained, which can be used as a further check on the instrumentation variability.

1.2 Mean Specific Impulse During M.D.C.

An engine's mean specific impulse during M.D.C will differ from its acceptance test performance because:

1.2.1 Its throat area will change during M.D.C. at a rate dependent upon the engine operating conditions and also upon its individual injector-to-chamber compatibility characteristics. Only in the case of the TRW engine is a quantitative measure of injector compatibility obtained during acceptance tests. This change in throat area will result in a change in nozzle area ratio and hence in engine specific impulse. Typically, the throat area will first decrease and will then increase with run time..

1.2.2 There will be changes in nozzle internal surface finish. Delamination of the nozzle material may occur. Deposition of glass may occur in the divergent position of the nozzle. There will be some reduction in nozzle thrust coefficient efficiency due to these causes.

1.2.3 In flight, the engine's propellants will be saturated with helium to some degree. All acceptance tests are carried out with unsaturated propellants and in the case of the TRW engine, the propellant tanks at the Capistrano test site are pressurized with nitrogen. The true effect upon an engine's specific impulse due to using helium saturated propellants is not presently known for any of the spacecraft primary propulsion engines. This effect, however, is assumed to be small and is ignored in M.D.C. specific impulse uncertainty estimates.

1.2.4 The propellants at the engine interfaces will not be at nominal temperatures and pressures. During qualification and D.V.T. test, sufficient data should have been obtained to enable these effects to be characterized and to make an estimate of the uncertainty of the resulting model.

1.2.5 In the case of the TRW LM descent engine, there is a further uncertainty caused by the fact that the engine is throttled. Throttling the engine changes both its specific impulse and its rate of change of throat area, which again affects specific impulse

1.3 Minimum Specific Impulse During M.D.C.

Except in the case of the TRW engine, where minimum specific impulse is dependent upon throttle setting, the minimum specific impulse of the spacecraft engines will occur at the end of their mission duty cycle. All of the uncertainties in 1.2 will also affect the uncertainty of the estimate of minimum specific impulse, but probably to a greater extent (more uncertainty as to propellant conditions, possibility of pressurizing gas entrainment, engine in high rate of erosion regime, etc.).

2.0 LM Ascent Engines - Rocketdyne and Bell (June 1968)

2.1 Introduction

A description of the methods used to determine the vacuum specific impulse of the Bell LM ascent engine is given in Reference 1. At the time that Reference 1 was written, no acceptance test procedures for the Rocketdyne engine were available, so these are described in some detail in Section 2.2, together with a comparison between the Bell and Rocketdyne acceptance test requirements.

Sections 2.3 and 2.4 compare the performance analyses methods and the specific impulse of the Bell and Rocketdyne engines. Much of the information contained in these sections was generated from work carried out in support of the ascent engine program review team (Reference 11, 12) and many of the conclusions listed in Section 2.5 are also given in the LM ascent engine performance evaluation report to the review team (Reference 11). The writer would like to express his appreciation to the performance member of the program review team (C. Verschoore - MSFC - R-Test-C) for the opportunity to participate in the review team activity.

2.2 Test Procedures

2.2.1 Bell Test Procedures

A general description of the Bell test procedures is given in Reference 1: for more details, see References 6 and 7. To summarize, the injector and valve assembly are first calibrated and then acceptance test fired in a water cooled steel chamber with an ablative liner. A single compatibility test of 460 second duration is then carried out. Finally, the injector and valve assembly are then assembled with their flight chamber and the complete engine is acceptance tested under simulated altitude conditions.

2.2.2 Rocketdyne Test Procedures

For full details of these procedures, see References 2 through 5.

2.2.2.1 Injector and Valve Assembly Calibration and Acceptance Test

These tests are carried out in the BRAVO 3A test stand at Santa Susana. The injector is fitted to a chamber with an ablative liner and water cooled throat. The assembly is mounted in the test stand to fire vertically downward and tests are carried out at local ambient temperature and pressure. The propellant tanks are pressurized with helium but no attempt is made to saturate the propellants on production injector acceptance tests. However, the propellants are temperature conditioned.

A minimum of four satisfactory valve-injector assembly acceptance tests of 15 second duration must be carried out. During these tests, no changes of injector, propellant ducts, orifices or valves are allowed: if there are any hardware changes, then four satisfactory tests with the new hardware are required. Throat diameter is measured prior to the first test and after every test series.

The tests must be carried out under the following conditions:

Propellant interface pressure	170 ± 5 psia
Propellant interface temperature	70 ± 10° F
Maximum difference between propellant temperatures	10° F
Environmental temperature	Ambient
Environmental pressure	Ambient

There is no specified maximum difference between the propellant interface pressures, though hypergolic Δp measurements are taken. Performance data are averaged over an interval of 1 second, with the mid point of this data interval being between 1.0 and 0.6 seconds prior to cutoff. For acceptance tests to be satisfactory, the following performance requirements must be met:

Mixture Ratio (corrected to standard conditions)

1.6:1 ± 0.016 (± 1%)
Mixture Ratio uncertainty (95% confidence). Less than ± 0.008 (± 1/2%)

Characteristic Velocity (corrected to standard conditions) minus the characteristic velocity uncertainty (95% confidence) must be greater than 5629 ft./sec.

Chamber Pressure (corrected to standard conditions)
122.1 ± 2.3 psia (± 2%)

Standard conditions are:

Environmental pressure	0 psia
Propellant interface pressures	170 psia

Oxidizer density	90.21 lb./ft.^{3}
Fuel density	56.39 lb./ft.^{3}

The uncertainty calculations are made from instrumentation calibration repeatability data: there are no specified limits on either the variability or the range of the corrected tests data.

2.2.2.2 Injector Compatibility Test

This test is of 460 $\pm$ 5 second duration and must be carried out under the same conditions as are required for the other injector and valve assembly acceptance tests. The compatibility test can be carried out at any time after two satisfactory injector acceptance tests have been completed.

The following performance conditions must be satisfied during the test:

Mixture Ratio (site)	1.6:1 $\pm$ 0.048 ($\pm$ 3%)
Chamber Pressure (site)	122.2 $\pm$ 2.4 psia ($\pm$ 2%)

At the completion of the compatibility test, there must be no gouges in the liner deeper than 0.25 inches.

2.2.2.3 Engine Acceptance Tests

These are carried out in B-4 test stand at the Nevada test site. The engine and its thrust measuring rig are mounted horizontally in a capsule. Altitude pressure is obtained by a steam ejector and maintained during engine firing by an exhaust driven diffuser. Propellant tanks are pressurized with helium. There is provision for both temperature conditioning and helium saturating the propellants.

A minimum of two satisfactory engine acceptance tests of 15 second duration have to be carried out. No hardware changes, except to the mounting pad bushings, are allowed. On acceptance tests, the propellants are not helium saturated, but on M.D.C. duration tests and on most of the D.V.T. performance tests fully saturated propellants are used. Throat and exit area measurements are taken prior to the first test and after the last in each test series.

The test conditions are the same as for the injector tests, except that the capsule pressure altitude must be greater than 90,000 ft. and the environmental temperature must be 70 $\pm$ 30°F. There is no specified maximum difference between the propellant interface pressures and when the writer inspected the test facility, hypergolic Δp pressure transducers were not installed.

For the acceptance tests to be satisfactory, the following performance requirements must be met:

Thrust (corrected to standard conditions)
3500 lb. ± 52 lb. (± 1 1/2%)
Thrust uncertainty (95% confidence) less than ± 35 lb. (± 1%)

Chamber Pressure (corrected to standard conditions)
120 psia ± 8 psia (± 6 1/2%)

Mixture Ratio (corrected to standard conditions) ± mixture ratio uncertainty (95% confidence) must be 1.6:1 ± 0.032 (± 2%).

Specific Impulse (corrected to standard conditions) minus the specific impulse uncertainty (95% confidence) must be greater than 306.3 seconds. This value has now been increased to 307.0 seconds (Reference 17).

Thrust Alignment - Displacement of the thrust vector from the engine reference line must be less than 0.300 inches. Angular deviation of the thrust vector from the engine reference line must be less than 30 minutes of arc.

2.2.3 Differences Between Bell and Rocketdyne Acceptance Test Requirements

Injector Acceptance Tests

	Bell	Rocketdyne
Number of satisfactory tests	6	4
Specified maximum difference between propellant interface pressures	Yes (0.9 psi)	No
Run to run limits on mixture ratio	Yes (.49%)	No
Minimum C* requirement	Not mandatory	Yes (5629 ft./sec. + C* uncertainty to 95% confidence)

	Bell	Rocketdyne
Chamber pressure level requirements	No	Yes (122.1 ± 2.3 psia)
Maximum gouge depth allowed on compatibility tests	3/8 " down to 4" from injector and 1/8" from 4" to 8"	0.25 "

Engine Acceptance Tests

	Bell	Rocketdyne
Run to run limits on thrust	Yes (.56% for 2 tests; .73% for 4 tests)	No
Run to run limits on I_{sp}	Yes (.3% for 2 tests; .39% for 4 tests)	No
Minimum allowable I_{sp} (corrected)	308.4 sec.	306.3 sec. ** + I_{sp} uncertainty to 95% confidence
Engine mixture ratio limits	No	Yes (1.6:1 ± 2% including M.R. uncertainty to 95% confidence)
Thrust vector requirements	Within 0.069" radius at both exit and forward planes	Specification limits (less than 0.300" and 30 minutes) *
Run to run limits on thrust vector	Yes. Location of thrust vector at exit and forward planes must not differ on two tests by more than 0.051"	No

* The Rocketdyne thrust vector requirements during acceptance test are being revised to take into account the characteristic shift which occurs during M.D.C.

** Now 307.0 sec. (Reference 17)

2.2.4 Comparison Between Bell and Rocketdyne Test Operations

Probably the most significant difference between the Bell and Rocketdyne test operations which affect I_{sp} determination is in the data reduction methods used on the engine tests.

The Bell test facility has an on site data processing center. There is a Beckman direct read out display in the control room, allowing the test engineers to set up test conditions using the same information as is fed into the data reduction program. Data reduction, into engineering unit data, is carried out immediately after a firing. Pre-set standard deviation limits are incorporated in this data reduction program and measurements which exceed these limits are flagged out C or F, depending on if they exceed the Coarse or Fine limits. After the performance measurements have been checked for precision, the performance reduction program is then run and the results are available within one to two hours after a test. These results are the final ones used for defining the performance of an injector or an engine. The two big advantages of this system are that the information presented to the test engineer is the same as that used by the performance reduction program and that the final reduced performance data is available to the on-site development engineers within an hour or two of a test. There is therefore no need to remove an engine from the test stand before the authorized performance data have been sufficiently analysed to ensure that all requirements have been met and if additional tests are required (due to exceeding the run-to-run limits) these can be carried out with a minimum of delay.

The Rocketdyne test facility at Reno has no on-site computation facilities. Test conditions are set up using chart recorders and immediate post-test data reduction is by influence coefficients and chart recorder data. The chart recorder data are also fed to a computer in the Los Angeles area by teleprinter and the results from this computer are available on the test site approximately twelve hours after a firing. The decision to remove or retest an engine has to be made on the basis of this preliminary information, since the results of the performance reduction of the digital data are not usually available until some four days after a firing, since the digital data tape has to be sent to Canoga Park for processing. If it is decided that the Rocketdyne engine will have to satisfy repeatibility limits on thrust, thrust vector and specific impulse, then improved on-site performance reduction capability will be necessary.

2.3 Performance Analysis Methods

The following comparison of the Bell and Rocketdyne performance analysis methods is based on an input by J.P.B. Cuffe to the LM ascent engine performance evaluation report given by C. Verschoore to the ascent engine program review team in April 1968 (Reference 11). The most important change in performance analysis procedure since this date is that Rocketdyne has now empirically characterized the effect of oxidizer temperature on the hydraulic resistance of the oxidizer injector (Reference 16). This characterization has recently been incorporated in the injector test performance analysis program and will shortly also be included in the engine analysis program. Since it is the effect upon the injector resistance which is characterized, and not simply the direct effect on mixture ratio (as is done on the Bell engine), standard performance values of flow rates, thrust and specific impulse are also all corrected for change in oxidizer temperature.

2.3.1 Bell Analysis

Site data are corrected to nominal interface temperatures (70° F) and pressures (170 psia) and ambient pressure (0 psia) by empirically derived linear gains. However, not all of the measured parameters are corrected and care has to be taken in comparing data obtained from tests which have been run under varied interface conditions.

2.3.1.1 Thrust

Site resultant thrust is converted to site vacuum thrust simply by adding the product of the nozzle exit area and cell pressure. Site vacuum thrust is corrected to nominal interface conditions by linear gains. No correction is made for the effect of high oxidizer temperature upon thrust. Specific impulse is calculated using site vacuum thrust, not corrected thrust: on acceptance tests, when the interface conditions are held close to the nominal levels, the difference between site vacuum I_{sp} and I_{sp} corrected to nominal interface conditions is very small. However, when limit tests are run, at high or low interface pressures, the site vacuum I_{sp} will differ significantly from the corrected I_{sp}.

Thrust coefficient is calculated using site vacuum thrust, not corrected thrust.

2.3.1.2 Flow Rates

Individual flow rates are calculated using the average of the measured flow rates from the two flow meters in each feed line. These average flow rates are added together to give total flow rate, which is then used in calculations of specific impulse and C*. Individual and total flow rates are not corrected to nominal interface conditions.

Mixture ratio is corrected to nominal propellant densities and similar feed system pressure drops by assuming that the hydraulic resistance of the feed lines remains unchanged. A further correction is applied for propellant temperatures, to take into account the increased oxidizer injector resistance which occurs with high oxidizer temperatures. This correction is a third order polynomial equation fitted to empirical data.

2.3.1.3 Thrust Chamber Pressure

Chamber total pressure is taken as 0.9914 X measured injector end chamber pressure. This value of chamber pressure is used for injector end C*I and C_f calculations on engine tests. If chamber static pressure is also measured at wall taps, upstream of the nozzle, total pressure at this location is taken as 1.0232 X measured static wall pressure. Thrust chamber C* TC is calculated using this value of total pressure on injector tests. Thrust chamber pressure measurements are not corrected to nominal inlet conditions.

2.3.1.4 Throat Area

On all runs of less than 15 seconds duration, pre-run throat area is used for C* and C_f calculations. If successive tests are carried out, then the measured area at the start of these tests is used. On runs of greater than 15 seconds duration, the C* at 15 seconds is assumed to remain constant for the rest of the test, so throat area may therefore be calculated. This calculated throat area is then used in C_f calculations.

2.3.1.5 Uncertainty Analysis

Measurement uncertainty is expressed as the sum of estimated bias error (95% confidence) and 3 sigma random error, the standard deviation being to 95% confidence. Traceability to standard and zero shift errors are included in the bias error, the random error being calculated from calibration equipment and instrumentation non-repeatibility. Taking into account instrumentation redundancy and the number of repeat tests required for each engine and injector, linear gains are used to calculate specific impulse, thrust and mixture ratio uncertainties from the measurement uncertainties. Run to run repeatibility criteria are specified for acceptance tests, to check the instrumentation non-repeatibility assumptions.

2.3.2 Rocketdyne Analysis

The Rocketdyne data reduction program is similar to the ones used on their launch vehicle engines. Data are presented in the following five forms:

Site Performance: Obtained from test measurements, at actual test conditions.

Site Vacuum Performance: Site data, corrected to 0 psia.

Site Vacuum (Standard Temperature) Performance: Site vacuum data, corrected to nominal propellant temperatures and densities.

Standard Performance: Site vacuum (standard temperature) data, corrected to nominal interface pressures.

Rated Performance: Engine performance, under standard conditions, at rated thrust and mixture ratio.

The basis of the calculations is the assumption that the site value of the feed system hydraulic resistances remains unchanged. Using a curve fit of the type C* = f (mixture ratio, thrust chamber pressure), the effect of changing site interface conditions to nominal may be obtained by iterating around thrust chamber pressure.

2.3.2.1 Thrust

Under acceptance test conditions, the Bell gains for correcting site thrust to nominal conditions result in values which are very close to those obtained from the Rocketdyne program. In comparing I_{sp} values obtained on acceptance tests, either the Rocketdyne site vacuum, Standard Performance or Rated Performance values may be compared directly with the Bell data: theoretically, when the specific impulse of different engines is being compared, rated performance values should be used. As with Bell, until recently no corrections were made for the effect of high oxidizer temperatures upon the resistance of the oxidizer injector and hence upon thrust, though this correction has now been incorporated in the engine analysis program. If specific impulse values obtained on limit tests are being compared, Rocketdyne site vacuum I_{sp} should be compared with Bell data, so long as the tests have been run under similar interface conditions.

2.3.2.2 Flow Rates

Flow rates are measured by two flow meters in series in each feed line. For each propellant, the average of the rate measured by the two meters is used in all C*, I_{sp} and mixture ratio calculations and the precentage relative agreement between the two flow meters is also calculated.

Correction is now made for the reduction in oxidizer flow rate which occurs at high propellant temperature (which, it is suspected, is caused by two phase flow in the oxidizer injector).

2.3.2.3 Thrust Chamber Pressure

C* and C_f are calculated from measured injector end chamber pressure. Nozzle total pressure is taken as 0.974 X measured injector end chamber pressure. Because of the different pressure tap locations, Rocketdyne and Bell C* data are not directly comparable.

2.3.2.4 Throat Area

Pre-run values of throat area are used in all C* and C_f calculations. If successive tests are carried out, the measured area at the start of these tests is used. The throat area is assumed to remain constant, for the purposes of C* and C_f calculations, on long duration tests.

2.3.2.5 Uncertainty Analysis

The measurement uncertainty of each instrument is calculated to a 95% confidence. The data used for these calculations are obtained from successive pre-test calibrations. The effects of these measurement uncertainties upon engine performance are then calculated by perturbing each measurement parameter in turn by an amount equal to its uncertainty, and obtaining, from the engine performance analysis program, the effect upon thrust, mixture ratio and specific impulse. On injector tests, the effect upon mixture ratio and C* are calculated. This process is repeated for each measurement parameter and the R.S.S. (root sum square) of the effects are calculated and used to express the engine's (or injector's) performance uncertainty to 95% confidence. These calculated uncertainties are for the results of one test: the fact that specific impulse is determined on two tests and injector mixture ratio on four is not taken into account in this analysis.

2.3.3 Summary of Bell and Rocketdyne Performance Analysis Methods

2.3.3.1 Values of specific impulse, thrust and mixture ratio, obtained on acceptance tests, may be directly compared.

2.3.3.2 When tests are run at high interface pressures, to make the engine operate at high chamber pressure, the Bell value of specific impulse will be high and should be compared with Rocketdyne site vacuum specific impulse data, obtained from tests run under similar interface conditions.

2.3.3.3 The effect of high oxidizer temperature causing an increase in injector resistance and hence a reduction in mixture ratio is not presently taken into account on the Rocketdyne program. However, the program is being modified to correct for this effect.

2.3.3.4 On long duration runs, the Rocketdyne program assumes that the throat area remains constant, whereas the Bell one assumes that C* does not change after 15 seconds.

2.3.3.5 Rocketdyne long duration firings are carried out using helium saturated propellants and the same performance reduction program is used for both helium saturated and unsaturated propellant tests. Bell do not normally saturate their propellants for these MDC tests.

2.3.3.6 The estimated values of performance uncertainty cannot be directly compared.

2.4 Results

2.4.1 Acceptance Test Data from Production Configuration Engines

2.4.1.1 Bell Engines

The following table shows corrected I_{sp} obtained on 15 second acceptance tests. These data are also shown plotted on Chart 4 of Appendix (B) and represent information available in May, 1968.

Injector	Engine	Run 1	I_{sp} Seconds Run 2	Average
E2C-50	Qual. 101 -	310.6	311.3	310.95
E2C-53	DVT 101A	311.4	310.7	311.05
E2C-56	LM-2	309.8	310.8	310.3
E2C-60		310.4	309.8	310.1
E2C-63	LM-3	309.3	309.4	309.35
E2C-66	LM-3 Replacement	309.5	309.0	309.25
E2C-108*	Qual. 103	309.6	309.7	309.65
E2C-108*	Qual. 104	309.5	309.3	309.4
E2CA-111	LM-4	310.4	309.8	310.1
E2CA	LM-3 Spare	310.3	310.1	310.2
	Average I_{sp}	310.0	310.0	310.0

I_{sp} standard deviation on the 20 tests ± 0.65 seconds

Standard deviation of the run to run differences in I_{sp} ± 0.61 seconds

*Injector E2C-108 was used on Qual. engine 103 and 104.

2.4.1.2 Rocketdyne Engines

The following table shows corrected I_{sp} obtained on the 15 second acceptance tests of ten of the Rocketdyne engines. These data are also shown on Chart 3 of Appendix (B).

Injector	Engine	Run 1	I_{sp} Seconds Run 2	Average
4094356	DVT201	308.1	307.9	308.0
4094355	DVT202	310.0	309.2	309.6
4094391	DVT203	309.5	309.2	309.35

Injector	Engine	I_{sp} Seconds Run 1	Run 2	Average
4094425	Qual. 0001A	309.2	308.9	309.05
4094433	Qual. 0004A	308.5	308.8	308.65
4094426	Qual. 0002A	308.4	308.2	308.3
4094430	Prod. 0003A	308.7	309.1	308.9
4094434	Prod. 0001B	310.1	309.4	309.75
4094436	Prod. 0002B	308.7	308.6	308.65
4094427	Prod. 0005A	309.3	308.2	308.75
	Average I_{sp}	309.05	308.75	308.9

I_{sp} standard deviation on the 20 tests ± 0.59 seconds

Standard deviation of the run to run differences in I_{sp} ± 0.56 seconds

Subsequent to this information being made available in May, the following engines have completed acceptance tests:

Injector	Engine	I_{sp} Seconds Run 1	Run 2	Average
4094435	Prod. 0006A	308.9	308.6	308.75
4094617	Qual. 0007A	309.9	309.8	309.85
4094619	Prod. 0003B	309.7	309.1	309.4

Taking the data from these three engines into account, the mean I_{sp} obtained on the acceptance tests of production configuration Rocketdyne engines is as follows:

Average I_{sp} (26 tests, 13 engines) 309.0 seconds

I_{sp} standard deviation on the 26 tests ± 0.60 seconds

Standard deviation of the run to run differences in I_{sp} ± 0.53 seconds

It may be seen that the run to run variations obtained on the Bell and Rocketdyne engine tests are similar: the repeatibility characteristics of the two test facilities may be therefore assumed to be essentially the same. To find out if there was any bias between the two test facilities, Rocketdyne engine #0005A was installed in the Bell engine test stand, and two acceptance tests, two performance tests (all of 15 second duration) and a full M.D.C. test series were completed. The propellants were not helium saturated on any of these tests. The results of these tests are summarized on Chart 5 and I_{sp}, thrust and mixture ratio data obtained from the Reno, Bravo and Bell test facilities are plotted on Chart 6. The values of specific impulse obtained on these tests are as follows:

		I_{sp} Second
Bravo	Predicted I_{sp} from injector test C*	309.0
Reno	Test #181, I_{sp} uncertainty $\pm$.35 sec.	309.3
	Test #182, I_{sp} uncertainty $\pm$.35 sec.	308.2
Bell	Test #497, first acceptance	309.8
	Test #502, second acceptance	309.7
	Test #503, first performance (High M.R.)	309.8
	Test #504, second performance (Low M.R.)	309.7
	Test #505, 15 seconds into M.D.C.	309.3

It will be seen that the mean value of I_{sp} obtained for the engine from its acceptance tests at Reno differs from the I_{sp} obtained of each of the acceptance tests by $\pm$0.55 seconds, which is considerably greater than the I_{sp} uncertainty given for these tests ($\pm$0.35 seconds). There is, however, very good agreement between the values of I_{sp} predicted from the Bravo tests, that measured on the first Reno test and the Bell test results. It was therefore deduced that there is no sensible bias between the two test facilities.

2.4.2 Mission Duty Cycle Data

One problem in comparing I_{sp} data obtained on M.D.C. tests is that these tests are carried out with unsaturated propellants at Bell, whereas Rocketdyne is required to use propellants which have been fully saturated with helium. The effect of using helium saturated propellants is to cause an apparent drop in specific impulse of approximately 1 second.

