Development of the Scorpius® LOX/Kerosene Engine Family

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Abstract

Microcosm and Sierra Engineering have been developing a series of low-cost pressure fed LOX/kerosene engines to power the Scorpius® family of launch vehicles. This paper focuses on the design, fabrication and testing of a 20K lb_f O-F-O triplet injector with a flight-type ablative chamber. The test results indicated good performance and stability. The ablation rates were higher than desired. Minor design modifications and additional testing will complete the engine verification process.

Introduction

Microcosm has been involved in the development of low-cost ablative engines since the inception of the Scorpius® program in 1992.[1] Low-cost pressure-fed engines constitute one of the three key technologies used on the Scorpius® family of sub-orbital and orbital launch vehicles. The other technologies are all-composite propellant tanks and High Performance Pressurization Systems (HPPS). The Scorpius® family of engines are low-cost, simple, pressure-fed designs of thrust classes ranging from 5K to 320K lb_f. The engines use LOX and Jet-A kerosene to power all stages. For example, the Sprite Mini-lift vehicle, shown in Figure 1, is capable of launching 800 lb_m to LEO.[2]

Microcosm's 20K lb_f engines and its derivative larger engines are configured in various launch vehicles within the Scorpius® family based on the payload requirements of the respective launch vehicle. A single 20K lb_f engine is used on the sub-orbital SR-M vehicle as well as for the individual pods of the Sprite booster and sustainer stages.

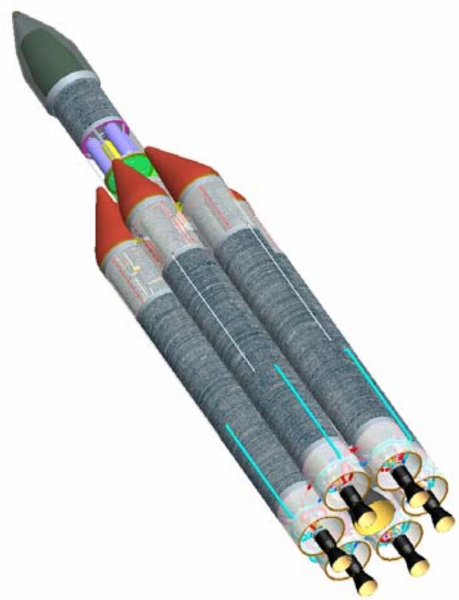


Figure 1: Sprite Mini-lift Launch Vehicle.

Larger vehicles use a cluster of these engines to meet the propulsion requirements. For example, the Eagle vehicle, capable of carrying two times the payload of the baseline Sprite vehicle (~1600 lb_m to LEO), is configured with two 20K lb_f engines in each of the pods of the first and second stages.

The Liberty vehicle, with roughly four times the payload of the Sprite vehicle, would require four 20K lb_f engines per pod. Instead, to minimize the complexity of the system, Microcosm plans to use a single 80K lb_f thrust engine in each pod of the Liberty vehicle.

The upper stages of the various launch vehicles in the Scorpius® family require lower thrust than the booster and 2nd stages. The upper stage weight also optimizes with reduced tank pressures (200 psi for the upper stage versus 550 psi for the booster). The lower tank pressure mandates reduced engine chamber pressure. The Eagle launch vehicle requires an 8K lb_f thrust class upper stage engine. Design studies have shown that the booster engine design can be operated at the reduced thrust levels with minor modifications. This not only minimizes the development cost of the engine, it keeps the production cost down as the engine shares many parts with the booster engine.

Engine Design

The engine technology started with the design, development, and flight demonstration of a 5K lb_f thrust-class engine (Figure 2). For expendable engines, ablatively-cooled chambers provide good performance at greatly reduced cost compared to regeneratively-cooled chamber designs. The flight engine utilized a like-on-like doublet injector and an ablative chamber, although an F-O-O-F split triplet was also tested.

Microcosm began working with Sierra Engineering (Sierra) to develop a low-cost 20K lb_f injector in late 2002. Scorpius® requirements for the 20K lb_f engine are listed in Table 1. Sierra traded seven different injection element concepts on a variety of criteria including performance, stability, compatibility, development cost and recurring cost. The trade ranked an O-F-O triplet, a like doublet and a pintle as the top three injector concepts. The O-F-O triplet injection element is ideally suited for LOX/hydrocarbon propellant combinations, where the injected O/F mixture ratio (MR) for optimal performance is about 2.6.

Figure 2: 5K lb_f Thrust Chambers.

Table 1: 20K lb_f Scorpius® Launch Vehicle Engine Requirements

• Propellants	
– Oxidizer	LOX per MIL-PRF-22508F
– Fuel	Jet-A per ASTM D1655-04
• Nominal Mixture Ratio (MR)	2.4
• Engine inlet pressure (downstream of valves)	500 psia maximum
• Vacuum Thrust	20,000 lb_f
• Cumulative Burn Duration	200 seconds
• Vacuum Specific Impulse (Isp)	280 lb_f-sec/lb_m
• Nozzle Expansion Ratio	6.56:1
• Maximum wall temperature	3900 °R
– Compatible with Silica phenolic chamber liner	

Triplet injectors offer the potential for higher performance than the doublet injector, even with the coarse injector pattern needed to achieve acceptable stability characteristics.[3] However, wall compatibility can be an issue with O-F-O triplet injectors. This injector is the baseline for the current family of Scorpius® launch vehicles.