Rocketdyne has found that there is a degradation in the relative agreement between the two flow meters in series in each propellant feed line when saturated propellants are used. It is thought that this is caused by helium, coming out of solution on passing through the upstream flow meters, causing the downstream flow meters to read high. This results in a reduction in indicated specific impulse. However, there does also appear to be some true reduction in specific impulse when helium saturated propellants are used, the magnitude of which has not yet been determined.

A second problem in comparing Bell and Rocketdyne M.D.C. I_{sp} data is that the Bell I_{sp} is not corrected to standard conditions, site flow rates and site vacuum thrust being used to calculate their I_{sp}. Hence, Bell M.D.C tests which have been run at high chamber pressure show high values of I_{sp} and these results cannot be directly compared with Rocketdyne data, unless the Rocketdyne tests have been run under similar conditions and site vacuum performance is used for the comparison. As an example, Bell qual. engine #104 (fitted with injector #E2C-108, which was also used in qual. engine #103) had acceptance test values of I_{sp} of 309.3 seconds, but when this engine was run through a M.D.C. test at high chamber pressure, the 15 second I_{sp} value was 311.0 seconds.

Typical variations in I_{sp} with run time during M.D.C. tests are shown in Chart 2 of Appendix (A). Under similar conditions, the variation in I_{sp} during M.D.C. of the Bell engine is greater than that observed on the Rocketdyne. It is also possible that the variability of the Bell engine, under similar run conditions, is greater than that of the Rocketdyne one. The effect of propellant temperature upon chamber erosion rate and hence upon I_{sp} variation has not yet been fully characterised, but it appears that the nominal temperature of 70°F results in the greatest performance shifts: both 40 and 100°F propellant temperature tests have less performance shift, particularly in the case of the Bell engine. Throat erosion during M.D.C. is strongly influenced by the amount of chamber run time prior to the start of M.D.C.

Minimum I_{sp} occurs at the end of the M.D.C. The greatest degradation observed during M.D.C. tests, comparing the final 460 second data slice with the 15 second one, was 1.55 seconds for Rocketdyne (DVT 201, 70°F propellant temperature) and 3.1 seconds for Bell (qual. 101).

2.5 Summary

2.5.1 There are several differences in the acceptance test requirements which have to be met by the Bell and Rocketdyne engines: the most important performance difference is that, unlike Rocketdyne, Bell specifies run to run limits.

2.5.2 If Rocketdyne is required to meet run to run limits on their engine tests, improved on site performance reduction capability will be necessary.

2.5.3 The performance reduction programs used by Bell and Rocketdyne, though using different techniques, give comparable corrections of acceptance test results. However, the specific impulse corrections when applied to off limit test data are not directly comparable.

2.5.4 Rocketdyne M.D.C. engine tests are carried out using helium saturated propellants: the Bell tests do not normally use saturated propellants. This means that M.D.C. I_{sp} results cannot be directly compared, even if the tests are carried out under the same interface conditions.

2.5.5 The acceptance test I_{sp} data obtained on the Bell and Rocketdyne facilities may be directly compared, there being no sensible bias or difference in variability between the two engine test stands.

2.5.6 The mean specific impulse obtained on twenty acceptance tests of ten Bell engines is 310.0 seconds, standard deviation $\pm$0.65 seconds.

2.5.7 The mean specific impulse obtained on twenty six acceptance tests of thirteen Rocketdyne engines is 309.0 seconds. standard deviation $\pm$0.60 seconds.

2.5.8 The Rocketdyne calculated I_{sp} uncertainty is an indication of the calibration repeatibility of the instrumentation system: it is not a realistic quantitative measure of the actual specific impulse uncertainty under flight conditions.

2.5.9 The true quantitative effect upon engine specific impulse of helium in the propellants is not presently known.

2.5.10 Unlike the TRW LM descent engine, no attempt is made to predict the throat area changes of a LM ascent engine during M.D.C. from that engine's compatibility acceptance test data.

2.5.11 Under similar conditions, the Bell engine shows a greater change in specific impulse during M.D.C. than the Rocketdyne one. However, for neither engine are the effects of change in interface conditions upon specific impulse during M.D.C. adequately characterised.

2.5.12 Suggested criteria for the LM ascent engine acceptance tests are given in Appendix (B). For these criteria to be properly utilized, further tests are required to adequately characterise the engine's performance under M.D.C. conditions and to correlate engine and barrel data.

2.6 References

1) Boeing Report D2-117060-1 "Apollo Spacecraft Engine Specific Impulse" May 6, 1968.

2) Rocketdyne Process Specification ST0230RA0034 "RS-18 Valve and Injector Assembly Acceptance Test Data Reduction Procedure" January 5, 1968.

3) Rocketdyne Process Specification ST0230RA0052 Revision A "RS-18 Rocket Engine Valve and Injector Assembly Acceptance Test" March 20, 1968.

4) Rocketdyne Process Specification ST0230RA0035 Revision A "RS-18 Rocket Engine Assembly, Acceptance Test Data Reduction Procedures and Performance Criteria" March 6, 1968.

5) Rocketdyne Process Specification ST0230RA0053 Revision A "RS-18 Rocket Engine Assembly Acceptance Test" March 8, 1968.

6) Bell Report #8258-928031 Revision B "Bell Model 8258 Injector and Valve Assembly Detail Test Plan" September 11, 1967.

7) Bell Report #8258-928015 Revision A, Amendment 2 "Bell Model 8258 Lunar Module Ascent Engine Acceptance Test Plan Production Engines P3 and Subsequent" November 30, 1967.

8) Bell Report #8258-927019 Revision B "Determination of Bell Model 8258 Altitude Performance" January 15, 1968.

9) Bell Note "Temperature Adjustment for Corrected Mixture Ratio Data"

10) Rocketdyne Draft Description of REA Steady-state Data Reduction Program, Presented at MSC 15-16 February, 1968.

11) LM Ascent Engine Performance Evaluation Report by C. Verschoore to the Ascent Engine Program Review Team (unpublished) April 1968.

12) Presentation produced by C. Verschoore and J.P.B. Cuffe "Performance Comparison Between Bell and Rocketdyne LM Ascent Engines". Incorporated in the LM Ascent Engine Program Review Team presentation given by J.G. Thibodaux at MSC May 23, 1968.

13) Rocketdyne "Lunar Module Ascent Engine Program Review" 21 May, 1968; 11 April 1968 (and Supplement); 18 March 1968 et seq.

14) Bell "LM Ascent Engine Program Review" 21 May 1968 #8248-910031; 10 April 1968 #8258-910030 et seq.

15) J. P. B. Cuffe Trip Reports - May 27, 1968 (MSC-Houston); April 16, 1968 (Bell, Rocketdyne); April 8, 1968 (Rocketdyne); March 29, 1968 (MSC-Houston).

16) Telecon. J. P. B. Cuffe to J. Ervin (Rocketdyne) June 13, 1968.

17) Rocketdyne internal letter IL 8114-4104 July 11, 1968, J. Ervin to S. J. Domokos, reviewing draft of Boeing Report D2-117060-2.

<u>Appendix (A): Presentation on performance comparison between Bell and Rocketdyne LM Ascent engines.</u>

The attached charts were produced by C. Verschoore and J. P. B. Cuffe for the LM Ascent Engine Program Review team (Reference 12) and are discussed in the following notes:

<u>Chart 1. LM Ascent Engine Performance Summary Chart.</u> The $C^* \eta$ (C^* efficiency) values are not strictly comparable, due to different pressure tap locations and injector end pressure to total pressure conversion constants. Therefore, the difference of 0.69% in $C^* \eta$ simply means that the Bell engine would be expected to have a slightly higher specific impulse and not necessarily one precisely 1.8 seconds higher.

The 15 second acceptance test data are directly comparable. These data are obtained from two tests on ten engines for both Bell and Rocketdyne. The standard deviations of the 20 I_{sp} values are $\pm$ 0.65 seconds (Bell) and $\pm$ 0.59 seconds (Rocketdyne).

The normalized results of the Rocketdyne and Bell performance reduction programs may be directly compared if data from acceptance tests are used, since these tests are carried out under conditions which are close to nominal. The Bell acceptance data are shown plotted out on chart 4, the Rocketdyne on chart 3.

The instrumentation uncertainty value of 2 seconds is essentially a subjective estimate, obtained by rounding the Bell 3σ uncertainty estimate to a whole number. Only M.D.C. tests using propellants at 70°F have been used in the calculations of integrated specific impulse. This temperature appears to result in the greatest change in I_{sp} during M.D.C.: 40°F and 100°F propellant temperature M.D.C. tests show less performance shift. The effect of the helium saturation used in the Rocketdyne M.D.C. tests has been corrected for to allow both contractor M.D.C. tests to be directly compared.

The end of M.D.C. I_{sp} has also been taken from the same two tests for each contractor.

<u>Chart 2. Performance Degradation During M.D.C.</u> The greater shift in performance during the M.D.C. is a characteristic of the Bell engine. This characteristic is confirmed by the greater change in measured throat area observed on the Bell engine. However, the measured change in throat area, and hence in nozzle area ratio, does not account for all of the observed shift in performance: the effect of erosion on the

nozzle contour also causes some reduction in performance.

Chart 3. Rocketdyne Acceptance I_{sp} Data. The data are from the final (15 second) time sample of two acceptance tests on ten different production configuration engines.

Mean I_{sp} of the 20 tests: 308.9 seconds

I_{sp} standard deviation on the 20 tests: $\pm$ 0.59 seconds

Standard deviation of the run to run differences in I_{sp}, 10 engines: $\pm$ 0.56 seconds

Chart 4. Bell Acceptance I_{sp} Data. The data are from the final (15 second) time sample of two acceptance tests on ten different production configuration engines.

Mean I_{sp} of the 20 tests: 310.0 seconds

I_{sp} standard deviation on the 20 tests $\pm$ 0.65 seconds

Standard deviation of the run to run differences in I_{sp}, 10 engines: $\pm$ 0.61 seconds.

From Charts 3 and 4, it may be seen that the Bell engine has a slightly higher specific impulse. The run to run and engine to engine repeatability data for the two contractors are comparable.

Chart 5-7. Rocketdyne engine #0005A, after acceptance test at Reno, was installed in the Bell engine facility. Four 15 second performance and one full M.D.C. test were carried out. The propellants were not saturated with helium on any of these tests. The results show that there is no significant bias between the two test facilities: it has already been shown that their repeatability was comparable. The predicted specific impulse (Chart 5) is calculated form the mean value of C* obtained on the injector tests at the Bravo facility. The two Rocketdyne acceptance tests show a greater variation in I_{sp} than do the Bell tests, the first Rocketdyne acceptance test and the Bravo predicted I_{sp} values. It is therefore probable that the I_{sp} on the second Rocketdyne acceptance test is in error: the Rocketdyne calculated I_{sp} uncertainty on the acceptance tests on this engine was $\pm$ 0.35 seconds, yet the difference in the I_{sp} on the two tests was 1.1 seconds.

The thrust vector shift characteristic of the Rocketdyne engine was confirmed on the M.D.C. test at the Bell facility. This shift is caused by the uneven ablation and erosion which occurs in the Rocketdyne chamber, a result of the asymmetric injector design. By offset drilling the mounting bushes by 0.100", Rocketdyne considers that the engine thrust center line can be kept within the specification limit of 0.3" of the thrust chamber center line.

Chart 8. Performance Evaluation Conclusions: This chart summarizes the information given in charts 1 through 7.

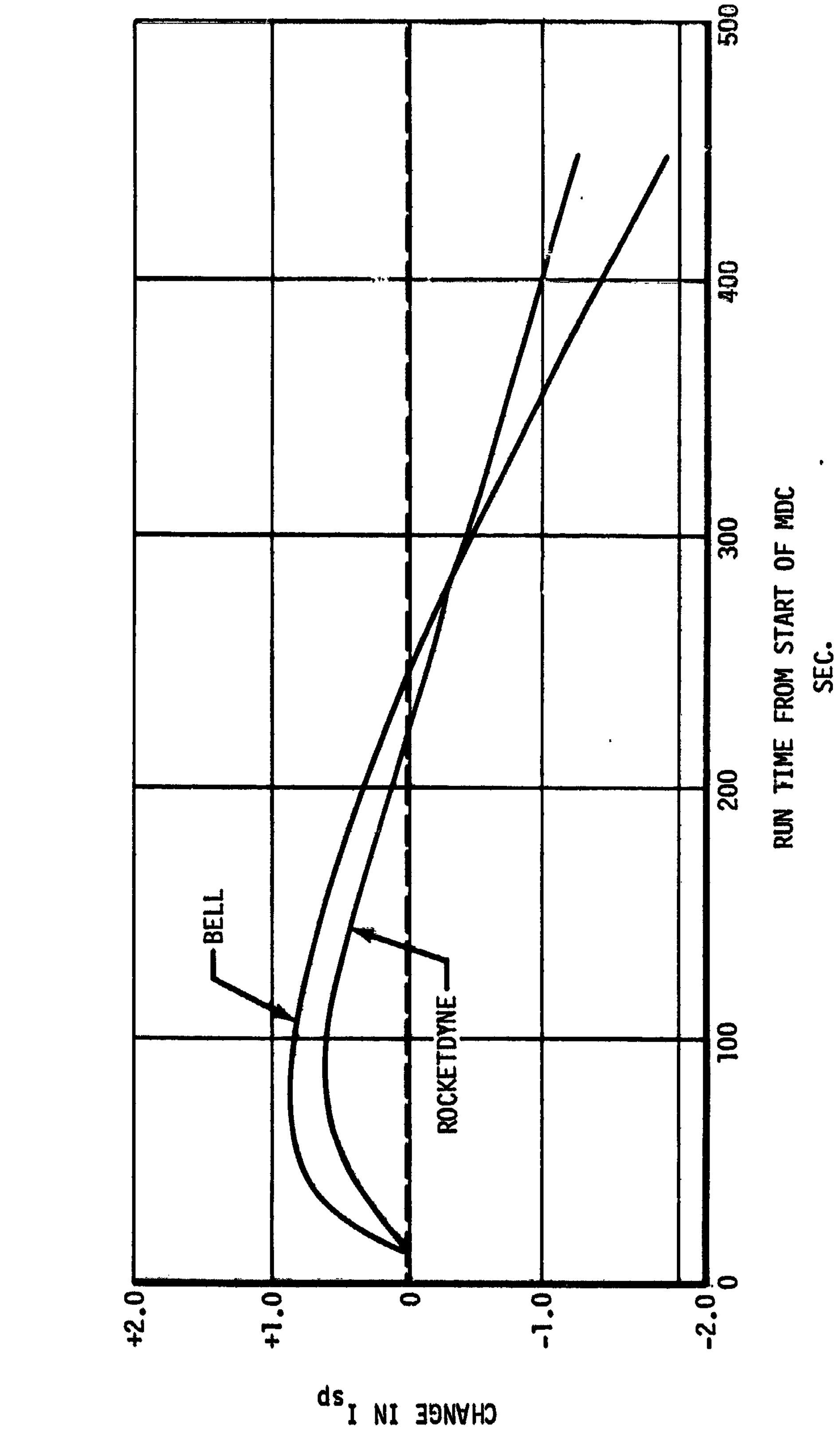
LM ASCENT ENGINE PERFORMANCE SUMMARY CHART

		I_{SP}			
	$C^* \eta$	15-SECONDS ACCEPTANCE	INTEGRATED MDC MINUS 2-SECONDS INSTRUMENTATION UNCERTAINTY*	END OF MDC MINUS 2-SECONDS INSTRUMENTATION UNCERTAINTY*	SPECIFICATION MINIMUM I_{SP} REQUIREMENT
BELL	96.8	310.0 (10 ENGINES)	308.0 (2 ENGINES)	306.2 (2 ENGINES)	306.3
ROCKETDYNE	96.2	308.9 (10 ENGINES)	306.6 (2 ENGINES)	305.7 (2 ENGINES)	306.3

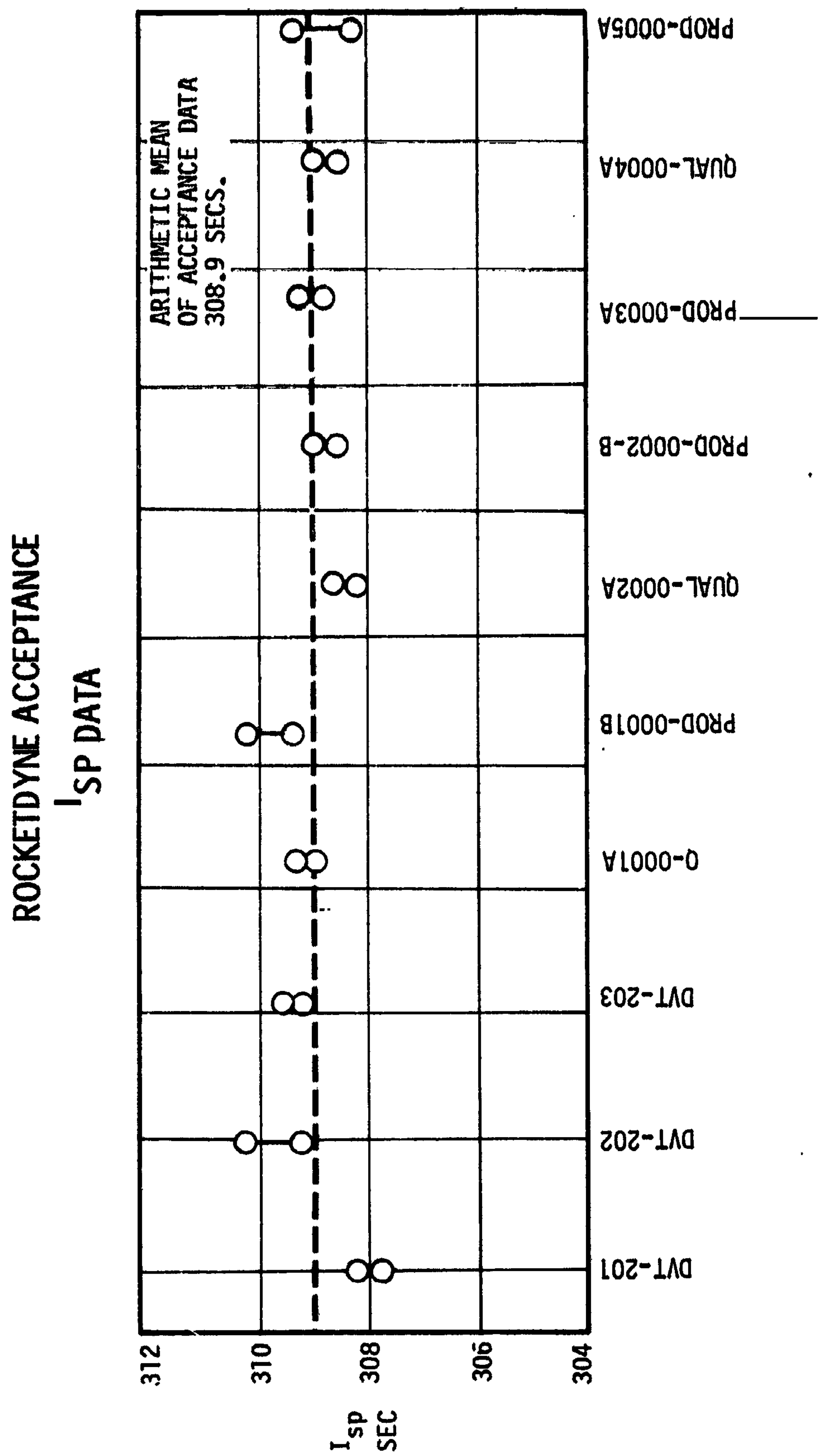
*INSTRUMENTATION UNCERTAINTY EQUAL FOR BOTH CONTRACTORS AND BASED ON HIGH CONFIDENCE LIMIT STATED AT BELL.

APPENDIX (A)
CHART 1

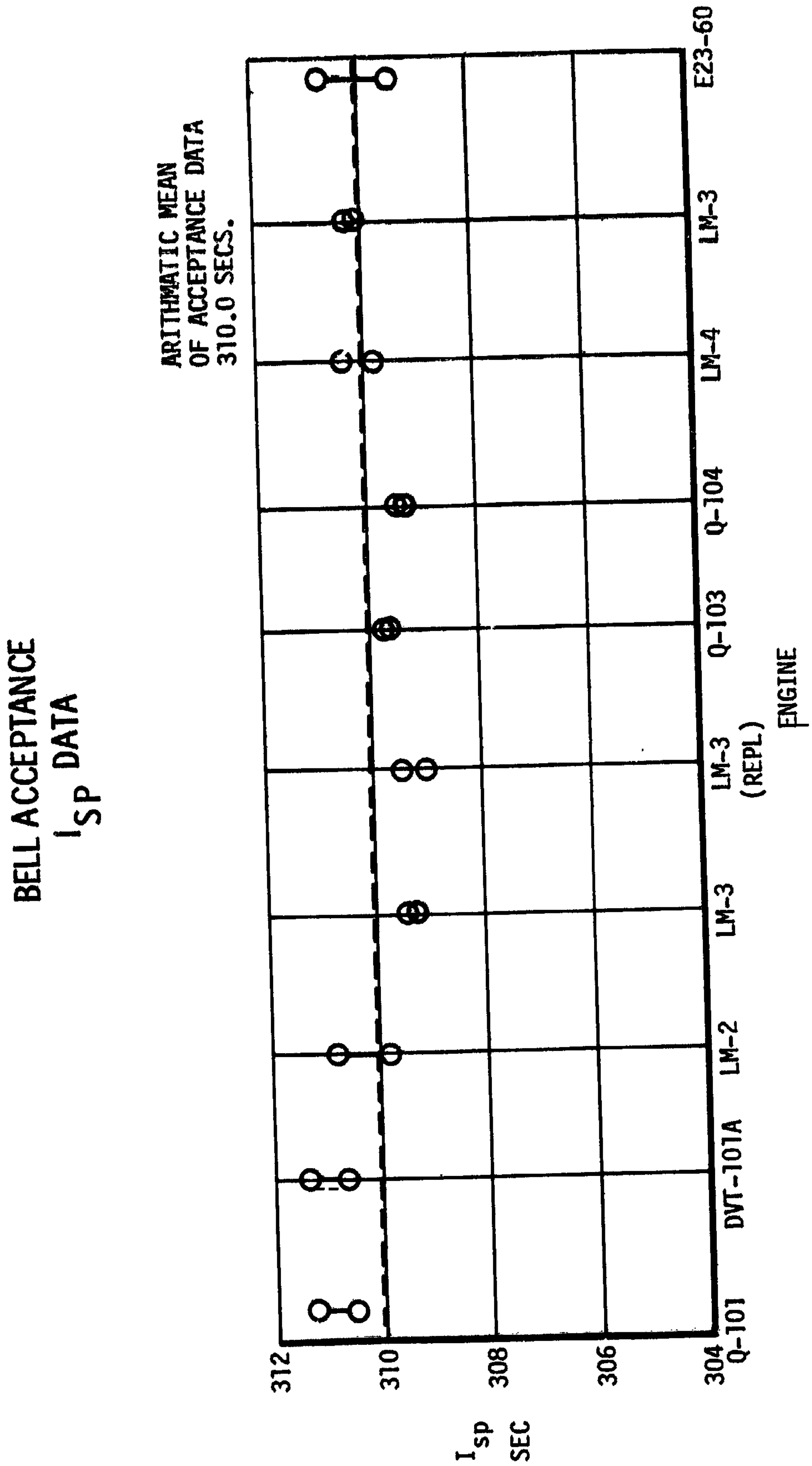
ROCKETDYNE AND BELL ENGINE PERFORMANCE DEGRADATION DURING MDC WITH 70°F PROPELLANTS



APPENDIX 'A' CON'T
CHART 2



APPENDIX A CON'T.
CHART 3



APPENDIX'A' CON'T
CHART 4

PERFORMANCE COMPARISON OF ROCKETDYNE ENGINE AT BACTEST FACILITY

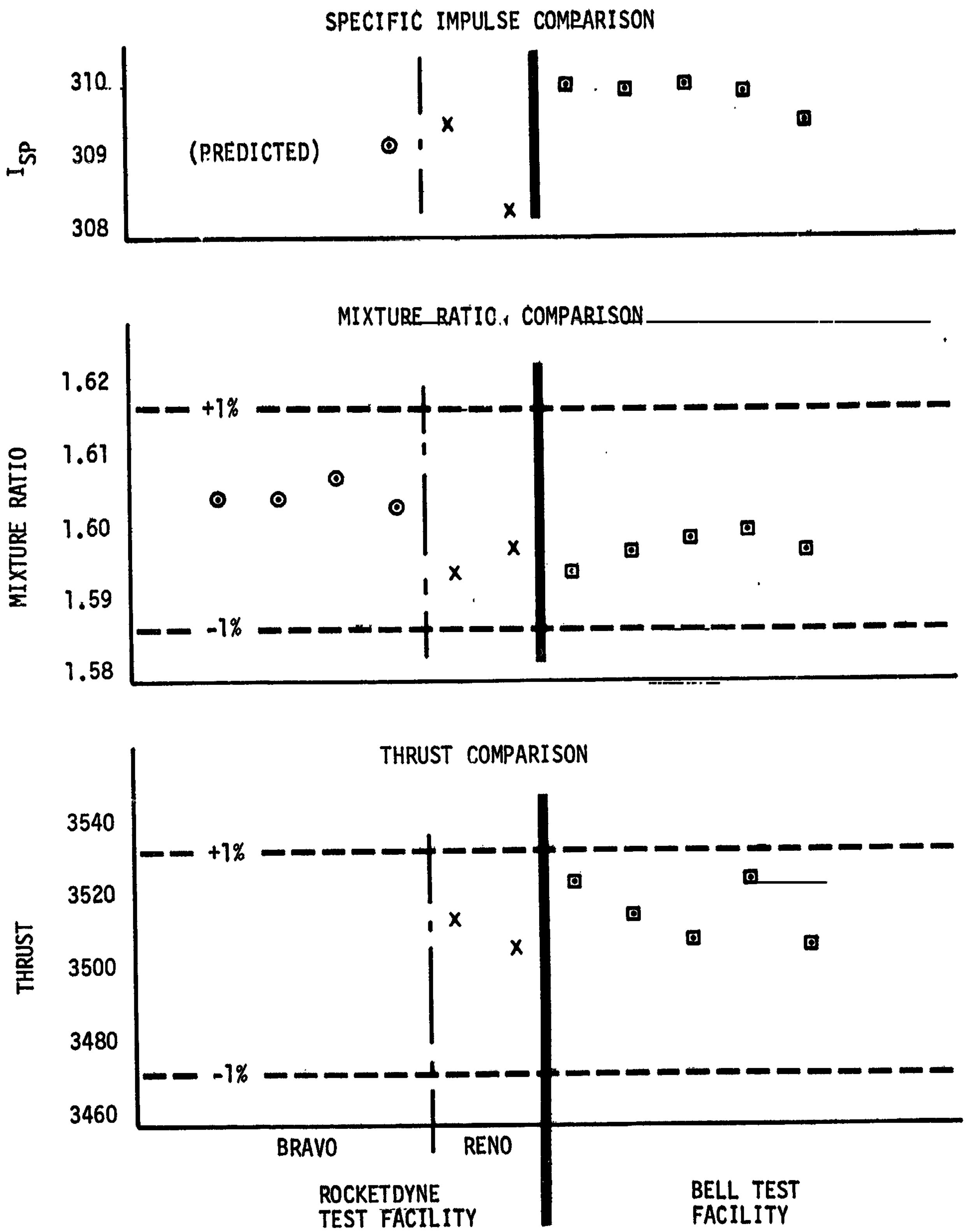
- COMPARABILITY OF I_{SP} DATA FROM ROCKETDYNE AND BELL TEST FACILITIES ESTABLISHED

- I_{SP}
M.R. GOOD AGREEMENT WITH ROCKETDYNE
F

- THRUST VECTOR SHIFT FOR ROCKETDYNE ENGINE CONFIRMED

APPENDIX 'A' CON'T.
CHART 5

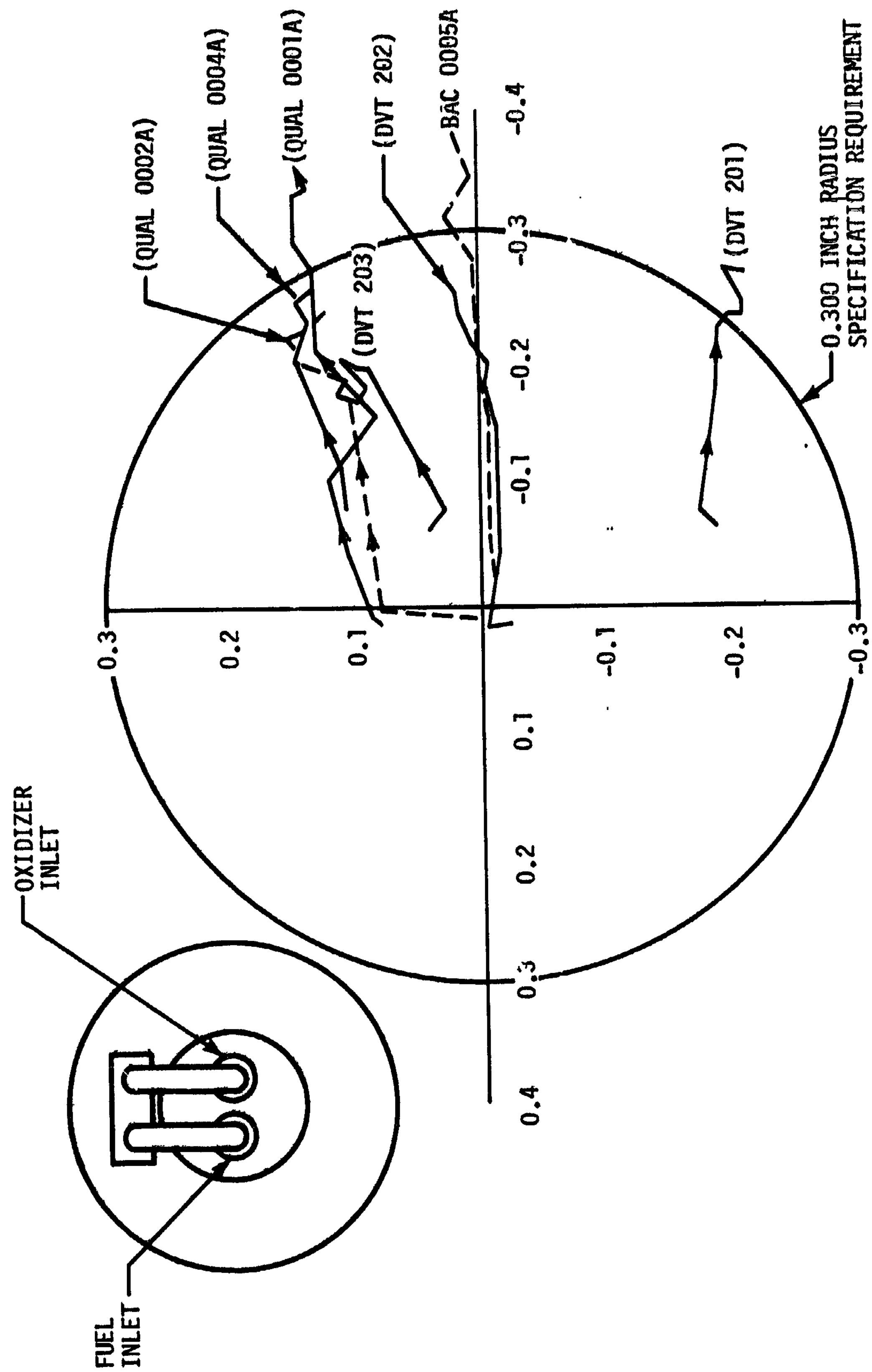
PERFORMANCE COMPARISON OF ROCKETDYNE ENGINE #0005A AT ROCKETDYNE AND BELL TEST FACILITIES



APPENDIX A CON'T.
CHART 6

THRUST VECTOR DISPLACEMENT AT REA INTERFACE PLANE OVER MDC DURATION

LOOKING AFT



APPENDIX 'A' CON'T
(CHART 7)

PERFORMANCE EVALUATION CONCLUSIONS

- I_{SP} - COMPARABLE WITH BELL APPROXIMATELY 1 SECOND HIGHER
- M.R. - COMPARABLE
- THRUST - COMPARABLE
- THRUST VECTOR SHIFT - BELL NO PROBLEM, ROCKETDYNE REQUIRES BUILT-IN OFFSET TO ACCOMODATE DEMONSTRATED SHIFT
- TEST FACILITIES & INSTRUMENTATION - COMPARABLE

(CHART FROM REFERENCE 12)

APPENDIX 'A' CON'T.
CHART 8

Appendix B. Suggested criteria for LM Ascent engine acceptance tests

The following criteria are based on an input by J. P. B. Cuffe to the LM Ascent engine performance evaluation report given by C. Verschoore to the ascent engine program review team in April 1968. (Reference 11). Specific numerical limits are not given, since these requirements are intended to be used as a basis for discussion. For these criteria to be valid and for their full benefit to be utilized, sufficient quantitative information must be available to be able to:

a) Correlate C* and compatibility data obtained from engine and barrel tests.

b) Define the effect of variations in:

Chamber pressure

Mixture ratio

Propellant temperature

Propellant helium content

upon each of the following:

Specific impulse

Thrust

Corrected mixture ratio

Chamber compatibility and throat erosion

SUGGESTED CRITERIA FOR ACCEPTANCE TESTS

1. Injector Tests

Liner tests, of 15 second duration. A diagram of the liner thrust chamber assembly is attached.

Minimum of four valid tests, with no hardware changes. For the Rocketdyne injector, the first test is not to be considered valid, to allow for the effects of the filters bedding down.

Limited range of allowable test conditions (interface pressures, pressure differences, propellant temperatures and temperature differences). Propellants to be temperature conditioned, but not helium saturated.

Mixture ratio limits and minimum value of C* to be specified, for data corrected to nominal conditions. (Standard performance data, using the Rocketdyne definition).

Accuracy criteria to be specified. The resulting uncertainties in corrected mixture ratio and C* to be subtracted from the allowable mixture ratio range and added to the minimum allowable C* respectively.

Repeatability limits on mixture ratio and C* to also be specified, possible by giving a maximum allowable value for the standard deviation of these parameters ($\pm$ 0.2%?). Penalty tests in excess of the four required may be necessary to meet this standard deviation limit. Alternatively, maximum allowable variations in corrected mixture ratio and C* to be specified.

2. Compatibility Test

One barrel test, of 500 second duration. A diagram of barrel thrust chamber assembly is attached.

Same test conditions as for injector tests, including propellants not helium saturated.

Quantitative data on effective throat area changes to be used for flight prediction of thrust, specific impulse and thrust vector variations.

No hardware changes to injector and valve assemblies.

Gouging and erosion limits to be specified.

Agreement with injector test mixture ratio required.

Accuracy criteria to be specified.

3. Engine Acceptance Tests

A minimum of two 15 second tests. No hardware changes to injector, propellant lines or valve assembly.

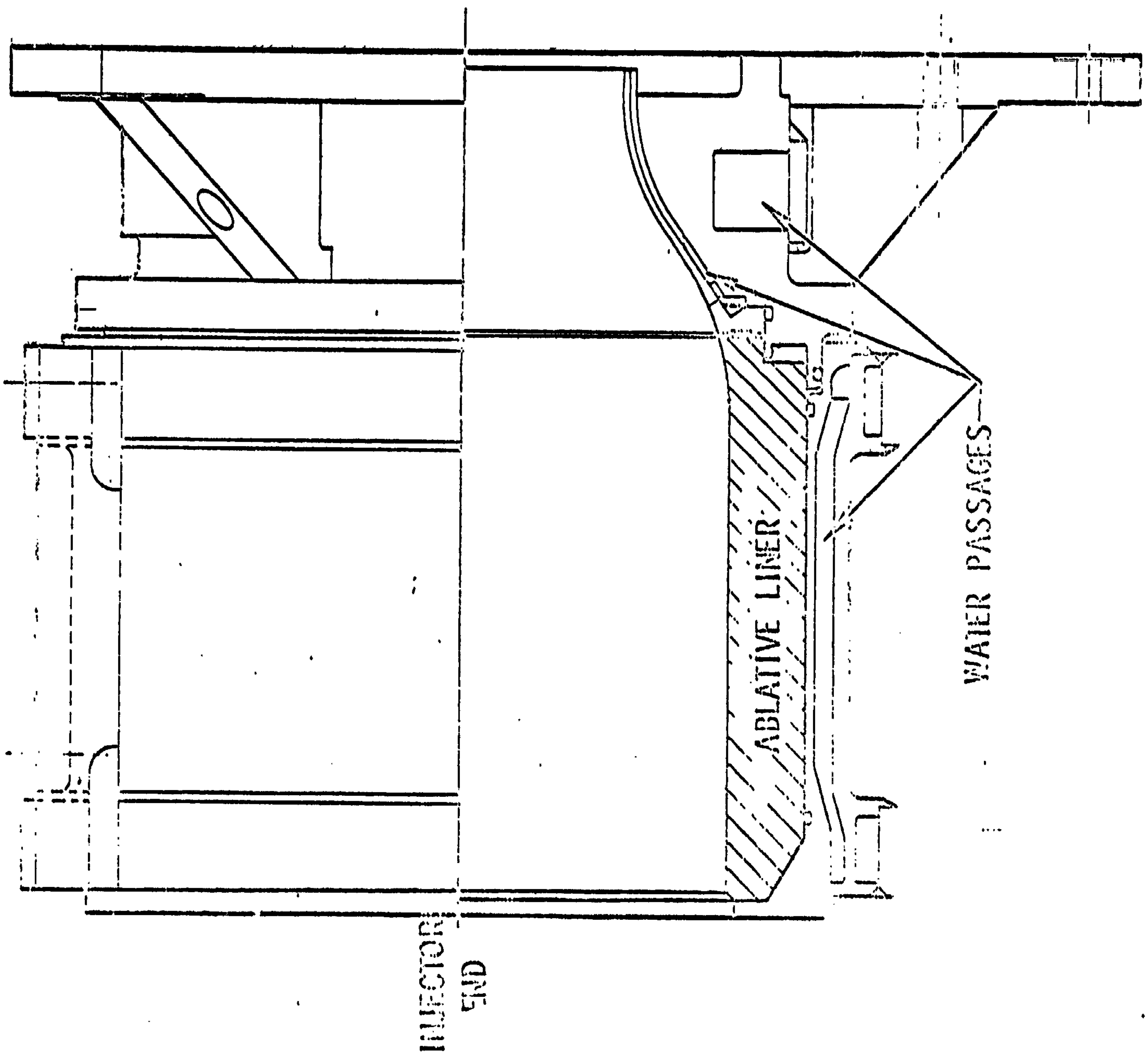
Same inlet conditions as for injector tests, together with a maximum cell pressure limit. Propellants not helium saturated.

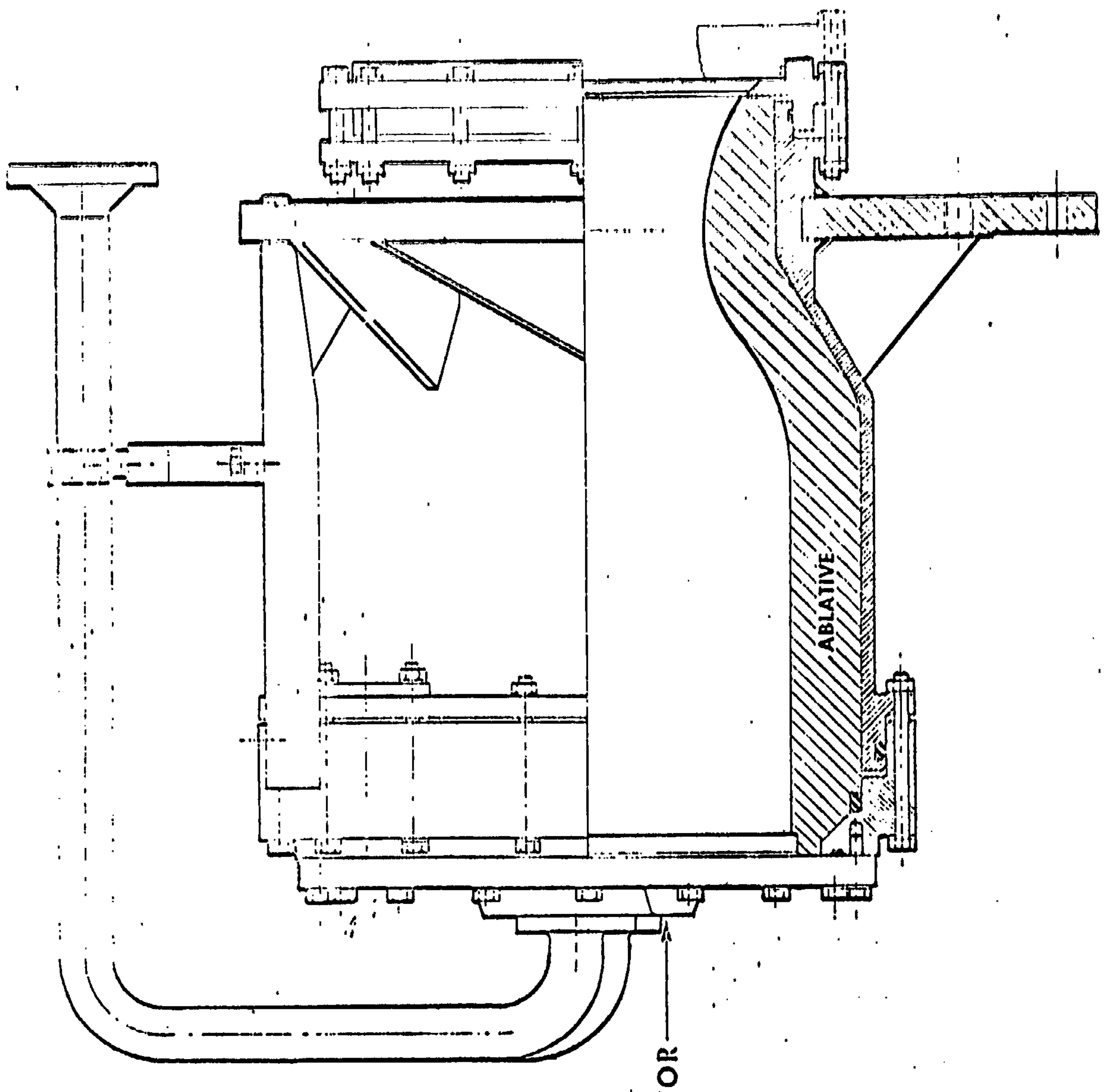
Thrust (resultant and vector) limits and minimum value of I_{sp} to be specified for data corrected to nominal conditions (Standard performance data, using the Rocketdyne definition). Predicted I_{sp} (from injector C* tests) and mean value of Standard performance I_{sp} obtained from engine tests to agree to within specified limit (1/2 second?). Also agreement with injector test mixture ratio and barrel test C* (after making correlation correction) required.

Accuracy criteria to be specified. The resulting uncertainties in corrected thrust and I_{sp} to be subtracted from the allowable thrust range and added to the minimum allowable I_{sp} respectively.

Repeatability limits on thrust, thrust vector and I_{sp} to also be specified. If these differ by more than specified amount on two tests, additional penalty tests to be carried out (Bell procedure).

WATER-COOLED THRUST CHAMBER




ABLATIVE
INJECTOR

3.0 TRW LM Descent Engine (June 1968)

3.1 Introduction

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. Prediction of the engine's performance during a full mission duty cycle is complicated by the large amount of throat erosion which occurs (up to 20%) and by the engine's sensitivity to small changes in interface conditions.

A description of the methods used to determine the vacuum specific impulse of the TRW engine is summarized in Section 3.2, together with full details of the acceptance test performance criteria. Performance analysis is discussed in Section 3.3 and available results are presented in Section 3.4.

3.2 Acceptance Test Procedures and Requirements

3.2.1 Test Procedures

A description of the LM descent engine acceptance test procedures is given in Reference 1: full details of the test plans and requirements are given in Reference 2. To summarize, the engine's head end assembly (H.E.A.) is fitted to a water cooled steel chamber and installed in one of the vertical engine test stands (V.E.T.S.). After calibration, two satisfactory acceptance tests are required. With the H.E.A. fitted to a chamber with a fiber glass throat, two compatibility test firings are then carried out. The "T_{120} time", a quantitative measure of the erosion characteristics of each H.E.A., is determined on these tests. This "T_{120} time" is the time taken to erode the fiber glass throat area by 20% when the H.E.A. is run under nominal conditions at F.T.P. Finally, the H.E.A. is installed on its flight ablative chamber and the complete engine is acceptance tested in the high altitude test stand (H.A.T.S.). Two satisfactory acceptance tests are again required.

The V.E.T.S. and H.A.T.S. facilities at the TRW Capistrano test site (C.T.S.) are primarily intended as performance, rather than as propulsion system development, test stands. The emphasis is on steady state performance accuracy, not on vehicle simulation. Unlike the LMD stage propulsion system, which uses helium, nitrogen is sed to pressurize the propellant tanks on both the V.E.T.S. and H.A.T.S. facilities. The engine's qualification test program (Reference 3) was also carried out at C.T.S.

3.2.2 Acceptance Test Requirements

Only performance requirements will be discussed in this section. All the information presented is from Reference 2.

Thrust Vector Alignment - Displacement of the geometric thrust vector line from the engine reference line at the gimbal plane must be less than 0.050 inches. Angular deviation of the geometric thrust vector line from the engine reference line must be less than 0.20 degrees of arc. The engine's thrust vector alignment is not measured on the test stand (a single thrust load cell is used at H.A.T.S.) but is optically determined.

Specific Impulse - The best estimate of H.A.T.S. vacuum specific impulse at nominal interface conditions and with zero throat erosion must be equal to or greater than:

304.0 seconds at F.T.P.
298.5 seconds at 50% thrust
294.5 seconds at 25% thrust

These limits are for normalized test data: estimated measurement uncertainty does not have to be subtracted from the test results. The estimated uncertainty in measured vacuum specific impulse, obtained from Reference 5, is shown in Figure 1. Using these uncertainty values, the 3σ minimum levels of specific impulse given by the acceptance test criteria are:

303.3 seconds at F.T.P.
297.3 seconds at 50% thrust
291.6 seconds at 25% thrust

F.T.P. Thrust - The best estimate of vacuum thrust after 5 seconds operation at F.T.P.. + the 3σ measurement uncertainty, must be within the band of 9,712 to 10,027 lb. The 3σ measurement uncertainty of vacuum thrust at F.T.P. is given in Reference 5 as 0.16% or 16 lb. The best estimate of vacuum thrust, after 5 seconds operation at F.T.P., must therefore be within the band of 9728 to 10,011 lb.

Thrust Repeatibility - The difference between the values of vacuum thrust, corrected to nominal conditions, obtained after 5 seconds at F.T.P. on the two acceptance tests must be less than or equal to 70 lb.

Mixture Ratio - The best estimate of mixture ratio, after 5 seconds operation and corrected to nominal conditions, will be calculated from both V.E.T.S. and H.A.T.S. data separately. These estimates of mixture ratio, $\pm$ their 3σ measurement uncertainty, must both lie within the following limits:

1.6:1 $\pm$ 0.014 at F.T.P.
1.6:1 $\pm$ 0.025 at 50% thrust
1.6:1 $\pm$ 0.035 at 25% thrust
1.6:1 $^{+0.100}_{-0.045}$ at 10% thrust

These mixture ratio limits are shown plotted in Figure 2. The values of 3σ measurement uncertainty were obtained from Figures A-13 through A-16 of Reference 13. Though Reference 13 is dated August, 1966, the engine acceptance test reports (Reference 9) show that the 3σ measurement uncertainty values given in this reference are still used for acceptance criteria.

Compatibility - The T_{120} times on the two compatibility tests must both exceed 75.0 seconds.