The injector includes 63 O-F-O triplets, with equal orifice diameters (0.116 inches). Canted showerhead fuel film cooling (FFC) orifices (66) are located around the periphery. Adjustment of the FFC orifice size and inclusion of a metering plate permit the FFC to be varied between 2 and 12%.

The injector layout incorporates a flooded LOX manifold for enhanced cooling of the injector face (Figure 3). Detailed transient thermal-structural of the injector assembly showed very little plasticity and no ratcheting. Cyclical Manson strain range is below 0.0059, with life predicted to exceed 580 cycles with a safety factor of 10.

The injector was designed to reduce production costs while enhancing reliability. Interpropellant leak paths were eliminated. Materials with excellent oxygen compatibility were selected.[4] Welds were reduced to two, with neither contributing to a CRIT1 failure. Brazes were eliminated. The component designs were iterated with machine shops to optimize them for CNC machining.

Pretest analysis suggested that the engine would be at least spontaneously stable without the use of stability aids. Testing was planned in both a steel hardwall chamber and a flight-type ablative chamber. Provisions were made to include a ¼-wave acoustic cavities if necessary – a cavity spool was built for the hardwall chamber and the ablative chamber could be cut to create the necessary recess.

The ablative combustion chamber is made of silica phenolic with a graphite-epoxy filament overwrap. This is a scale-up of the 5K lb_f chamber fabrication process (Figure 4).

Ignition of the 1st stage booster engine will utilize a ground ignition system, likely a pyrotechnic or bipropellant torch. However, the 2nd and 3rd stages rely on pyrophoric ignition for high-altitude ignition and restart capability. The injector incorporates three ports that can be used for both chamber pressure measurement and injection of the pyrophoric ignitier fluid. Initial ground testing utilized a pyrotechnic ignition system.

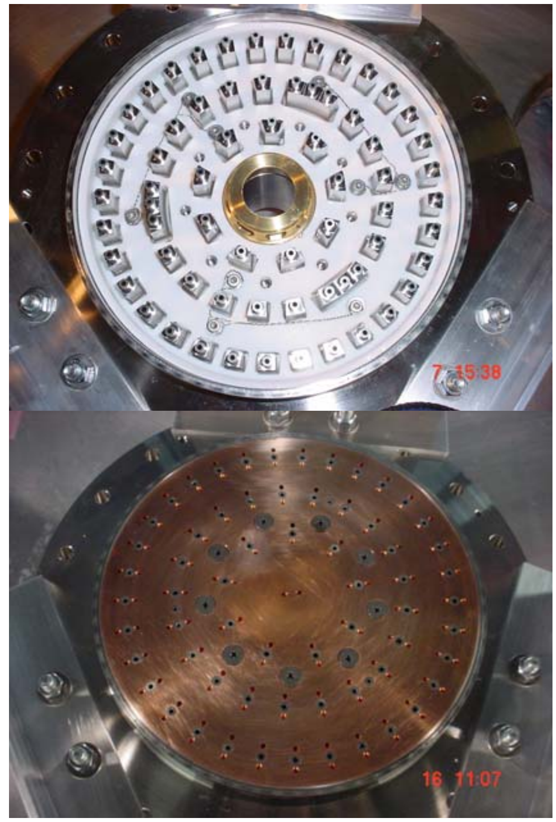


Figure 3: 20K lb_f Triplet Injector Before (T) and After (B) Faceplate is Attached

The development of an 80K lb_f thrust engine for the Liberty vehicle, derived from the 20K lb_f triplet engine, was initiated under a Phase I SBIR from AFRL. Parallel development is proceeding.

Figure 4: 20K lb_f Ablative Chambers.

Engine Testing

A 2-week engine test campaign was conducted in the North test cell of Test Stand 2A at Edwards AFB during May 2005. Testing was performed using both the hardwall steel chamber (Figure 5) and three flight-type ablative chamber (Figure 6). Two injectors were tested, S/N 001 with 3.8% FFC and S/N 002 with 7.1% FFC.

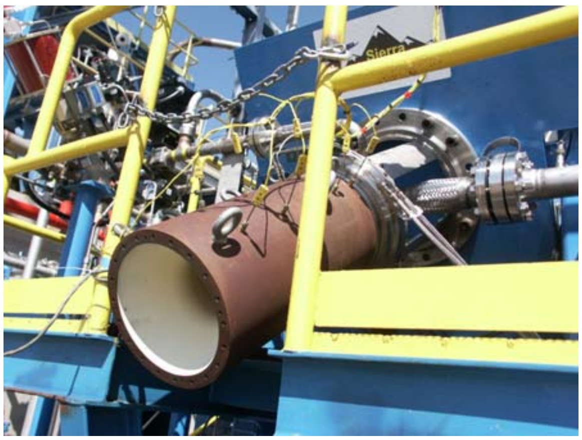


Figure 5: **20K lb_f with Hardwall Chamber on Test Stand 2A at Edwards AFB.**

There were twelve successful hot fire tests (Table 2), out of eighteen attempted tests. Most failed tests were associated with facility redline kills. Test chamber pressure (PC) ranged from 243 to 392 psia and covered a MR range from 2.18 to 2.36. The maximum test duration was 30 seconds. Up to 5 starts were performed on a single ablative chamber.