There is no GAEC criterion for values of C* obtained on the V.E.T.S. tests. However, TRW lists C* performance data in the engine acceptance test reports and combines these data with H.A.T.S. I_{sp} and C* measurements to calculate a "merged" I_{sp} estimate. The method used for this calculation is described in Reference 1. Both "merged" and H.A.T.S. I_{sp} values are listed in an engine's acceptance test report, but GAEC uses only the H.A.T.S. data for their acceptance criteria.

There is also no GAEC criterion for a minimum value of average M.D.C. I_{sp}. TRW predicts the throat area changes and specific impulse of each engine during a standard M.D.C., giving the results of these predictions in the engine's acceptance test report (see Section 3.3) but these results are for information only and are not used as a basis for engine acceptance.

3.3 Performance Analysis

The methods used to normalize acceptance test data to standard conditions are given in Reference 1. Because acceptance tests are carried out under conditions which are maintained very close to the standard values, the magnitude of the normalization corrections are small, in the order of only 0.1 seconds in specific impulse.

3.3.1 Performance Analysis at Off Nominal Conditions

Using performance data obtained from qualification test configuration LM descent engines (Reference 3), a set of influence coefficients were calculated and included in Reference 4, Volume 1. These influence coefficients are listed in Table 1 and are plotted in Figures 3, 4 and 5. These figures show the effect of a unit change in independent variable upon a dependent variable. The independent variables considered are oxidizer and fuel inlet pressures (p.s.i.), oxidizer and fuel inlet temperatures (°F) and throat area (%). The dependent variables are vacuum specific impulse (seconds), vacuum thrust (lb.) and mixture ratio. It will be seen that the engine's characteristics in the throttled region are usually different from those obtained at F.T.P. When throttled, flow into the engine is controlled by conditions at the cavitating flow control valves, variation in thrust chamber conditions having no effect. On the other hand, at F.T.P. the engine flow rates are determined by the hydraulic resistances and pressure drops through the feed systems and are therefore influenced by changes in thrust chamber conditions.

Specific Impulse (Figure 3)

Change in throat area appears to have a constant effect upon specific impulse, irrespective of throttle setting. At the end of M.D.C., an engine with a low, though still acceptable, T_{120} time might be expected to have its throat area increased by some 20%, resulting in a reduction of 5.2 seconds in specific impulse. At F.T.P., variations in engine inlet temperature and pressures have little effect upon specific impulse. However, when throttled, these variables appear to have a strong effect on specific impulse. In fact, at 10% thrust, a change of 1°F in propellant temperature is predicted to change specific impulse by as much as 1 second. No explanation is offered for these very high temperature gains and it is suggested that at present they be considered suspect.

Thrust (Figure 4)

At F.T.P., increase in throat area results in an increase in propellant flow rate sufficient to cause an increase in thrust, despite the lower specific impulse. When throttled, increase in throat area has no effect upon flow rates, so there is a decrease in thrust because of the specific impulse degradation. The percentage change in thrust due to change in propellant inlet temperature and pressures is approximately the same during F.T.P. and throttled operation.

Mixture Ratio (Figure 5)

Throat area change has no effect upon mixture ratio. The effect of change in inlet pressure upon mixture ratio is twice as high at F.T.P. as when the engine is throttled. At F.T.P., the effect of a difference between fuel and oxidizer inlet pressures is considerable: if both inlet pressures increase or decrease by 1 psi, the effect on mixture ratio is negligible, but if one inlet pressure was increased and the other decreased by 1 psi, there would be a shift of 1% in mixture ratio.

3.3.2 Prediction of Throat Area Changes During M.D.C.

The effect of change in throat area upon engine performance was discussed in the previous section. The prediction of throat area change during M.D.C. is performed by the Victory VII computer program (References 6 and 7). This program is based upon a regression analysis of data obtained on the qual B tests (Reference 3). This analysis shows that the rate of change of throat area at any time is a function of the sum of the following factors:

- $(T_{120}\ \text{time})^{-4.9}$
- Amount of throat erosion which has already taken place
- Throttle setting
- Mixture ratio and injector Δp ratio
- Total impulse already produced by engine

The T_{120} time is measured for every engine on acceptance test: the other four factor are determined by how the engine is operated during M.D.C.

The following examples give an indication of the approximate quantitative effect upon the rate of change of throat area due to variation in some of these factors:

- Increase in mixture ratio from 1.6 to 1.7 increases throat erosion rate by a factor of 4.
- Decrease in propellant temperature from 90 to 50°F increases throat erosion rate by a factor of 2.

- Qual Engine #4 (T_{120} time 80 seconds) had 50% more throat erosion on a full M.D.C. than did Qual Engines #5 (T_{120} time 130.5 seconds) and #6 (T_{120} time 96.5 seconds).

3.3.3 Uncertainty Analysis

In Reference 8, the following values of 1σ specific impulse uncertainty are quoted.

- Uncertainty in specific impulse measurement on acceptance test $\pm$ 0.52 seconds. This value is in reasonable agreement with Figure 1.
- Engine run to run variability, $\pm$ 0.46 seconds.
- Uncertainty in prediction of actual engine operating conditions during M.D.C. $\pm$ 0.54 seconds.
- Uncertainty in prediction of change in specific impulse during M.D.C. with assumed operating conditions $\pm$ 0.40 seconds. This value is in good agreement with Section 3.4.2.
- Engine to engine variability $\pm$ 0.91 seconds.

If flight performance predictions are made using individually selected engines, then the 1σ uncertainty for the in-flight specific impulse prediction is the root sum square of the first four items, which is $\pm$ 0.97 seconds. However, if flight performance predictions are made on the basis that any engine which has passed acceptance test could be used, the engine to engine variability has to be included in the prediction uncertainty. In this case, the 1σ uncertainty for the in-flight specific impulse prediction is the root sum square of all of the uncertainties and variabilities, which is $\pm$ 1.33 seconds.

3.4 Results

3.4.1 Acceptance Test Specific Impulse Data

TRW LM Descent Engine Specific Impulse

Engine #	Acceptance Test Data (Reference 9) Date	T_{120} (sec)	I_{sp} (sec) FTP	I_{sp} (sec) 50%	I_{sp} (sec) 25%	Predicted Average I_{sp} On LLM-5 MDC (Reference 8)
1015	1/24/68	100.1	303.0*	299.1	295.3	
1020	4/19/67	89.7,94.3	304.4	300.4	294.1*	301.6(O.K.)
1021	7/14/66	73.0,72.5	305.0	294.0*	297.8	
1023(qual #7)			302.3*	299.0	294.1	299.56
1024(qual #6)	5/9/67	96.5	303.7*	300.7	298.7	301.3(O.K.)
1025(qual #4)			304.8	299.1	299.4	300.0
1025(c)(qual#8)	4/28/67	80.4	304.5	301.3	297.2	300.74
1026(LM-1)	5/12/67	116.6, 108.6	303.7*	297.9*	297.9	300.92
1028	6/2/67	85.5,84.1	303.2*	300.7	297.3	300.3
1030(LM-3)	12/5/67	83.0,78.9	303.0*	299.1	295.5	300.0
1034(qual #5)	3/3/67	102.8, 104.2	303.8*	301.5	298.5	301.92(O.K.)
1036						299.2
1037(LM-2)	6/20/67	97.9, 108.5	304.2	301.4	297.1	301.4(O.K.)
1038(qual #9)	8/6/67	86.7,90.9	303.6*	299.4	295.5	300.2
1039	12/2/67	81.9,82.0	303.3*	300.8	295.5	300.0
1042	3/15/68	91.4,83.0	302.6*	300.4	297.2	300.8
		Acceptance Test Minimum Requirements				
		75.0,75.0	304.0	298.5	294.5	298.8 (NOTE)

*Do not satisfy acceptance test minimum requirements.

(O.K.) Greater than or equal to 301.3 seconds, the 97.5% probability criteria for the LLM-5 M.D.C. (Reference 8).

NOTE: This prediction was made assuming lowest allowable level of thrust at F.T.P., highest allowable value of mixture ratio when throttled and worst case injector pressure drop (as observed on #1025).

All of the engines had adequate T_{120} times.

Most of the engines were accepted with a waiver on their specific impulse. All cases of acceptance test specific impulse being below minimum requirements are marked *. It will be seen that <u>every</u> engine acceptance tested after June 1967 was below minimum required specific impulse at F.T.P.

In Reference 8, the minimum specification average I_{sp} requirement for the LLM duty cycle is quoted as 299.4 seconds. This average I_{sp} requirement is not included in the acceptance test criteria for the engine. In fact, if a LLM-5 M.D.C. analysis is carried out using the minimum acceptable values of I_{sp} and T_{120}, an average I_{sp} of 298.8 seconds is predicted. The one sigma (1σ) uncertainty in M.D.C. average I_{sp} for any particular engine (that is, excluding engine to engine variability) is given in Section 3.3.3 as $\pm$ 0.97 seconds. Taking an uncertainty of 1.96σ, to give a 97.5% probability that the real average I_{sp} will exceed the required minimum of 299.4 seconds, the predicted average I_{sp} must exceed a value of 299.4 + 1.96 X .97 or 301.3 seconds. Of the 14 engines whose average I_{sp} on the LLM-5 M.D.C. has been calculated, only four (4) meet or exceed this criteria of 301.3 seconds, even though three of these four engines failed to meet all of the acceptance test I_{sp} requirements. All four of these engines were acceptance tested prior to July 1967 and are marked (O.K.)

The discrepancy between the required average I_{sp} and that which could be obtained from an engine which meets the acceptance test requirements will be noted. The predicted average I_{sp} from an engine with a T_{120} time of 75 seconds and acceptance test specific impulse values of 304.0 (F.T.P.), 298.5 (50%) and 294.5 (25%) is 298.8 seconds. Taking into account the 1.96σ uncertainty of 1.9 seconds, this value is 2.5 seconds lower than the vehicle requirement of 299.4 seconds, despite the fact that the engine would have passed its acceptance tests with no waiver.

3.4.2 M.D.C. Specific Impulse Data

Six engines were used in the Phase B qualification test program. The predicted and actual values of average M.D.C. specific impulse obtained on these qual engines are listed below. The predictions were carried out using the

Victory VI program, which has since been updated to the Victory VIIa (Reference 6, 7). The predictions for the LLM-5 M.D.C., which is a different duty cycle from that to which the qualification engines were tested, were made by a more rigorous program than the Victory VI, which accounts for the slight differences in predicted average I_{sp}.

Engine #	Average I_{sp} on M.D.C. (Seconds) Predicted	Actual	Difference (Actual - Predicted)
1025 (qual 4)	300.5	301.0	+.5
1034 (qual 5)	301.3	300.6	-.7
1024 (qual 6)	300.8	300.8	0
1023 (qual 7)	298.9	298.6	-.3
1025(c)(qual 8)	299.5	299.2	-.3
1038 (qual 9)	300.4	300.7	+.3

Standard Deviation $\pm$.45 seconds

3.4.3 Performance on Recent Engines

It was noted in Section 3.4.1 that every engine acceptance tested during the last year was only accepted with a waiver on F.T.P. specific impulse: in the case of one engine (1042) the specific impulse was 1.4 seconds low. TRW have been actively investigating the reasons for the apparent loss of some 0.7 seconds in specific impulse at F.T.P. which has been occurring during the last year (References 10, 12).

The biggest single cause identified has been the ballistic calibration of the engine flow meters, instead of the weigh tank method previously employed. A description of these two calibration systems is given in Reference 1. Presently, ballistic calibration of the flow meters is not used and the old method of weigh tank calibration is now employed. In Reference 12, TRW accounts for approximately half of the observed reduction in specific impulse to be due to the change in flow meter calibration methods. It is not known which of these two methods gives the more accurate results. TRW accounts for a change of 0.2 seconds in specific impulse because of a systematic change in fuel specific gravity.

A further performance anomaly is the apparent 1% (50 ft./sec.) reduction in F.T.P. characteristic velocity observed on H.E.A. tests in the V.E.T.S. facility. This reduction in performance, if true, would be expected to result in a reduction of some 3 seconds in specific impulse, which has not been observed. TRW has investigated, and eliminated, the following sources of the C* reduction (Reference 12):

Chamber pressure measurement
Flow measurement
Data reduction
Propellant composition
Propellant leakage
Injector hardware
V.E.T.S. hardware

Presently. TRW has not resolved the reasons for this apparent reduction in C*, despite a thorough investigation which is still being carried out. This investigation includes the sampling of propellants during a test, to see if there is any nitrogen entrained or in solution.

3.5 Summary

3.5.1 The engine acceptance test criteria for specific impulse, unlike the criteria for thrust and mixture ratio, do not require the 3σ measurement uncertainty to be added or subtracted from the test data.

3.5.2 The engine acceptance test criteria for specific impulse and throat erosion (T_{120} time) do not guarantee the requirement for 299.4 seconds average specific impulse over the LLM duty cycle. An acceptable engine could have a true average specific impulse which was 2.5 seconds low.

3.5.3 Due to throat erosion, the specific impulse of an engine may degrade by some 5 seconds during full LLM duty cycle.

3.5.4 At F.T.P. setting, the engine's mixture ratio is strongly affected by differences in engine interface pressures, a difference of 2 psi causing a shift of some 1% in mixture ratio.

3.5.5 Operating conditions have a strong effect upon throat erosion rate. For example, a decrease in propellant temperature from 90 to 50° F doubles the erosion rate and an increase in mixture ratio from 1.6 to 1.7 causes the erosion rate to be increased by a factor of 4.

3.5.6 Taking into account measurement, run to run, prediction and operating environment uncertainties, the 1σ uncertainty in prediction of average specific impulse for any one engine is approximately $\pm$ 1.0 seconds.

3.5.7 Every engine acceptance tested after June 1967 had a specific impulse at F.T.P. which was below the minimum acceptance test criteria value (304.0 seconds).

3.5.8 Out of 14 production configuration engines, only four have a predicted average I_{sp} which is adequate to satisfy the LLM-5 requirement of 299.4 seconds with 97.5% probability.

3.5.9 During the last year, there has been an apparent reduction of 0.7 seconds in engine specific impulse. The reasons for this reduction are being actively investigated by TRW, but have not yet been fully resolved.

3.5.10 More recently (during the last few months) there has been a reduction of approximately 1% in the characteristic velocity measured on the head end assembly tests, which has not been confirmed by a similar reduction in specific impulse when these head end assemblies are fired in the engine test stand. This anomaly is presently being investigated by TRW, but has not yet been resolved.

3.6 References

1) D2-117060-1 "Apollo Spacecraft Engine Specific Impulse". J.P.B. Cuffe. May 6, 1968.

2) TRW Report 01827-6070-T000. 18 April 1967, revised 18 July 1967 "TRW LM Descent Engine End Item Acceptance Test Plan and End Item Test Inspection Plan".

3) TRW Report 01827-6115-T000. 11 October 1967. "TRW LM Descent Engine Phase B Qualification Test Program Summary Report".

4) TRW Report 01827-6119-T000. 10 November 1967. "Characteristics of the TRW Lunar Module Descent Engine". Volumes 1 through 4.

5) TRW Note 4721.3.67-91. 19 April 1967. "Instrumentation Uncertainty of LMD Vacuum Specific Impulse"

6) TRW Note 4721.3.67-284. 6 December 1967. "Summary of Methods Used to Predict LMDE Duty Cycle Performance, for Engines Equipped for Shallow Throttling (Victory VII)".

7) TRW Note 4721.3.68-36. 6 February 1968. "Victory VIIa Revision to the Victory VII LMDE Flight Simulation Program".

8) TRW Note 68.4711.8-109. 1 May 1968. "DPS Mission Duty Cycle Average Specific Impulse for the LLM".

9) TRW Acceptance Test Reports for the Following LM Descent Engines: #1015, 1020, 1021, 1023, 1024, 1025, 1026, 1028, 1030, 1034, 1036, 1037, 1038, 1039 and 1042.

10) Trip Report to TRW, Redondo Beach, California. J.P. B. Cuffe. June 4, 1968.

11) TRW LM Descent Engine Program Review for NASA/GAEC. 26 February 1968.

12) TRW Presentation at GAEC/TRW Technical Interchange at GAEC. May 8 - 9, 1968.

13) TRW Note 01827-6002-R000. 31 August 1966. "LEMDE Instrumentation Error Analysis".

14) TRW Note 4721.3.68-212. 18 July 1968. "Review of Boeing Document No. D2-117060-2". "Apollo Spacecraft Engine Specific Impulse - Part 2".

TABLE 1

Specific Impulse Influence Coefficients

Changes in Vacuum Specific Impulse: Seconds

Engine Throttle Position	10%	20%	30%	40%	50%	60%	F.T.P.
1% change in throat area	-0.26	-0.26	-0.26	-0.26	-0.26	-0.26	-0.26
1 psi change in oxidizer inlet pressure	-0.226	-0.196	-0.139	0.0559	0.0545	0.192	-0.0388
1 psi change in fuel inlet pressure	0.112	0.136	0.123	0.0711	-0.0187	-0.147	0.0358
1° F change in oxidizer inlet temperature	-0.994	-0.375	0.0246	0.202	0.156	-0.116	-0.0722
1° F change in fuel inlet temperature	1.05	0.374	-0.0776	-0.303	-0.301	-0.0686	0.0755

Thrust Influence Coefficients

Change in Vacuum Thrust: lb.

Engine Throttle Position	10%	20%	30%	40%	50%	60%	F.T.P.
1% change in throat area	-0.935	-1.86	-2.77	-3.66	-4.55	-5.42	19.1
1 psi change in oxidizer inlet pressure	0.671	1.56	2.96	5.16	8.44	13.1	19.5
1 psi change in fuel inlet pressure	1.26	2.70	3.92	4.51	4.09	2.24	8.61
1° F change in oxidizer inlet temperature	-4.27	-4.18	-2.05	-0.248	-1.14	-7.15	-4.14
1° F change in fuel inlet temperature	3.53	2.31	-1.31	-4.92	-6.11	-2.43	2.17

Mixture Ratio Influence Coefficients

Change in Mixture Ratio

Engine Throttle Position	10%	20%	30%	40%	50%	60%	F.T.P.
1% change in throat area	0	0	0	0	0	0	0
1 psi change in oxidizer inlet pressure	0.00364	0.00365	0.00366	0.00368	0.00371	0.00374	0.00751
1 psi change in fuel inlet pressure	-0.00344	-0.00345	-0.00346	-0.00348	-0.00350	-0.00353	-0.00710
1°F change in oxidizer inlet temperature	-0.00190	-0.00191	-0.00191	-0.00192	-0.00193	-0.00194	-0.00192
1°F change in fuel inlet temperature	0.000641	0.000642	0.000642	0.000643	0.000645	0.000646	0.00172

UNCERTAINTY OF LMD ENGINE MEASURED VACUUM SPECIFIC IMPULSE

DATA FROM TABLE 2 OF TRW NOTE #4721.3.67-91.

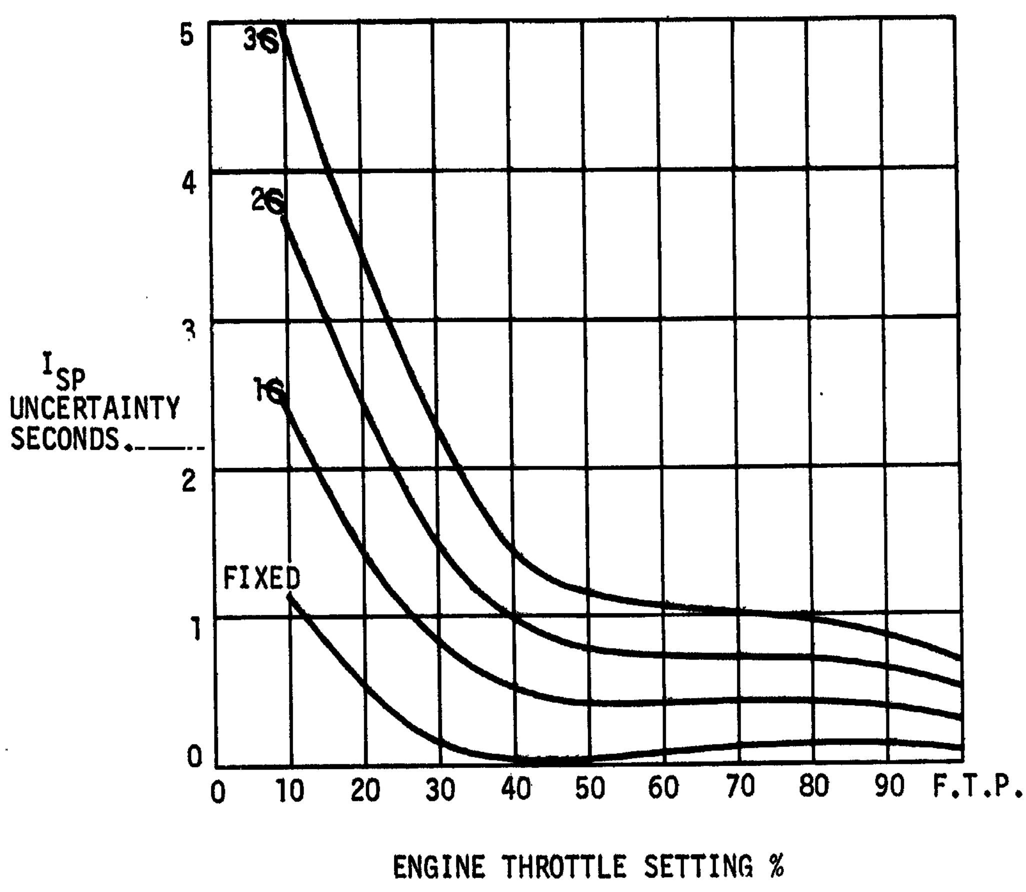


FIGURE 1

11

MIXTURE RATIO LIMITS

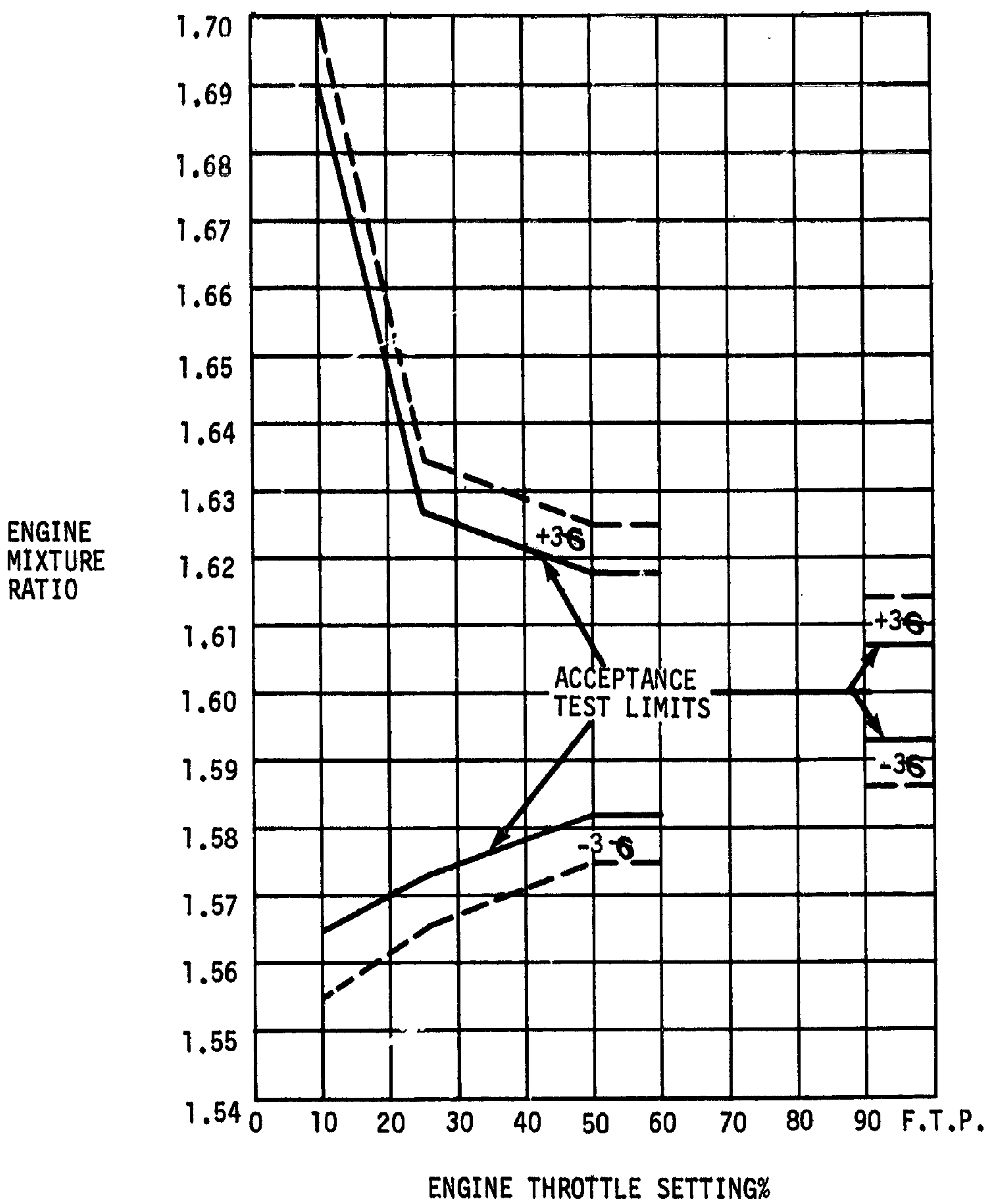


FIGURE 2

SPECIFIC IMPULSE INFLUENCE COEFFICIENTS

(ALL DATA FROM TABLE 3.4 OF TRW #01827 - 6119 - T000 VOLUME 1)

UNIT CHANGE IN INDEPENDENT VARIABLE - CHANGE IN I_{SP}.

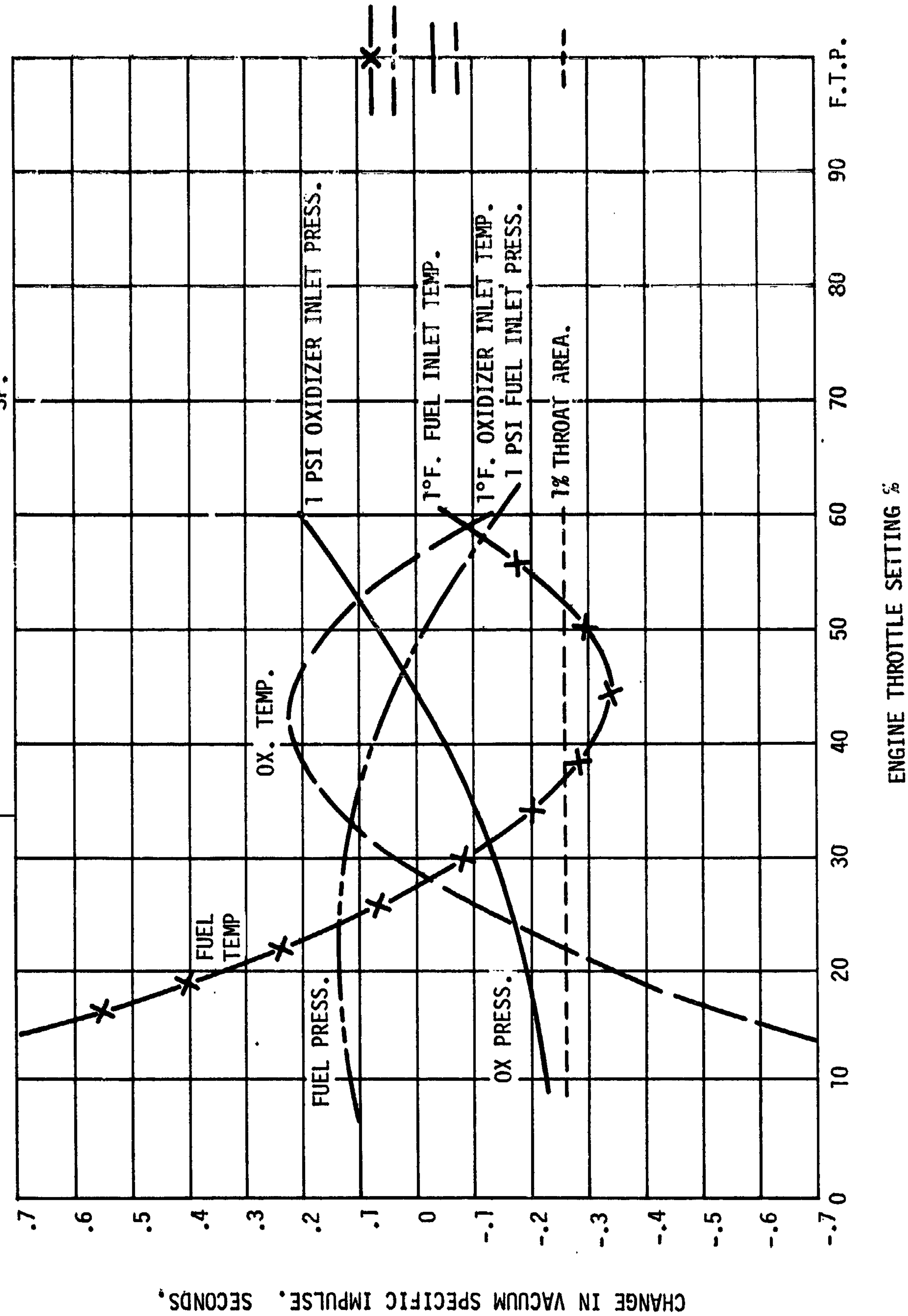


FIGURE 3

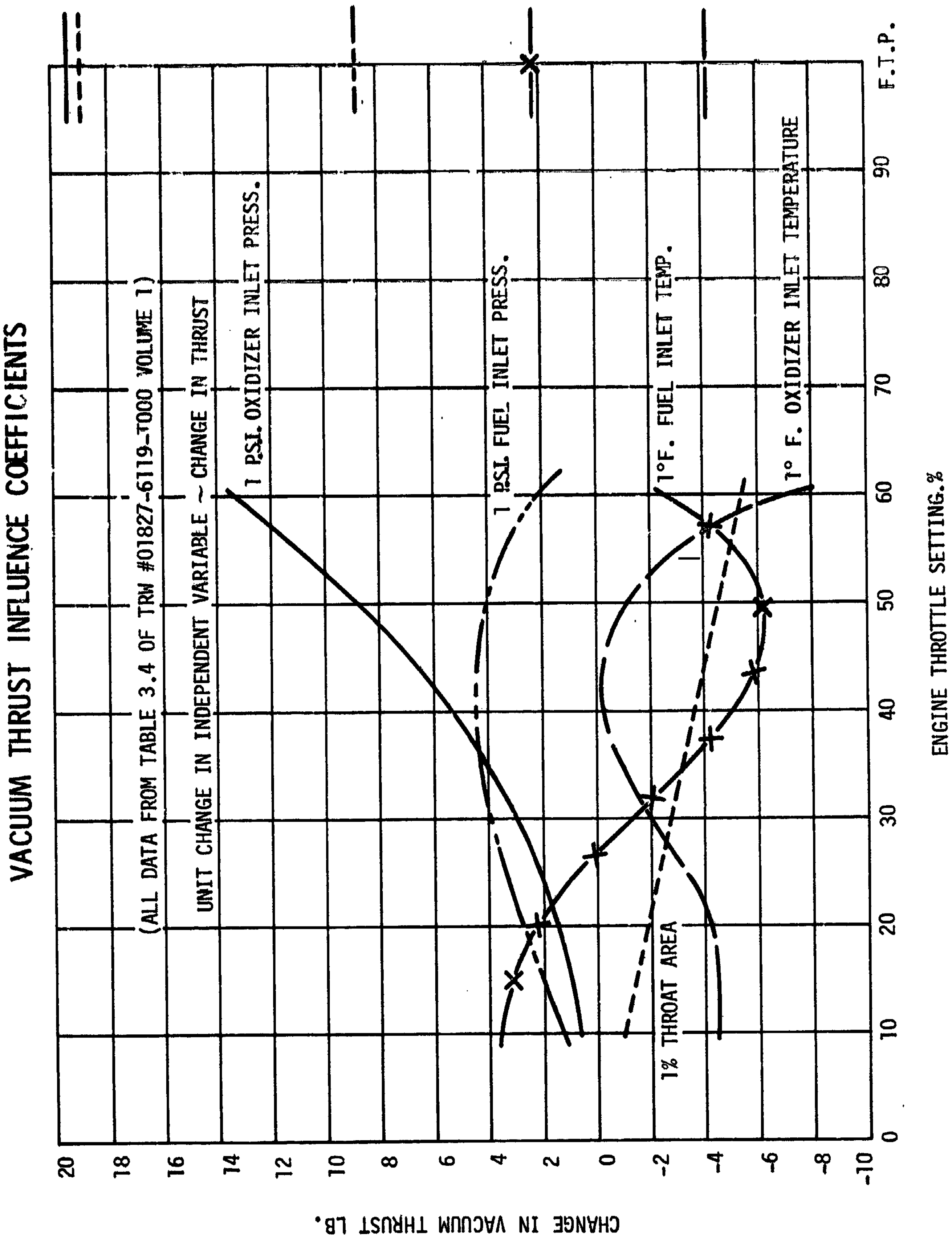

VACUUM THRUST INFLUENCE COEFFICIENTS
(ALL DATA FROM TABLE 3.4 OF TRW #01827-6119-T000 VOLUME 1)
UNIT CHANGE IN INDEPENDENT VARIABLE ~ CHANGE IN THRUST
1 PSI. OXIDIZER INLET PRESS.
1 PSI. FUEL INLET PRESS.
1°F. FUEL INLET TEMP.
1° F. OXIDIZER INLET TEMPERATURE
1% THROAT AREA
20
18
16
14
12
10
8
6
4
2
0
-2
-4
-6
-8
-10
CHANGE IN VACUUM THRUST LB.
0
10
20
30
40
50
60
70
80
90
F.T.P.
ENGINE THROTTLE SETTING, %

FIGURE 4

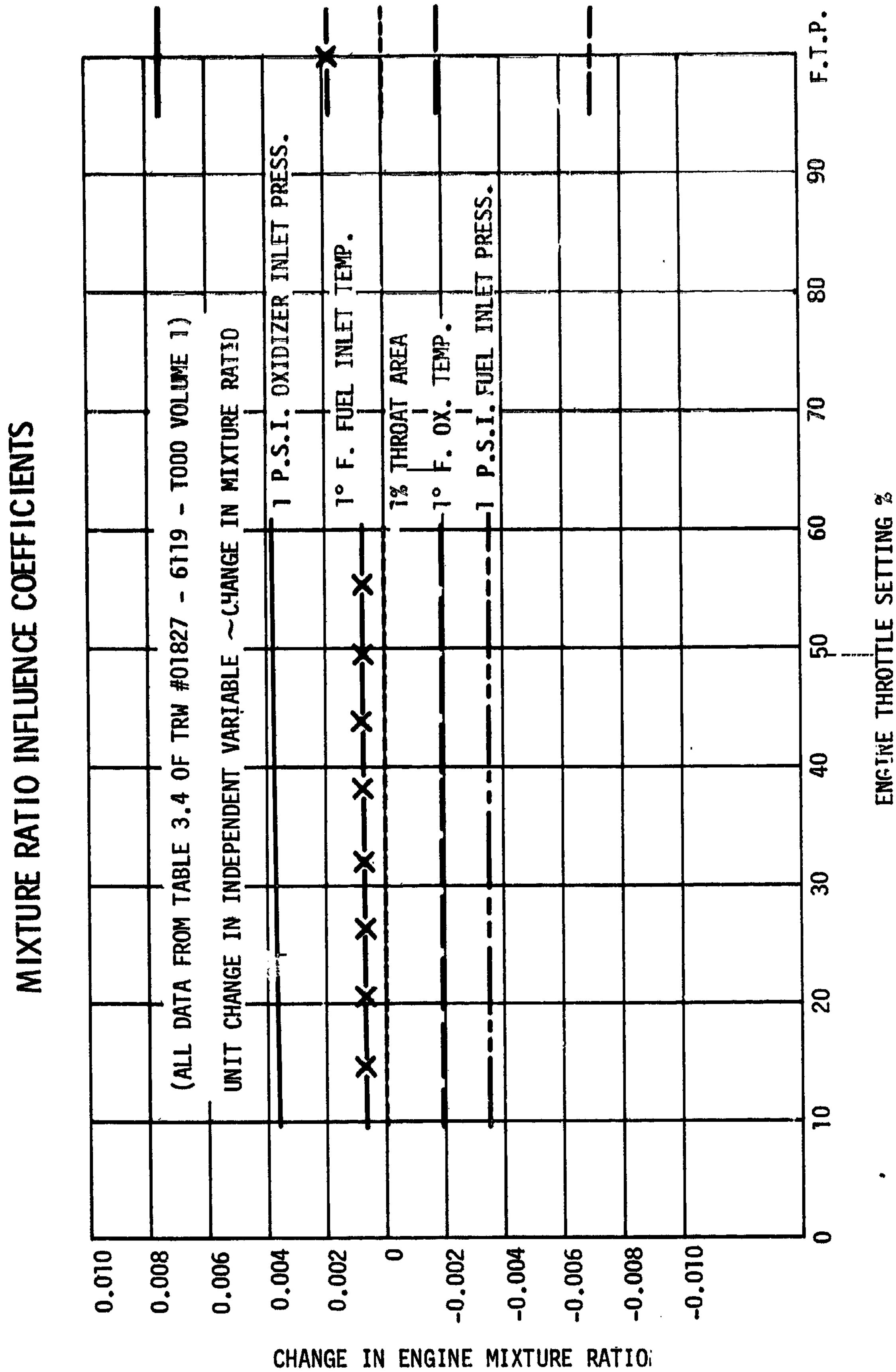

MIXTURE RATIO INFLUENCE COEFFICIENTS
(ALL DATA FROM TABLE 3.4 OF TRW #01827 - 6119 - TODD VOLUME 1)
UNIT CHANGE IN INDEPENDENT VARIABLE ~CHANGE IN MIXTURE RATIO
1 P.S.I. OXIDIZER INLET PRESS.
1° F. FUEL INLET TEMP.
1% THROAT AREA
1° F. OX. TEMP.
1 P.S.I. FUEL INLET PRESS.
0.010
0.008
0.006
0.004
0.002
0
-0.002
-0.004
-0.006
-0.008
-0.010
CHANGE IN ENGINE MIXTURE RATIO
0
10
20
30
40
50
60
70
80
90
F.T.P.
ENGINE THROTTLE SETTING %

FIGURE 5

4.0 Aerojet SPS Engine (September 1968)

4.1 Introduction

Unlike the manufacturers of the other engines used in the spacecraft primary propulsion systems (TRW, Bell and Rocketdyne), Aerojet has no altitude simulation facilities at Sacramento suitable for testing the SPS engine and therefore must predict engine specific impulse from injector performance data.

A summary of injector and engine acceptance test procedures and requirements is given in Section 4.2. Performance analysis is discussed in Section 4.3 and available results from production engines are presented in Section 4.4. A comparison between Aerojet SPS and Rocketdyne LM Ascent engine injector performance variability is included in Section 4.5, together with an evaluation of the Aerojet method of determining altitude specific impulse when applied to other spacecraft engines.

4.2 Acceptance Test Procedures and Requirements

Full details of the acceptance test procedures and requirements are given in Reference 3 (Injector tests) and Reference 4 (Engine tests).

Because Aerojet does not have the facilities for firing the SPS engine under simulated altitude conditions, engine specific impulse performance is derived indirectly from injector test data, using an empirical correlation factor obtained from the Block II engine qualification test program run at AEDC Tullahoma (References 5 through 7). Specific impulse for production engines is therefore determined on their injector acceptance tests (4.2.1) but thrust and mixture ratio are obtained from the results of the engine acceptance tests (4.2.3).

4.2.1 Injector Acceptance Tests

The injector is fitted to an uncooled steel chamber of 1.5:1 area ratio. All flight injectors so far tested have been fired in the same steel chamber, #004. These tests are carried out under local ambient pressure conditions. There is provision for temperature conditioning the propellants but not for helium saturating them. Propellant tanks are pressurized with nitrogen.

These tests have to be carried out under the following conditions, where 'Old' refers to the requirements for the injectors which have been tested to date and 'New' defines the conditions for injectors which will be tested in the future.

Required Test Conditions	Old	New
Number of tests	4 (Minimum)	Same
Test Duration, seconds	$5 \, {}^{+2}_{-0}$	Same
Chamber pressure, psia	97 ± 3	99 ± 3
Mixture ratio, o/f	1.6 ± 0.02	1.6 ± 0.05
Chamber temperature above ambient, °F	50 (Maximum)	Same
Propellant inlet temperature, °F	70 ± 30°	70 ± 10°
Difference between propellant temperatures, °F	10° (Maximum)	Same
Data summary period from start command, seconds	2.5 to 4.4	Same
Chamber cooling time between tests, minutes.	Not Specified	10

Nominal Injector Operating Conditions

The measured (site) values of C* are corrected to the following standard conditions: 99 psia chamber pressure

1.6:1 mixture ratio

70°F propellant temperature

This correction is carried out by means of a covariance equation (see Figure 1) which is derived from empirical data obtained on a Mod O injector. It will be seen that using the old propellant temperature limits, the C* correction for temperature could be as much as +48 ft/sec (2 1/2 seconds I_{sp}), where as with the new limits, this correction will be less than 13 ft/sec. On the other hand, increasing the allowable mixture ratio range from ± 0.02 to ± 0.05 enables additional data to be used, which would otherwise be rejected and only increases the possible C* correction, due to mixture ratio, from 6 to 13 ft/sec. These corrected values of C* are then averaged: if the C* on any test differs from the average by more than 19 ft/sec, then this test is ignored and a new average calculated. Nominal altitude I_{sp} is simply obtained by multiplying the average C* by 0.05338, which is the empirical value of F (see Reference 1). Minimum altitude I_{sp} is defined as the nominal value of I_{sp}, less 1.5 seconds.

The Aerojet injectors are classed into three grades, using the following criteria:

- ME 270-0004-0002. Injectors whose minimum altitude I_{sp} is not less than 313.0 seconds (nominal I_{sp} not less than 314.5 seconds). Engines using these injectors are planned for use on the first two lunar landing missions.

- ME 270-0004-0003. Injectors whose minimum altitude I_{sp} is not less than 311.0 seconds (nominal I_{sp} not less than 312.5 seconds). These injectors will be used on later lunar missions.

- ME 270-0004-0001. Injectors whose minimum altitude I_{sp} is not less than 310.5 seconds (nominal I_{sp} not less than 312.0 seconds). These injectors will only be used on earth orbit missions.

- Injectors with a minimum altitude I_{sp} of less than 310.5 seconds (nominal I_{sp} less than 312.0 seconds) are not presently acceptable for flight purposes.

4.2.2 Injector Compatibility Tests

A chamber of 1.5:1 or 6:1 area ratio of flight type ablative material is used and one satisfactory test has to be carried out under the following conditions:

Duration	305 $\pm$ 5 seconds
Chamber Pressure	97 $\pm$ 5 psia
Mixture Ratio	1.6:1 $\pm$ 0.1 (for last 290 seconds)

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. The same chamber may be used for several compatibility tests. If an injector fails the compatibility test, it is probably due to either an error in injector drilling or else to contamination, so the injector is decontaminated, inspected, reworked if necessary and can then be resubmitted for compatibility tests.

4.2.3 Engine Tests

The injector, feed lines ~~and~~ ball valve assembly are fitted to a work horse (not flight, as was stated in Reference 1) ablative thrust chamber. On these tests, the engine feed system is calibrated to give the required thrust and mixture ratio. The firings are carried out under local ambient pressure, with the nozzle extension not fitted. The calibrating tests, where the engine is orificed to give the correct thrust and mixture ratio when firing in the upper, dual and lower bipropellant valve operation modes, are referred to as 'Balance firings'. Subsequent to the 'Balance firings', 'Acceptance firings' are carried out.

The engine is mounted in a simulated service module propulsion system for these tests. Helium is used for propellant tank pressurization, though nitrogen is allowable for the Acceptance tests. Both the 'Balance' and 'Acceptance' firings are carried out under the same test conditions.

Required Test Conditions:

Propellant Temperature	70°F ±30°F
Fuel Temperature = Oxidizer Temperature	±10°F
Propellant interface pressures (at engine start):	
Oxidizer interface pressure	178 ± 4 psia
Fuel interface pressure	178 ± 4 psia
Difference between interface pressures	<4 psia
Propellant interface pressure (steady state):	
Oxidizer interface pressure	162 ± 4 psia
Fuel interface pressure	169 ± 4 psia
Fuel interface pressure - oxidizer interface pressure	7 ± 2 psia

Sequence of Operation During Tests

'Balance' firing sequence

Engine start	open valve bank 'B'
Engine start + 4 $\left(\pm{}^{.5}_{0}\right)$ second	open valve bank 'A'
Engine start + 7 $\left(\pm{}^{.5}_{0}\right)$ seconds	close valve bank 'B'
Engine start +11 $\left(\pm\frac{1}{2}\right)$ seconds	close valve bank 'A'

'Acceptance' Firing Sequence

One 10 $\left(\pm{}^{1}_{0}\right)$ second duration test with valve banks 'A' & 'B'

One 5 $\left(\pm{}^{1}_{0}\right)$ second duration test with valve bank 'A' only

One 5 $\left(\pm{}^{1}_{0}\right)$ second duration test with valve bank 'B' only

Nominal Engine Operating Conditions

Propellant temperature	70°F
Oxidizer interface pressure	163 psia
Fuel interface pressure	169 psia
Combustion Chamber throat area	121.680 ins^2
Ambient pressure	0 psia

A nominal value of throat area has to be used, since the ablative chamber that will be used for flight is fitted after the engine has completed its acceptance tests.

Details of the methods used to normalize engine acceptance test data to nominal engine operating conditions are given in Section 4.3.3. It will be seen that thrust on engine acceptance tests is obtained from measured propellant flow rates and predicted specific impulse, which was determined on the injector acceptance tests.

Required Performance on Engine Acceptance Tests:

When engine acceptance test data are corrected to nominal engine operating conditions, the following performance requirements must be met:

Thrust. Dual bore operation, 20,500 ± 205 lb (± 1%)

Single bore operation, greater than 19,475 lb.

where "Dual bore operation" = both valve banks 'A' and 'B' open and "single bore operation" = only valve bank 'A' or 'B' open.

Mixture ratio. Dual bore operation } 1.60:1 ± 0.02 (±1 1/4%)
Single bore operation

Only one satisfactory acceptance test under each of the three required modes of operation is necessary.

Allowable Testing of Production Engines.

Engine operation, during Balance and Acceptance firings, must not exceed the following limits:

Allowable number of engine starts	10 maximum
Allowable accumulated engine run time	100 seconds maximum

4.3 Performance Analysis

4.3.1 Injector C* Performance

Injector C* is defined as: $\frac{Pc.\ At.\ g}{\dot{W}}$ ft/sec.

where Pc = Chamber pressure in psia measured at injector pressure tap.

g = Gravitational constant = 32.174 ft/sec^2

$\dot{W}$ = Total propellant flow rate lb/sec.

A_t = Throat area of steel chamber in square inches.

The throat area is measured prior to a test series and the throat area on any test in that test series is the measured value minus a correction factor. This correction factor is an empirical curve fit of the observed change in throat area with total run time on the steel chamber. From Figure 2 it will be seen that the throat area on steel chamber 004 has decreased by some 2 1/2% and it will be noted that the injectors used to determine the value of R (see Section 4.3.2) were all tested when the throat area was approximately 2% more than it is at present and was changing comparatively rapidly (1% in less than 100 seconds). All production injectors have been tested in this steel chamber. The test values of C* are then corrected to nominal operating conditions (Section 4.2.1) using the covariance equation given in Figure 1. These corrections for propellant temperature, chamber pressure and mixture ratio were obtained from performance characterization tests carried out on injector #097, which was a Mod 0 design. It will be seen that chamber pressure changes have negligible effect upon C*, but that reduction in propellant temperature from nominal has a very significant gain. Aerojet is confident that the performance characteristics of the Mod 0 and Mod IV injectors are the same, but has proposed a test program to characterise the Mod IV (Appendix C). Some of the changes from the Mod 0 to the Mod IV injector design, which might affect the injectors' characterization, are:

- Percentage of film coolant: from 6.6 to 5.3%.
- All oxidizer orifices counter-bored to reduce L/D to $<$1.0
- All fuel orifices enlarged; some fuel orifices counter-bored.
- Depth of radial baffles at tip extended from 2 to 4 inches.

4.3.2 Injector to Engine Performance Correlation

Because Aerojet does 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:

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

or

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

where: $I_{sp\ (62.5:1\ vacuum)}$ = 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.

A summary of the data obtained on the Block II engine tests at AEDC is given in Table 1 on page 76.

It will be seen that there are changes in both C* and I_{sp} values in the data extracted from Reference 5, the Aerojet Block II engine performance analysis report, and that obtained from Reference 8, the Aerojet performance presentation at MSC in February 1968. The differences in I_{sp} (0.1 sec) are the result of the change in the nominal chamber pressure from 97 to 99 psia (see Figure 5). The differences in C* are primarily due to the use of the normalized curve fit for the throat area of the steel chamber (Aerojet explaination given at MSC presentation February 1968). The revised C* values not only reduce the variability of the R factor by a factor of two and a half, but also increase the numerical value of R, presumably because of the reduction in throat area obtained on successive tests (Figure 2). The effect of the increase in R is to raise the predicted value of I_{sp} for a production injector by 0.35 seconds, 0.25 seconds resulting from the revised C* values and 0.1 seconds because of the increase in nominal thrust chamber pressure to 99 psia.

The tests using injectors 104, 115 and 103 were also analysed by TRW (Ref. 5) and ARO (Ref. 6). Their normalized specific impulse values have been adjusted to the revised nominal chamber pressure of 99 psia. A comparison of the Aerojet, TRW and ARO values of engine specific impulse, plotted against injector C*, is shown in Figure 3. It may be seen that for two of the three injectors analysed by ARO and TRW, the data points lie on or outside of the Aerojet 3σ R factor limits. The average value of specific impulse for injectors 104, 115 and 103 is also lower than that given by Aerojet: 0.7 seconds and 0.5 seconds lower from the TRW and ARO calculations respectively.

The R factor is calculated on the assumption that engine performance is only determined by its injector and that the effect of change in thrust chamber can be ignored. Figure 4 shows the Aerojet values of I_{sp} (Reference 5, corrected to 99 psia chamber pressure) for each thrust chamber plotted against injector C*. It will be seen that two of the four injectors are within the $\pm$ 3σ R factor limits, one is on the $\pm$ 3σ R factor limit and one (injector #115) is well outside of these limits. If R is calculated for each thrust chamber/injector combination, then its' value is not changed, but its standard deviation is increased by a factor of three and a half to $\pm$ 0.14%.

4.3.3 <u>Engine Acceptance Test Performance</u>

Engine thrust and mixture ratio values are obtained from the engine acceptance tests (Section 4.2.3). Feed system (engine interface to chamber pressure) hydraulic resistances are calculated for fuel and oxidizer systems from test data. Since the C* for the injector is also known, flow rates and chamber pressure at nominal interface conditions are calculated by assuming that the feed system hydraulic resistances remain constant.

I_{sp} at nominal thrust chamber operating conditions is calculated directly from the injector C* (Section 4.3.1). Knowing the engine mixture ratio and chamber pressure at nominal interface conditions, the I_{sp} correction is obtained from Figure 5. Thrust is then calculated from these corrected values of flow rates and I_{sp}.

Further corrections (obtained from Reference 4) are necessary to normalize performance data when the engine is run under off-nominal conditions. The effect of oxidizer temperature upon system resistance shows negligible change between 30 and 70°F, but a 1% increase in resistance occurs when the temperature is increased from 70 to 100°F. The effect of propellant flow rate upon feed system resistance is also characterized in Reference 4, and the approximate effects are:

Oxidizer system.	1% increase in flow rate results in 0.08% reduction in feed system hydraulic resistance
Fuel system.	1% increase in flow rate results in 0.07% reduction is feed system hydraulic resistance.

4.3.4 Specific Impulse Uncertainty Analysis:

The 1σ 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 does not use these values of random measurement uncertainty, but allocate as a best guess a possible nonrandom error in the AEDC data of 0.2 seconds in I_{sp} vacuum.

The following expression for calculating the 3σ $I_{sp\ vacuum}$ uncertainty was given by Aerojet at their MSC presentation (Reference 8) and again at Sacramento (Reference 2):

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

where: $R = 0.05338\ \text{sec}^2/\text{ft}$

$B = 1\sigma\ C^*_{inj}$ 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 = 1σ deviation of R = 0.0004

C = AEDC uncertainty = 0.2 seconds

Using the above values, the uncertainty is $\pm$ 1.1 seconds. However, Aerojet have been directed to use an uncertainty value of 1.5 seconds.

The Aerojet estimate of a 3σ I_{sp} uncertainty of 1.1 seconds is lower than the values given by both TRW and Bell for their LM engines: the Rocketdyne uncertainty value being expressed as a 95% confidence limit, is not directly comparable. Some of the reasons for the low value of I_{sp} uncertainty given by the Aerojet analysis include:

- No allowance is made for errors resulting from the normalization of C^*_{inj} test data. Characterization data obtained from an injector of different design are used to normalize C^*_{inj} test data. The correction in predicted I_{sp} resulting from this normalization can amount to 2 1/2 seconds.

• It is assumed that there is no change in injector performance on acceptance tests (use of 1/3 in first term): see Figure 6.

• Engine specific impulse is assumed to be determined only by injector C*: if the variability of R is determined on the basis of each injector/thrust chamber assembly used in the Block II qualification program at AEDC, then the value of A is increased to 0.0014 (Table 1) and the I_{sp} uncertainty, due only to the variability of R, exceeds $\pm$ 1.1 seconds. i.e.: 3σ uncertainty in I_{sp}, due to variability to R,

$$= \pm 3 \; (C^*_{inj} \times R \times A)$$

$$= \pm 1.32 \text{ seconds}$$

• A very small value (0.2 seconds) is used to account for possible errors in the AEDC data. It is not known how this number was determined.

• No allowance is made for errors resulting for the normalization of the AEDC test data. These tests were run over a very wide range of engine mixture ratio, chamber pressure and propellant temperature, so the magnitude of the normalization corrections were considerable. When these tests were analysed by different organizations, corrected mean values of specific impulse for a particular injector could differ by as much as 0.9 seconds (Table 1).

4.4 Results

4.4.1 Production Engine Performance

Flight injector performance data, obtained from Reference 2, are plotted on Figure 6 and are attached in Appendix A. These data are summarized in the following table:

Injector	Engine	Spacecraft (Note)	Injector C* ft/sec	I_{sp} sec. Nominal	Minimum
099	56	?	5869	313.3	311.8
100	57	103	5884	314.1	312.6
101	52	106	5896	314.7	313.2
120	62	104	5863	313.0	311.5
121	63	?	5878	313.8	312.3
122	61	107	5888	314.3	312.3
124	60	?	5902	315.0	313.5
127	58	103	5852	312.4	310.9
129	59	101	5855	312.5	311.0
		Average Values	5876	313.7	312.2

Note: Spacecraft Engine Allocation as of 8-9-68.

The injector-to-injector C* standard deviation is $\pm$ 18 ft/sec (0.3%). The run-to-run C* standard deviation for each injector is shown on Figure 6. The RMS of these deviations is $\pm$ 10.6 ft/sec if all data points are included, but is reduced to $\pm$ 8.5 ft/sec if the last data point on injector #101 is ignored (because it differs by more than 19 ft/sec from the mean value of C* for this injector). The apparent trend to reduction in C* with successive tests is well shown in Figure 6. If it is assumed that there is no real shift in injector combustion efficiency, then it is possible that the throat area of the steel chamber expands on successive firings and does not contract in the manner shown in Figure 2.

Using the criteria defined in Section 4.2.1, the nine production injectors can be classed as follows:

-0002 (Initial Lunar)	-0003 (Nominal Lunar)	-0001 (Earth Orbit Only)
Inj 101 in S/C 106	Inj 099	Inj 127 in S/C 103
Inj 124	Inj 100 in S/C 103	Inj 129 in S/C 101
	Inj 120 in S/C 104	
	Inj 121	
	Inj 122 in S/C 107	

4.4.2 Mission Duty Cycle Data

As was explained in Reference 1, no usable MDC I_{sp} data are available from the Block II qualification test program run at AEDC because the thrust measuring system in J-3 cell gave suspect results after approximately 40 seconds of engine running.

The TRW analysis of the qualification test program (Reference 7) showed that throat area decreased during the first 300 seconds of engine operation by some 0.7% and then started to slowly erode. However, there was considerable variation between the different chambers, as is shown in the following table:

Test Series	EA	EB	EC	ED	EE	EF
Thrust Chamber	311	315	313	320	318	324
% Change in A_t	-.02%	+.53%	+.16%	-1.2%	-0.65%	
% Change in Weight		-10.5%	-8%		-9%	
Total firing time sec	754	753	755	754	755	753
Propellant temperature	Nom	Cool	Hot	Nom	Warm	Cold
Chamber pressure	Nom	Hi	Hi	Low	Nom	Nom
Mixture ratio	Nom	Nom	Nom	Nom	Low	Nom

As might be expected, it appears as if the erosion rate increases with increase in chamber pressure, but that under nominal conditions, the throat area after 750 seconds should differ little from its initial value. The reduction in throat area which occurs during the first part of the MDC should give a slight increase in specific impulse. Since the effect of run time upon nozzle efficiency is not known, it is suggested that the I_{sp} increase due to throat area reduction may be assumed to compensate for any reduction in nozzle efficiency which might occur later on in the duty cycle.

The effect upon engine performance of using helium saturated propellants is not known, since no performance tests have been carried out with propellants in this condition.

4.5 $C^*_{inj} \sim I_{sp}$ Correlation

To investigate the Aerojet method of determining engine specific impulse only on the basis of injector C* performance, an analysis was made of data obtained on acceptance tests of Rocketdyne LM Ascent production engines. These data are tabulated in Appendix B and include:

- Injector serial number
- C* obtained on Bravo IVA tests, mean value of C* and run-to-run standard deviation of C*. Standard performance (corrected to nominal interface conditions) values of C* are used.
- I_{sp} obtained on engine altitude acceptance tests at NFL and mean value of I_{sp}. Again, standard performance values are used.
- $R = \dfrac{\bar{C}^*}{\bar{I}_{sp}}$, the Aerojet empirical correlation factor between injector C* performance and engine I_{sp}.

4.5.1 C* Comparison

Figure 7 shows C* values plotted against injector serial number for 20 production Rocketdyne LM Ascent engine injectors. These data may be compared with the corresponding plot of Aerojet SM injector C* values shown in Figure 6, the same scale for C* being used in both cases. It will be seen that there is much less scatter in run-to-run data on the Rocketdyne injector, with no tendency to an apparent reduction in C* on successive tests. The Rocketdyne data also show a smaller injector to injector variability. The C* variability of the Aerojet and Rocketdyne injectors is compared in the following table:

	Aerojet	Rocketdyne	$\dfrac{\text{Aerojet}}{\text{Rocketdyne}}$
Number of production Injectors considered	9	20	
Mean value of C* ft/sec ($\bar{C}^*$)	5876	5658.5	
1σ $\bar{C}^*$ ft/sec (Injector to injector)	$\pm$18 ft/sec	$\pm$7 ft/sec	2.57

	Aerojet	Rocketdyne	Aerojet / Rocketdyne
RMS of 1 σ values	±10.6 ft/sec(1)	±1.6 ft/sec	6.6(2)
of run-to-run C* variability	±8.5 ft/sec(2)		5.3(2)

Note: (1) All test data on injector #101 included.
(2) Last test on injector #101 excluded.

To summarize, the injector-to-injector variability of the Aerojet engine is some two and a half times that of the Rocketdyne one and the run-to-run variability is approximately five times as much. The fact that the Rocketdyne injector orifices are EDM (Electro Discharge Machined), whereas the Aerojet ones are drilled, may account for some of the differences in variability. Also, the Rocketdyne injector tests are carried out under a smaller range of test conditions (Section 2.2.2.1) than the Aerojet ones (Section 4.2.1) and are of longer duration, allowing more time for conditions to stabilize before performance data are taken. In addition, the Aerojet Mod IV injector has a greater variability, both injector-to-injector and run-to-run, than any of the other Block II candidate injectors. For example, if the injector-to-injector variability of the Mod II and Mod IV designs are compared, the C* variability of the Mod IV is twice that of the Mod II and the I_{sp} variability is three times as great (Reference 5).

4.5.2 R Factor Comparison

Figure 8 shows C^*_{inj} (Bravo tests) plotted against engine I_{sp} (NFL tests) for 15 production Rocketdyne LM Ascent engines. Figure 8 may be compared with the Aerojet data plotted on Figures 3 and 4, the same scales being used on all three graphs. Despite the fact that the Rocketdyne injectors have less run-to-run and injector-to-injector variability and that more injectors are included in their R factor calculation, the variability of the Rocketdyne R factor is much greater. The R factor variability of the Aerojet and Rocket engines is compared in the following table:

	Aerojet	Rocketdyne	Aerojet / Rocketdyne
Number of injectors	4	15	
Number of engines	8	15	
1σ variability in R factor %	±0.04% (1)	±0.215%	0.19
	±0.1 % (2)		0.46
	±0.14% (3)		0.65

Note: (1) R factor determined using revised C* values (Reference 8) and mean specific impulse for each injector

(2) R factor determined using initial C* values (Reference 5) and mean specific impulse for each injector.

(3) R factor determined using revised C* values and and mean specific impulse for each engine (Figure 4)

The Rocketdyne R factor variability is more than five times as great as that claimed by Aerojet and has a 3σ value of ± 0.645% or ± 2 seconds in I_{sp}.

4.5.3 TRW LMD Engine Experience

As was described in Section 3.4 of Reference 1, TRW make a 'Best Estimate' of an engines' specific impulse by combining the measured specific impulse, obtained on engine test, with a predicted value calculated from C* measurements obtained on the HEA (head end assembly) tests.

As might be expected on a throttled engine, the correlation between HEA C* and engine specific impulse gets worse as the engine's thrust level is reduced. However, as the following table shows, at FTP the correlation between HEA C* and engine specific impulse is similar to that obtained on the Rocketdyne LM Ascent engine.

	Engine Throttle Setting		
	FTP	50%	25%
1σ variability in correlation between HEA C* and engine I_{sp}	$\pm 0.227\%$	± 0.52	$\pm 0.57\%$

The above information was obtained from Appendix A of Reference 9 and represents data obtained on 17 engine and HEA acceptance tests.

Recently, there has been a decrease in HEA C* which is not reflected in engine specific impulse and TRW are now not merging C* and engine data to obtain a 'Best Estimate' value of specific impulse (see Section 3.4.3).

4.5.4 Aerojet Proposed Performance Characterization Test Program

A summary of this proposed program, extracted from Reference 2, is attached as Appendix C.

This program is intended to characterize the Mod IV injector and to confirm the value of R factor presently used. If this program is carried out, the calculated I_{sp} uncertainty will probably increase, since it is most unlikely, based on Rocketdyne and TRW experience, that the variability of R will remain as low as 0.04%.

4.6 Summary

4.6.1 Aerojet-General does not have the facilities for testing production engines under simulated altitude conditions.

4.6.2 The predicted specific impulse for production engines is obtained from their injector acceptance test C* performance multiplied by an expirically obtained conversion constant (R factor). The value of R was obtained from tests on eight Block II engines, using four Mod IV injectors, in J-3 cell at AEDC, Tullahoma.

4.6.3 The normalization of injector C* performance is carried out using characterization data obtained from a Mod 0 design injector. Production Block II engines use the Mod IV design injector. Aerojet has proposed a performance test program to characterize the Mod IV design.

4.6.4 The claimed 3σ I_{sp} uncertainty is $\pm$ 1.1 seconds. Aerojet have been directed by MSC through N.A.R. to use a 3σ I_{sp} uncertainty of $\pm$ 1.5 seconds. It is the opinion of the writer that the I_{sp} uncertainty of the Aerojet SPS engine is not presently known.

4.6.5 Aerojet has proposed a performance test program to be run under altitude conditions in J-3 test cell, AEDC, Tullahoma, to confirm the value of 'R' factor presently being used.

4.6 Nine production Block II SPS injectors have been acceptance tested. Based on their predicted values of specific impulse, these engines are classed as follows:

Class	Use	Nominal I_{sp}	Number of Engines
-0002	Initial Lunar Mission	$\geq$314.5 seconds	2
-0003	Nominal Lunar Mission	$\geq$312.5 seconds	5
-0001	Earth Orbit Only	$\geq$312.0	2

4.6.7 The integrated mean specific impulse, during a nominal mission duty cycle, should not be less than that obtained at the start of the MDC, if the unknown effects of helium saturation of the propellants are ignored.

4.6.8 The injector-to-injector C* variability of the Aerojet SPS engine is more than twice that of the Rocketdyne LM Ascent engine and the run-to-run variability is five times as great. However, the variability of the Rocketdyne correlation between injector C* and engine I_{sp} is five times that obtained by Aerojet.

4.6.9 Until recently, the variability of the correlation between HEA C* and engine I_{sp} at FTP for the TRW LM Descent engine was similar to that obtained by Rocketdyne. However, TRW is presently unable to obtain a satisfactory correlation between HEA C* and engine I_{sp} performance.

4.7 References

1) D2-117060-1 "Apollo Spacecraft Engine Specific Impulse" J. P. B. Cuffe, May 6, 1968.

2) Aerojet Presentation at Sacramento "Procedure for establishing Apollo SPS performance" 5th September 1968.

3) Aerojet Specification AGC-46847A. 12 September 1967, Updated through Change #5 (6 June 1968), Acceptance Test, Injector Assembly AJ 10-137.

4) Aerojet Specification AGC-46846C, 6 July 1968, Updated through Change #4 (23 August 1968), AJ 10-137 (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 (AJ 10-137).

7) TRW report 05952-H220-R000, June 1967, "Final Performance Characterization for the SPS Block II Engine".

8) Aerojet Presentation at MSC, Houston "Engine Analysis" 13 February 1968.

9) TRW LM Descent Engine #1037 "Acceptance test performance report". 29 June 1957, report #01827-6098-T000.

10) Trip report to Aerojet-General, Sacramento, California J. P. B. Cuffe, 10 September 1968.

BLOCK II ENGINE TEST AT AEDC DATA SUMMARY

Test Series	DH	DI	EA	EB	EC	ED	EE	EF	MEAN	1σ
Injector #	105		104		115		103			
C^*_{inj} ft/sec	5869		5877		5855		5916		5879	
I_{sp} sec Ref 5	313.3		313.5		312.1		315.1			
R	0.05338		0.05334		0.05330		0.05326		0.05332	± 0.1%
C^*_{inj} ft/sec	5871		5872		5851		5905		5875	
I_{sp} sec Ref 8	313.4		313.6		312.2		315.2		313.7(1)	
R	0.05338		0.05341		0.05336		0.05338		0.05338	± 0.04%
I_{sp} sec (TRW - Ref. 7)			313.35		311.34		314.37		313.0(1)	
I_{sp} sec (ARO - Ref. 6)			313.5		311.3		314.8		313.2(1)	
Thrust Chamber #			311	315	313	320	318	324		
C^*_{inj} ft/sec (Ref. 8)	5871		5872		5851		5905			
I_{sp} sec (Ref. 5)	313.3	313.5	313.6	313.6	312.9	311.5	316.9	315.6	0.05338	± 0.14%
R	.05336	.05340	.05341	.05341	.05348	.05324	.05333	.05345		

Note: (1) Does not include injector #105 data.

Table 1

APOLLO BLOCK II SPS ENGINE CHARACTERISTIC EXHAUST VELOCITY CORRECTION VS. MR, T_P, P_C

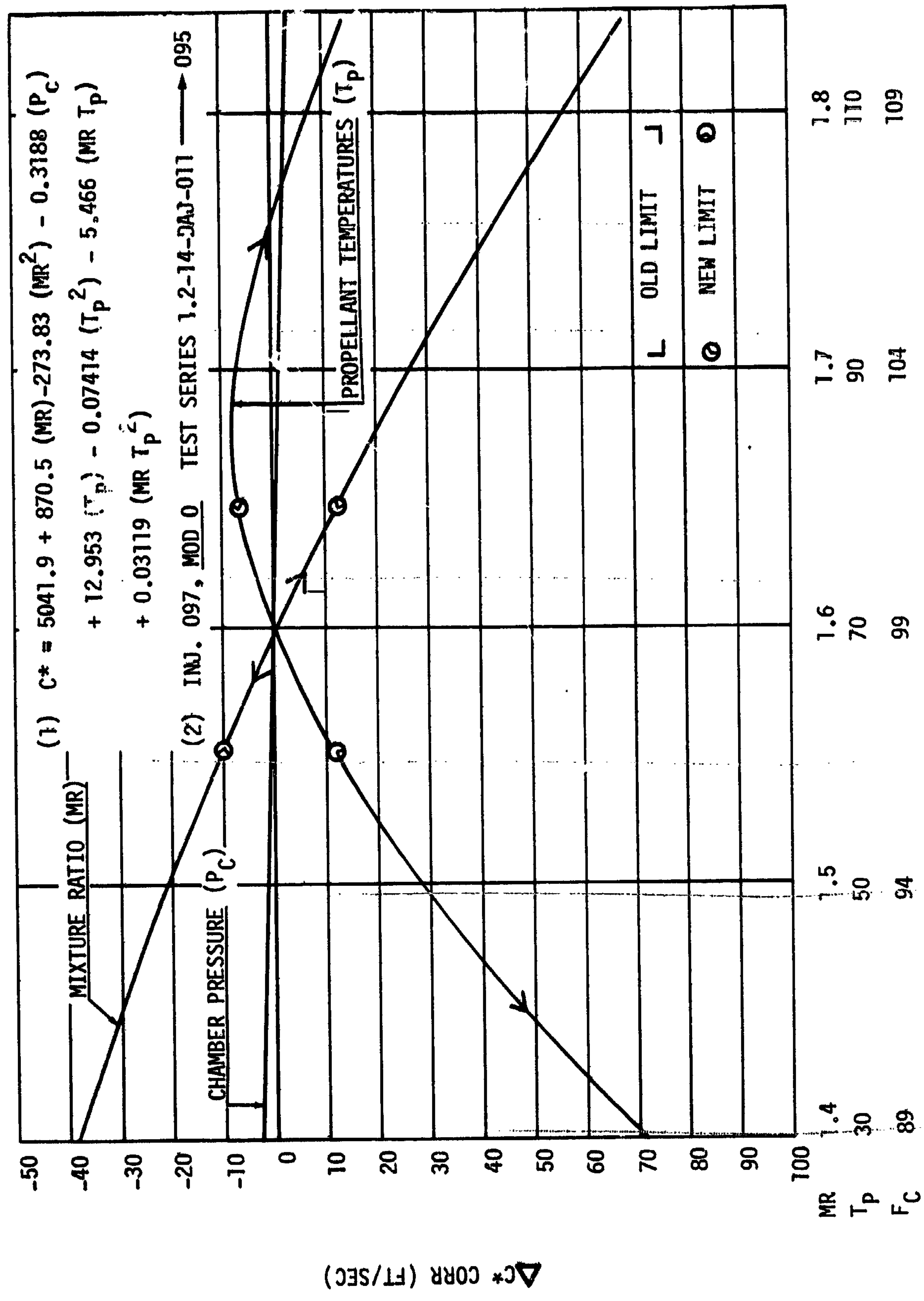


FIGURE 1
(FROM REFERENCE 2)

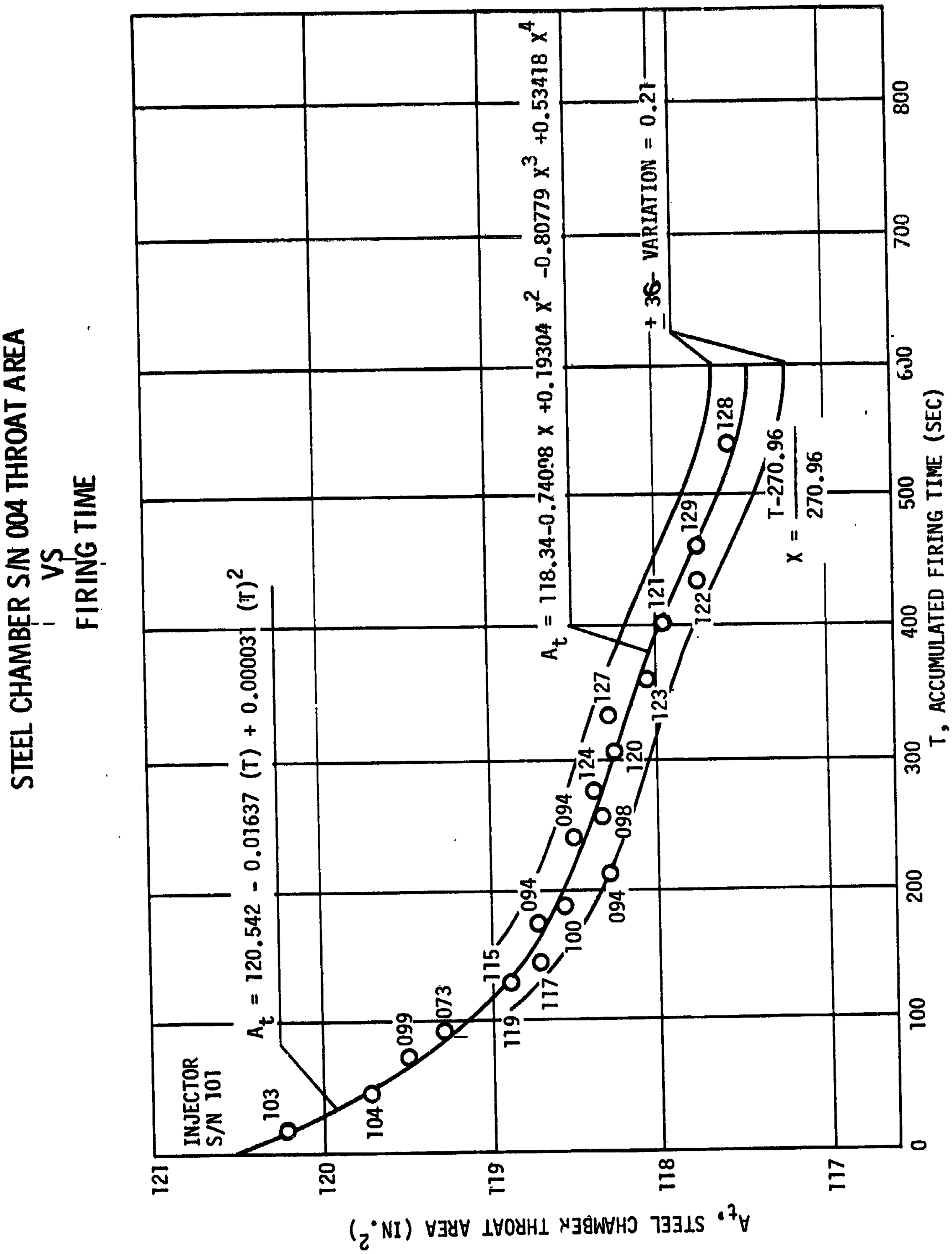
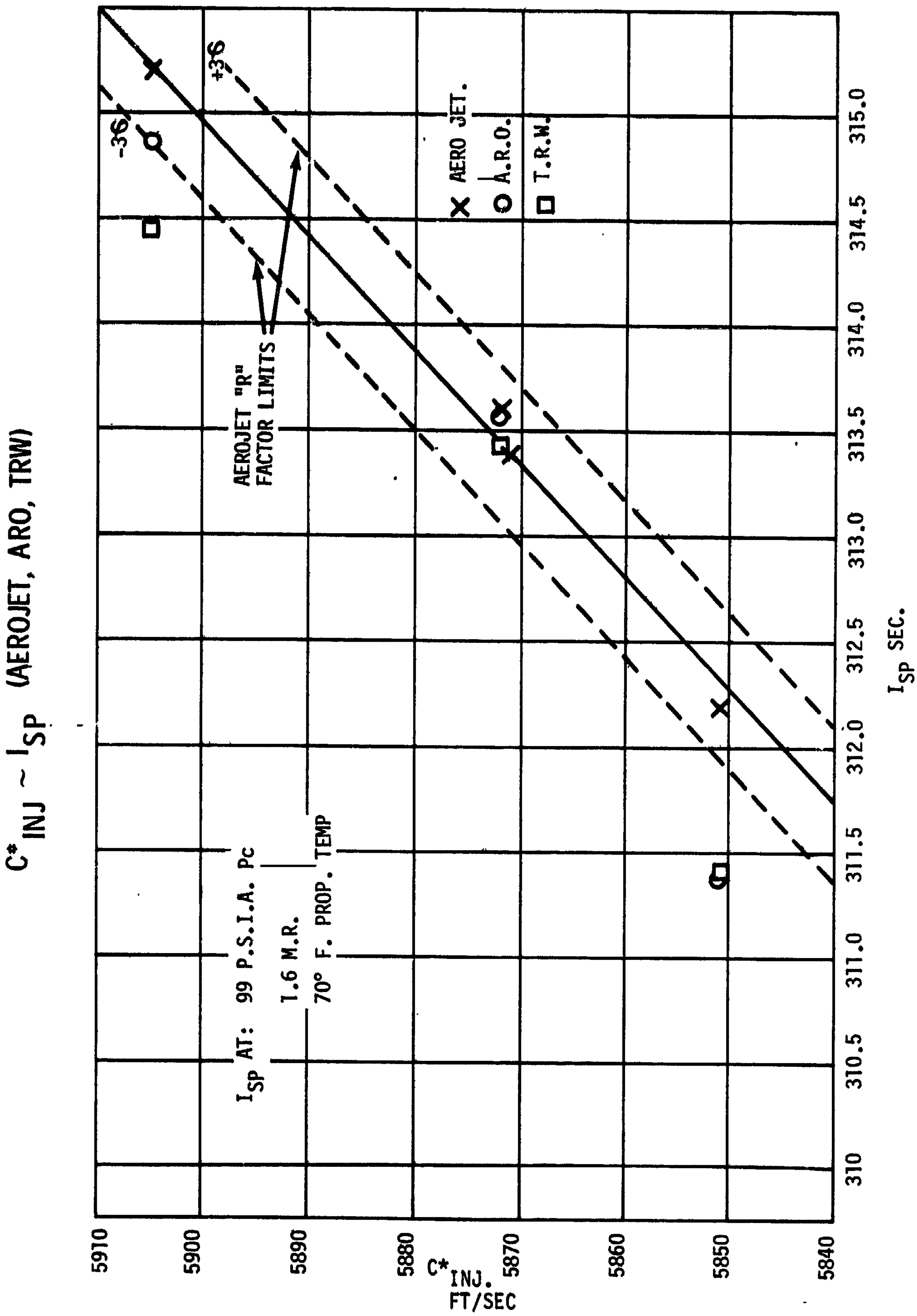


FIGURE 2
(FROM REFERENCE 2)



C*INJ ~ ISP (AEROJET, ARO, TRW)
ISP AT: 99 P.S.I.A. Pc
1.6 M.R.
70° F. PROP. TEMP
AEROJET "R" FACTOR LIMITS
+3σ
-3σ
AERO JET.
A.R.O.
T.R.W.
5910
5900
5890
5880
5870
5860
5850
5840
C*INJ. FT/SEC
310
310.5
311.0
311.5
312.0
312.5
313.0
313.5
314.0
314.5
315.0
ISP SEC.

FIGURE 3

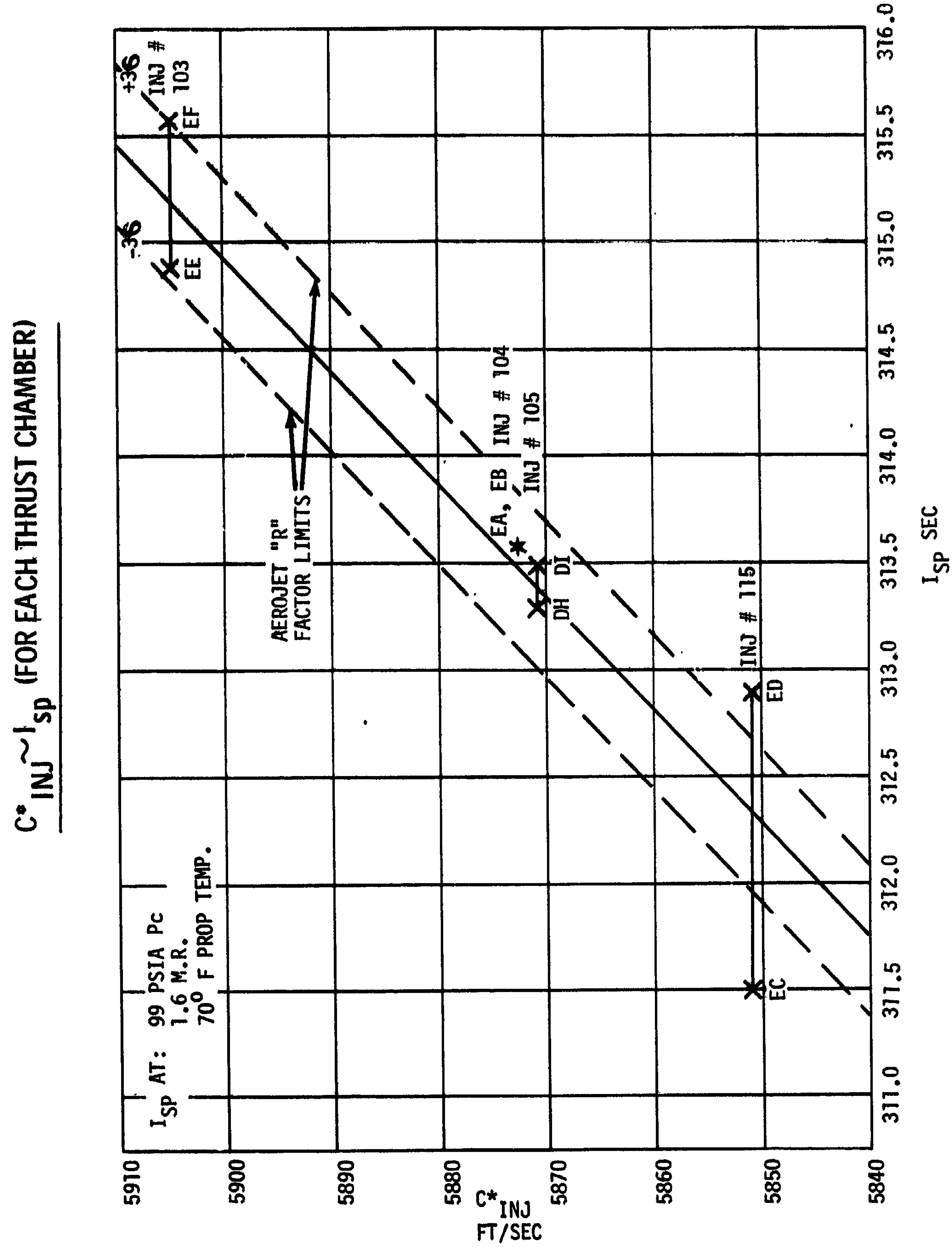

C*INJ ~ Isp (FOR EACH THRUST CHAMBER)
ISP AT: 99 PSIA Pc
1.6 M.R.
70° F PROP TEMP.
AEROJET "R"
FACTOR LIMITS
+3σ
-3σ
INJ # 103
EF
EE
EA, EB INJ # 104
INJ # 105
DH DI
INJ # 115
ED
EC
5910
5900
5890
5880
5870
5860
5850
5840
C*INJ
FT/SEC
311.0
311.5
312.0
312.5
313.0
313.5
314.0
314.5
315.0
315.5
316.0
ISP SEC

FIGURE 4

SPS ENGINE PERFORMANCE AS A FUNCTION OF CHAMBER PRESSURE, MIXTURE RATIO AND PROPELLANT TEMPERATURE

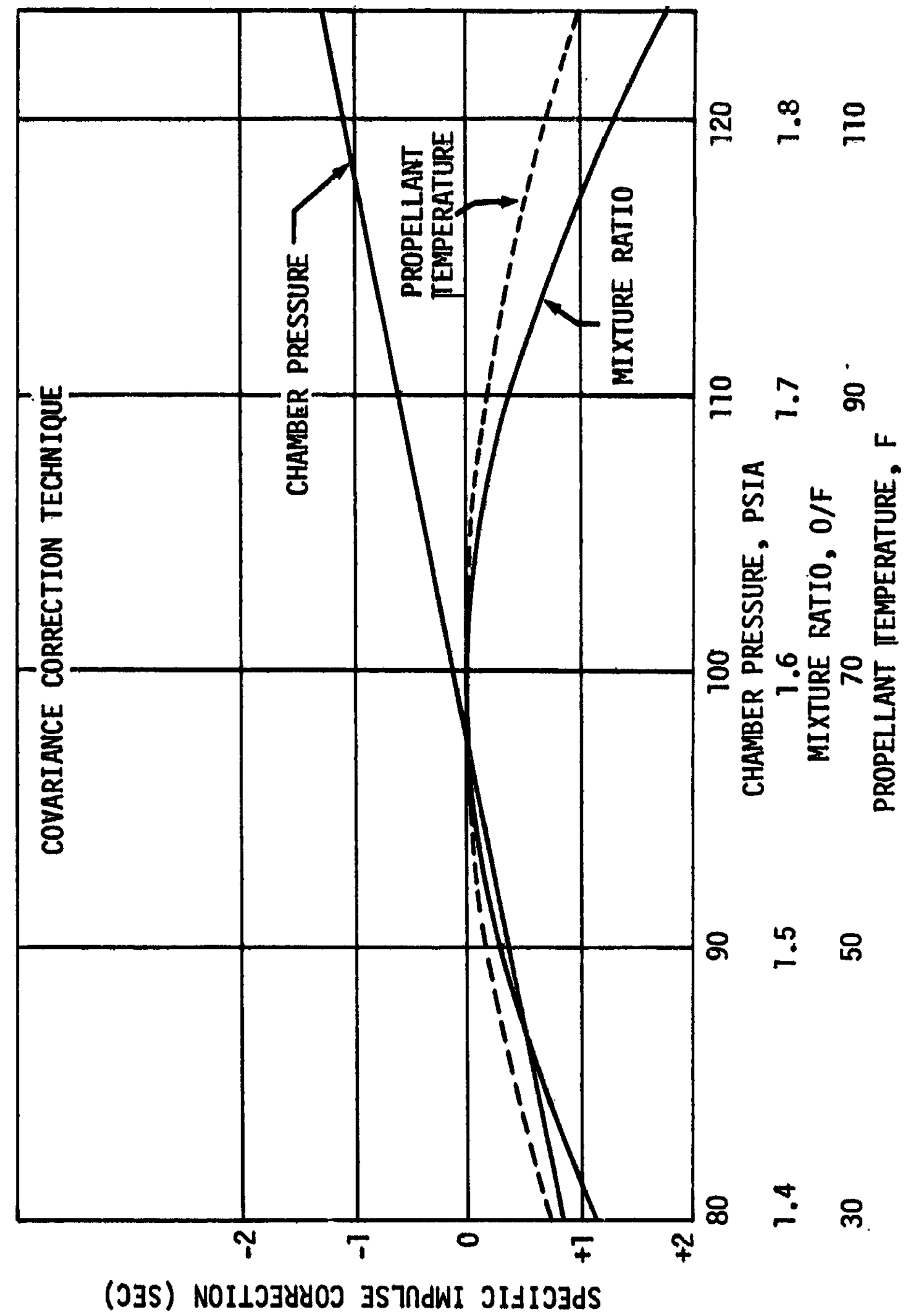
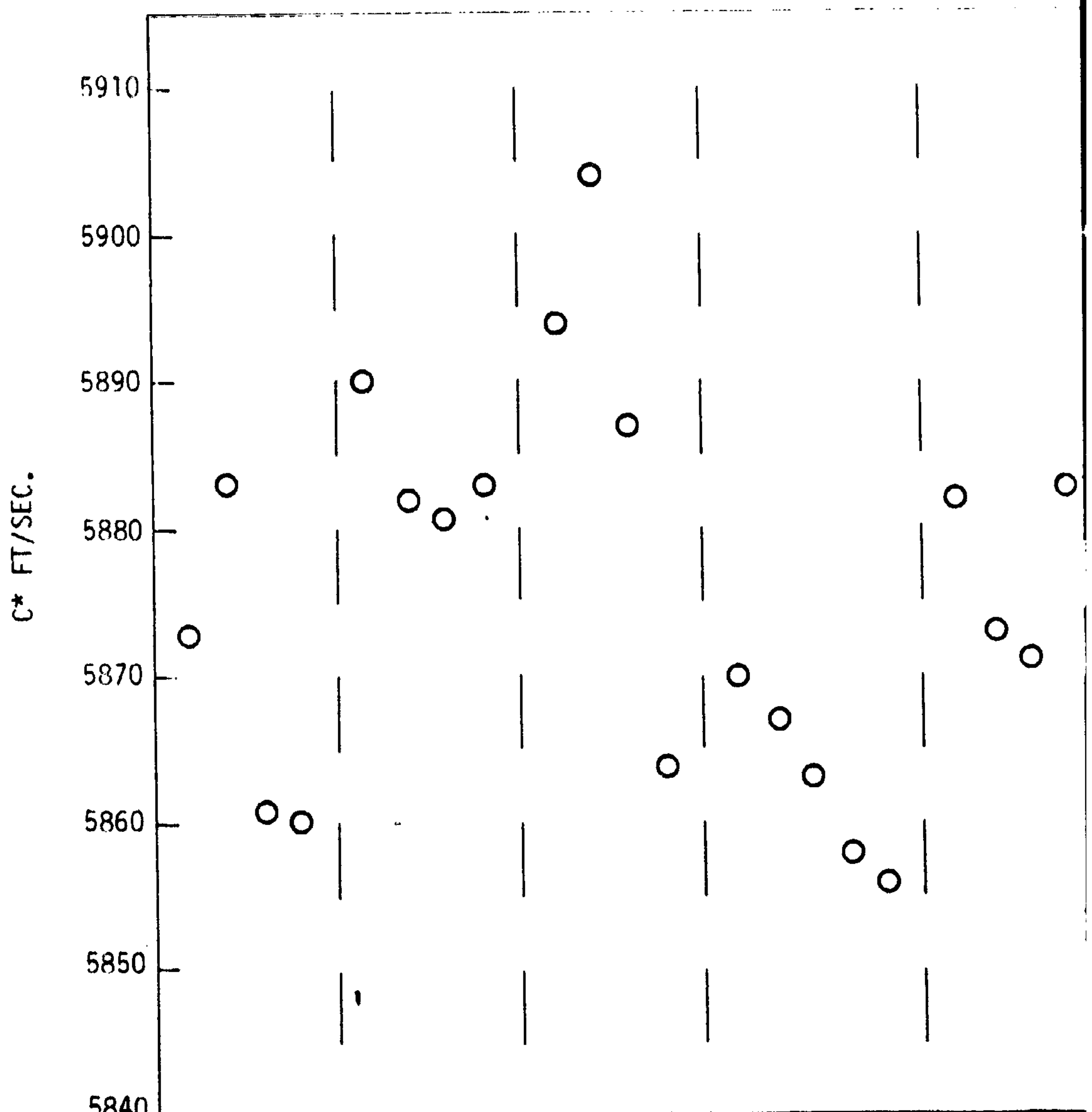


FIGURE 5
(FROM REFERENCE 2)

C*~ INJECTOR # (A



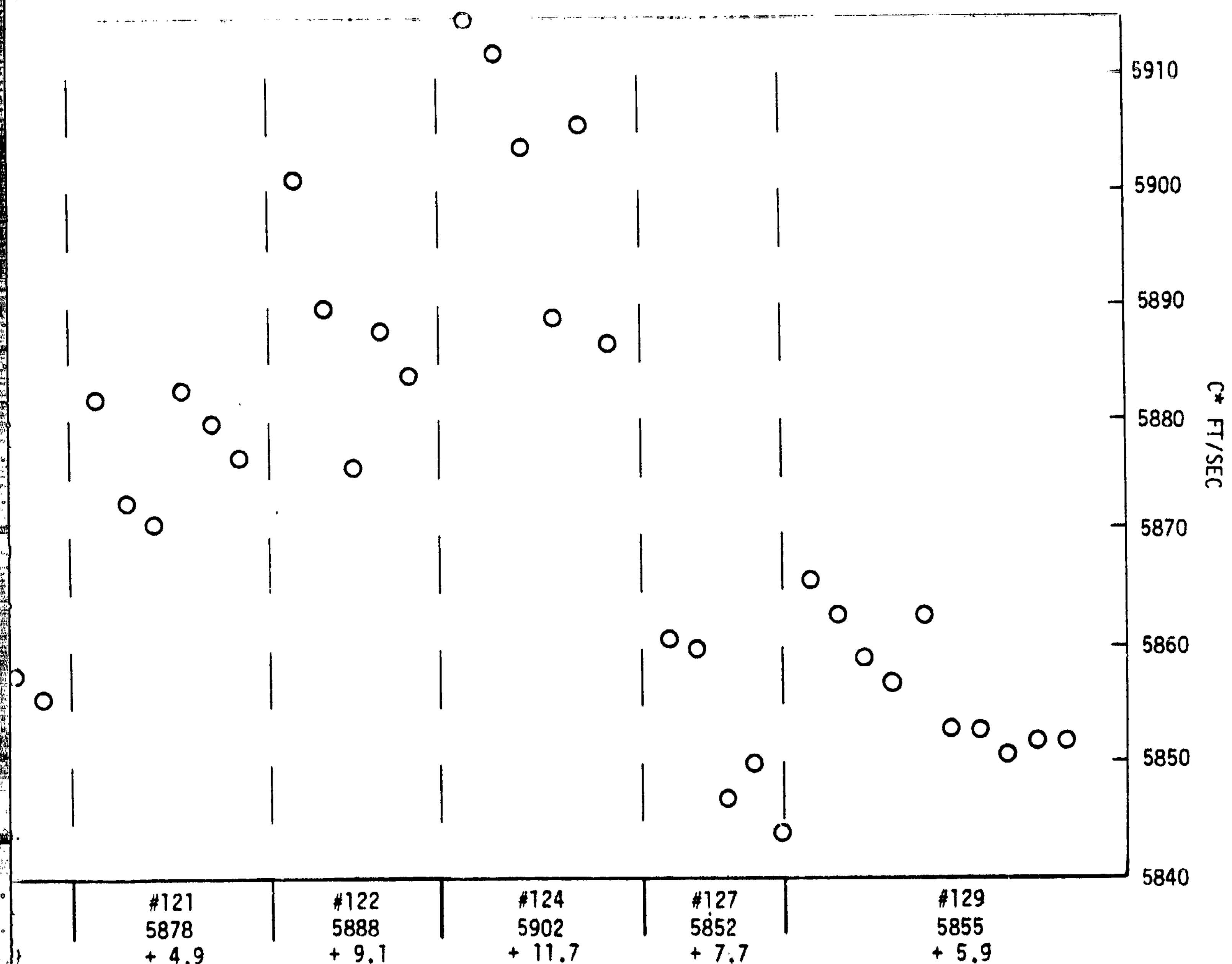
INJ #	#099	#100	#101	#120	#121
$\bar{C}^*$ FT/SEC.	5869	5884	5896	5863	5878
σ(RUN TO RUN)	± 10.6	± 4.1	± 19.8	± 5.9	± 4.[illegible]

FOLDOUT FRAME 1

2200

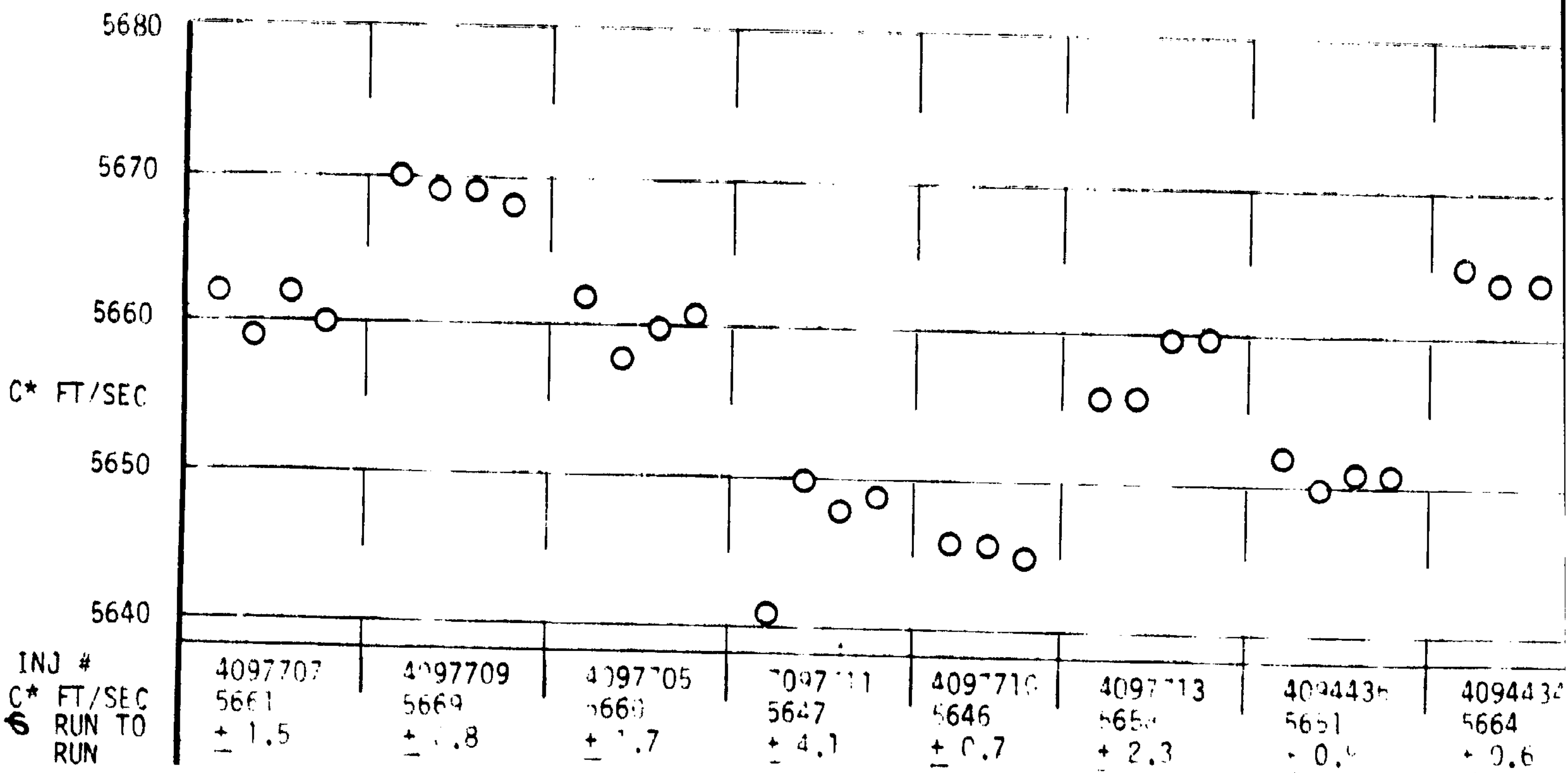
FIGURE 6

INJECTOR # (AEROJET S.M. ENGINE)



FOLDOUT FRAME 2

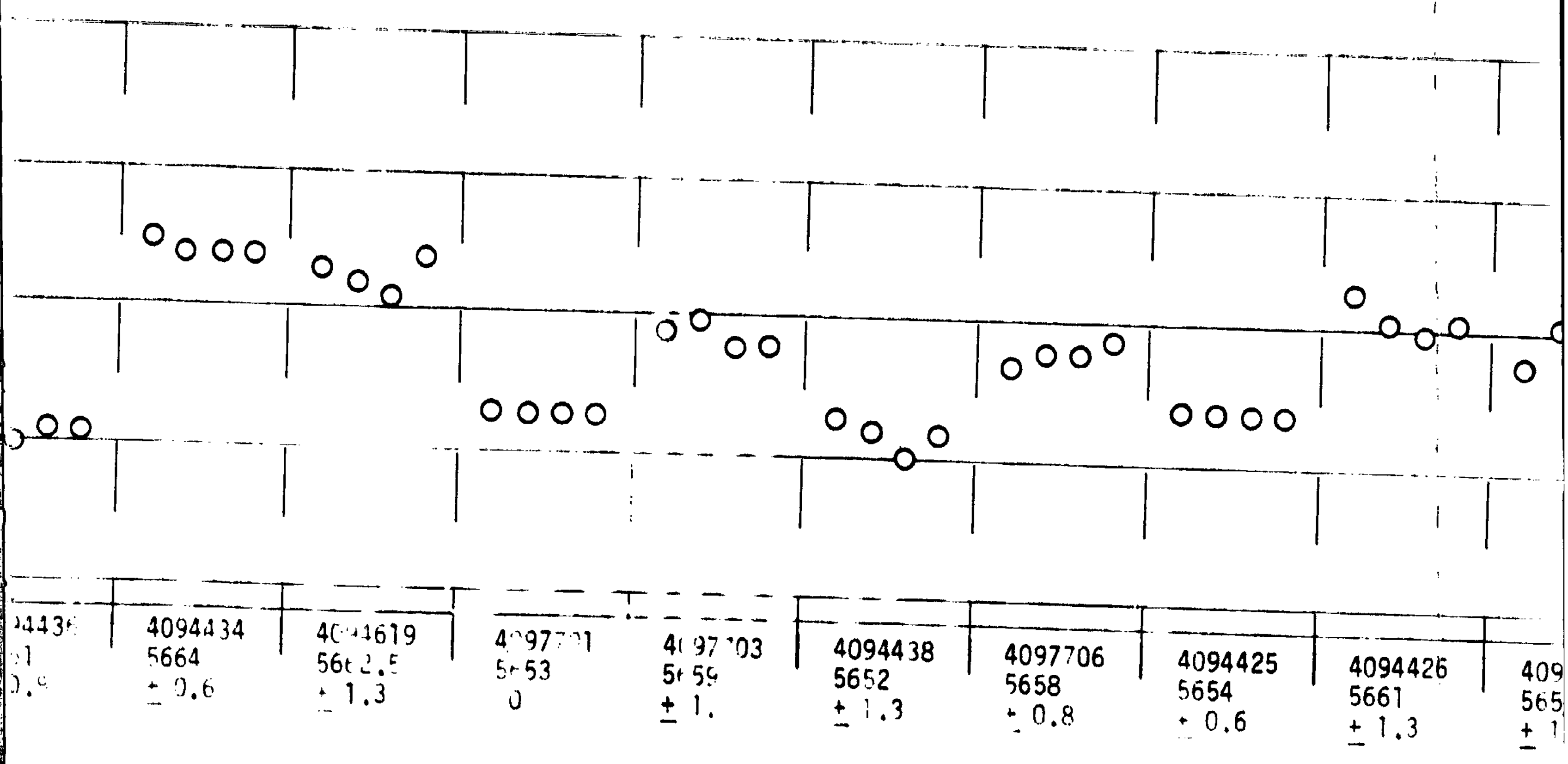
C* - INJECTOR # (



2200

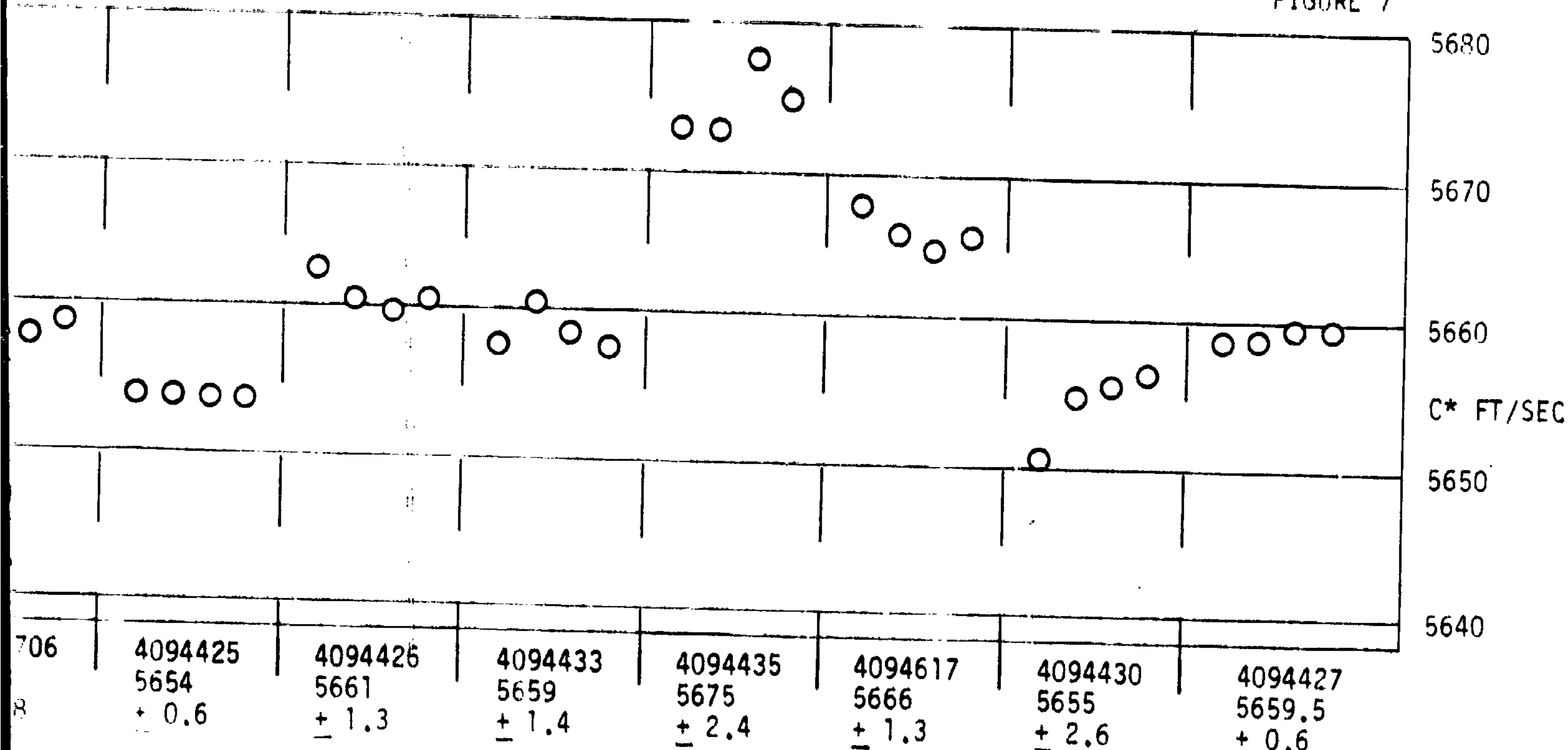
FOLDOUT FRAME

INJECTOR # (R L.M. ASCENT ENGINE INJECTOR DATA OBTAINED ON I.V.A. TESTS)

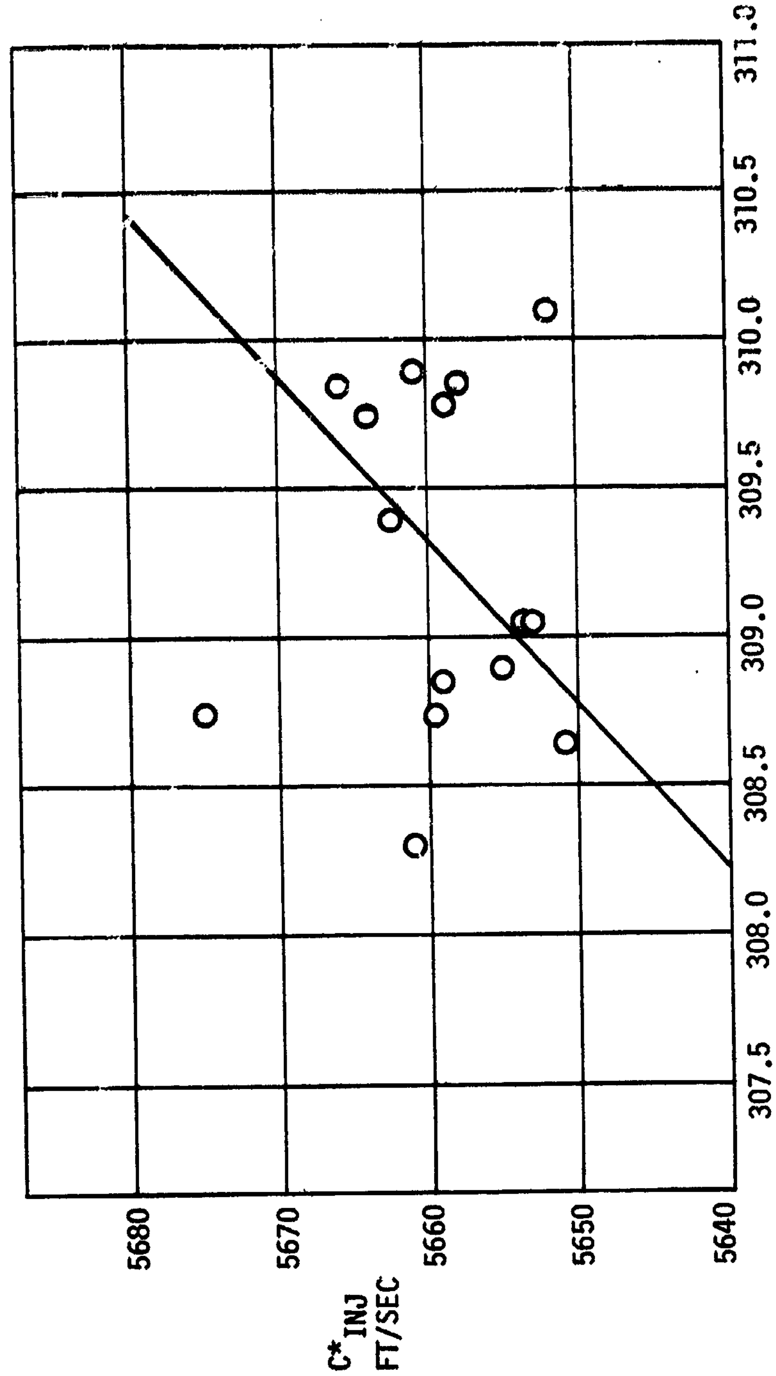


ON I.V.A. TESTS)

FIGURE 7



L.M. ASCENT ENGINE. $C^* \sim I_{sp}$



NOTE: C^*_{INJ} = MEAN VALUE OBTAINED ON I.V.A. TEST WITH STEEL CHAMBER (BRAVO)

I_{SP} = MEAN VALUE OBTAINED ON ENGINE ACCEPTANCE TESTS (NFL)

FIGURE 8

MOD IV FLIGHT INJECTOR PERFORMANCE

Inj No.	Steel Chamber Test No. 3.5-15-DAJ-XXX	Propellant Temperature °F (3)	MR	Individual Steel Chamber C* ft/sec	Average Steel Chamber C*, (1) ft/sec	1σ Variability in C* ft/sec (Run to Run	Nominal Isp, (6) sec	Minimum Isp, (5) sec
099	-033	62.2	1.55	5873				
	-034	62.2	1.62	5883	5869	± 10.6	313.3	311.8
Eng #56	-035	62.7	1.59	5861				
S.C. #?	-036	63.0	1.60	5860				
100	-059	46.9	1.62	5890				
	-060	46.8	1.60	5882	5884	± 4.1	314.1	312.6
Eng #57	-061	46.6	1.59	5881				
S.C. #103	-062	47.0	1.60	5883				
101	-015	64.7	1.55	5896				
	-016	64.2	1.62	5904	5896	± 19.9	314.7	313.2
Eng #52	-017	64.0	1.63	5887		± 8.2(4)		
S.C. #106	-018	63.8	1.60	5864(4)				
120	-118	66.4	1.59	5870				
	-119	67.3	1.60	5867	5863	± 5.9	313.0	311.5
Eng #62	-120	67.9	1.60	5863				
S.C. #104	-121	68.0	1.61	5858				
	-122	67.5	1.61	5856				
121	-140	56.8	1.59	5882				
	-141	56.8	1.60	5873				
Eng #63	-142	56.9	1.60	5871	5878	± 4.9	313.8	312.3
S.C. #?	-143	57.0	1.61	5883				
	-144	57.2	1.61	5880				
	-145	57.5	1.61	5877				

MOD IV FLIGHT INJECTOR PERFORMANCE (CONT'D)

Inj No.	Steel Chamber Test No. 3.5-15-DAJ-XXX	Propellant Temperature °F(3)	MR	Individual Steel Chamber C* ft/sec	Average Steel Chamber C*, (1) ft/sec	1σ Variability in C* ft/sec (Run to Run)	Nominal I_{sp}, (6) Sec	Minimum I_{sp} (5) Sec
122	-147	61.6	1.63	5901				
	-148	61.8	1.62	5890				
Eng #61	-149	61.6	1.61	5876	5888	± 9.15	314.3	312.8
S.C. #107	-150	61.3	1.63	5888				
	-151	61.0	1.62	5884				
124	-112	60.0	1.63	5915				
	-113	61.9	1.62	5912				
Eng #60	-114	62.9	1.61	5904				
S.C. #?	-115	63.7	1.61	5889	5902	± 11.7	315.0	313.5
	-116	75.8	1.59	5906				
	-117	76.3	1.60	5887				
127	-124	72.6	1.60	5861				
	-125	73.2	1.60	5860				
Eng #58	-126	73.2	1.60	5847	5852	± 7.75	312.4	310.9
S.C. #103	-127	72.5	1.61	5850				
	-128	71.3	1.60	5844				
129	-152	61.0	1.78	5866(2)				
	-153	6.12	1.66	5863(2)				
Eng #59	-154	61.1	1.64	5859				
S.C. #101	-155	61.0	1.64	5857				
	-156	61.0	1.65	5863	5855	± 5.9	312.5	311.0
	-157	61.1	1.62	5853				
	-158	61.1	1.62	5853				
	-159	60.9	1.62	5851				
	-160	60.8	1.61	5852				
	-161	60.7	1.62	5852				

Notes:

(1) Average of tests within an MR range of 1.55 to 1.65.

(2) Excluded from test series average (out of 1.55 to 1.65 MR range).

(3) Average of TOFM and TFFM.

(4) Excluded from test series average (deviation from average greater than 19 ft/sec).

(5) Nominal I_{sp} minus 1.5 sec.

(6) Average C* times 0.05338 $sec.^2/ft.$

(7) Average value of C* is 5875 ft/sec based on 49 tests.

PRODUCTION ROCKETDYNE LM ASCENT ENGINE $C^* \sim I_{sp}$ CORRELATION

Inj #	C* ft/sec	$\overline{C^*}$ ft/sec	1σ Variability in C* ft/sec (Run-to Run)	I_{sp} sec	$\overline{I_{sp}}$ sec	R
4097707	5662 5659 5662 5660	5661	± 1.5	310.0 309.8	309.9	0.05475
4097709	5670 5669 5669 5668	5669	± 0.8	Not Yet	Tested	At NFL
4097705	5662 5658 5660 5661	5660	± 1.7	Not Yet	Tested	At NFL
4097711	5641 5650 5648 5649	5647	± 4.1	Not Yet	Tested	At NFL
4097710	5646 5646 5645	5646	± 0.7	Not Yet	Tested	At NFL
4097713	5656 5656 5660 5660	5658	± 2.3	Not Yet	Tested	At NFL
4094436	5652 5650 5651 5651	5651	± 0.8	308.6 308.7	308.65	0.05462
[illegible]94434	5665 5664 5664 5664	5664	± 0.7	310.1 309.4	309.75	0.05469

PRODUCTION ROCKETDYNE LM ASCENT ENGINE $C^* - I_{sp}$ CORRELATION

Inj #	C* ft/sec	$\overline{C^*}$ ft/sec	1σ Variability in C* ft/sec (Run to Run)	I_{sp} sec	$\overline{I_{sp}}$ sec	R
4094619	5663 5662 5661 5664	5662.5	± 1.3	309.7 309.1	309.4	0.05464
4097701	5653 5653 5653 5653	5653	0	309.0 309.1	309.05	0.05467
4097703	5659 5660 5658 5658	5659	± 1	309.1 308.6	308.85	0.05458
4094438	5653 5652 5650 5652	5652	± 1.3	310.1 310.1	310.1	0.05487
4097706	5657 5658 5658 5659	5658	± 0.8	310.0 309.7	309.85	0.0547
4094425	5654 5654 5655 5654	5654	± 0.7	309.2 308.9	309.05	0.05466
4094426	5663 5661 5660 5661	5661	± 1.3	308.4 308.2	308.3	0.05446
4094433	5658 5661 5659 5659 5658	5659	± 1.4	309.9 309.7	309.8	0.05474

PRODUCTION ROCKETDYNE LM ASCENT ENGINE $C^* \sim I_{sp}$ CORRELATION

Inj #	C^* ft/sec	$\overline{C^*}$ ft/sec	1σ Variability in C^* ft/sec (Run to Run)	I_{sp} sec	$\overline{I_{sp}}$ sec	R
4094435	5673 5673 5678 5675	5675	± 2.4	308.9 308.6	308.75	0.05441
4094617	5668 5666 5665 5666	5666	± 1.3	309.9	309.85	0.05469
4094430	5651 5655 5656 5657	5655	± 2.6	308.7 309.1	308.9	0.05462
4094427	5659 5659 5660 5660	5659.5	± 0.6	309.3 308.2	308.75	0.05455
	Average	5658.5				0.05465
1 Standard Deviation		± 7	± 1.6			± 0.0001174 = ± 0.215%

Appendix C
(from Ref. 2)

D2-117060-2

Aerojet - SPS Engine

Future Development

1. Continued use of steel chamber, S/N 004. Build new chamber and run correlation test.
2. Possible specification changes to MR, P_c, and T_p limits and limit on time between tests.
3. Test programs to determine C* corrections for Mod IV and confirmation of R factor:

   A. Sea Level Testing

      (1) Acceptance test 3 injectors per ATP 46847 (Ref. 3). One injector $T_p = 40 \pm 5°F$. One injector $T_p = 30 \pm 5°F$.

      (2) Five 20-second firings (each injector) at nominal conditions on a water-cooled combustion chamber.

   B. Simulated Altitude Testing
   At AEDC per Chart.

Appendix C
Cont'd

PROPOSED MOD IV INJECTOR PERFORMANCE
CORRECTION ALTITUDE TEST PROGRAM

Run No.[1]	Duration, sec	P_c psia	MR	Propellant Temp, °F
1	20.0	99	1.4	70
2	20.0	99	1.5	70
3	20.0	99	1.6	70
4	20.0	99	1.7	70
5	20.0	99	1.8	70
6	15.0	80	1.4	70
7	20.0	80	1.6	70
8	16.0	80	1.8	70
9	15.0	110	1.4	70
10	20.0	110	1.6	70
11	15.0	110	1.8	70
12	15.0	99	1.4	40
13	15.0	99	1.5	40
14	20.0	99	1.6	40
15	15.0	99	1.7	40
16	15.0	99	1.8	40
17	15.0	80	1.4	40
18	20.0	80	1.6	40
19	15.0	80	1.8	40
20	15.0	110	1.4	40
21	20.0	110	1.6	40
22	15.0	110	1.8	40
23	15.0	99	1.4	90
24	15.0	99	1.5	90
25	20.0	99	1.6	90
26	15.0	99	1.7	90
27	15.0	99	1.8	90
28	15.0	80	1.4	90
29	20.0	80	1.6	90
30	15.0	80	1.8	90
31	15.0	110	1.4	90
32	20.0	110	1.6	90
33	15.0	110	1.6	90
34	400.0	99	1.6	70
35	0.5	99	1.6	70
36	0.5	99	1.6	70

Note: (1) Pulse test to be accomplished during first firing of each air period.