Post-test inspection of the injectors showed good hardware durability. There was no damage to the injector face (Figure 7). A couple small pits were found on the injector periphery; one was associated with a remanufactured film cooling injection hole (Figure 8). Throat erosion appeared to be uniform (Figure 7).

Characteristic velocity (C*) was the primary performance metric, as thrust was not measured during this test series. Redundant measurements included propellant flowrate and injector face pressure. Delivered C* calculations accounted for a throat C_D and total pressure loss between the injector face and throat. The throat area was estimated from pre- and post-test throat measurements using conservative assumptions for throat growth rate. A detailed uncertainty analysis was performed on the calculated C*; uncertainty was computed to be ±1.39% (about 76 ft/s C*). Specific impulse (ISP) values were estimated from delivered C* and predicted nozzle efficiency. This translates into an ISP uncertainty of ±4 lb_f-s/lb_m.

Figure 6: **20K lb_f Engine Firing with Flight-type Chamber.**

The delivered C* ranged from 5423 to 5547 ft/s during long duration tests. The estimated vacuum ISP ranged from 281 to 288 lb_f-s/lb_m, in all cases exceeding the required 280 lb_f-s/lb_m. There is a weak trend of increasing engine performance with increasing chamber pressure (Figure 9 and Figure 10). The data also shows a weak downward trend in C* with increasing MR (Figure 10). There is no clear decrease in performance as FFC percentage is increased (Figure 9). Test 19 was very near the nominal engine operating condition, producing an average C* of 5387 ft/s (93.4% efficiency) and a vacuum ISP of 285 lb_f-s/lb_m.

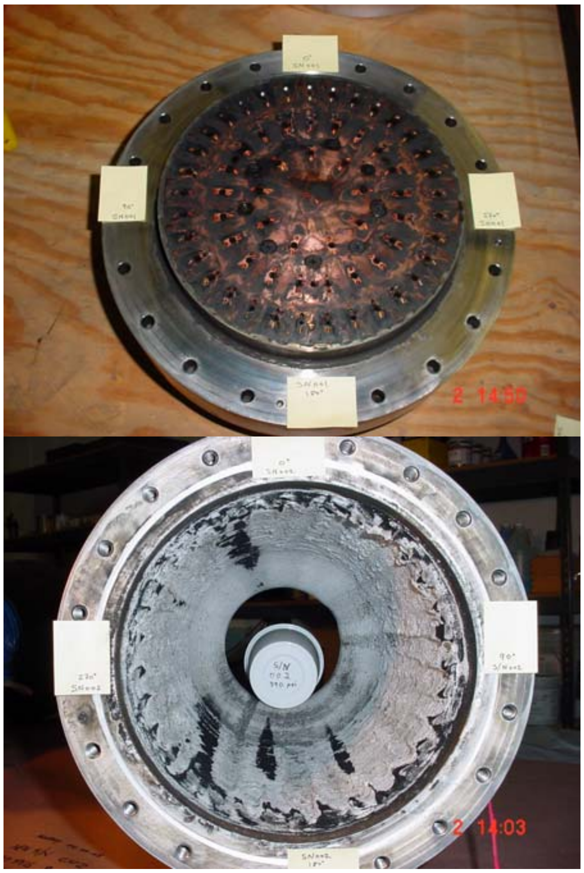


Figure 7: Post-test Images of Injector s/n 001 (T) and Chamber s/n 002 (B).

Dynamic combustion stability was never demonstrated per CPIA 655.[5] The injector was initially tested in a steel hardwall chamber that did not include acoustic cavities. During the second bipropellant test (Test 3), the injector transitioned from rough combustion (8% peak-to-peak roughness) to an organized limit-cycle 1T instability (2800 Hz). The instability resulted in only minor damage to the injector (pitting of the face bolts). However, errors in the engine shutdown sequence resulted in damage of the hardwall chamber, yielding it unusable on subsequent tests.

Figure 8: Close-up of Pit on Injector Lip Associated with Remanufactured FFC Orifice.

Subsequent testing was performed using ablative chambers. The chamber head-end was modified to create a ¼-wave acoustic cavity. The ablative chambers did not include high-frequency chamber pressure measurements. However, the injector instrumentation included triaxial shock accelerometers. Data from Test 3 showed good frequency correlation between the accelerometers and the chamber pressure measurement, and it permitted the accelerometer amplitude to be roughly correlated with chamber pressure oscillations (Figure 12). Data from Test 17 indicates a steady acceleration of approximately 70 g peak-to-peak, which corresponds to about 20 psi peak-to-peak (about 7.5% of PC). The effectiveness of the acoustic cavity was also demonstrated. During the course of Test 17, there were several large amplitude (about 80 psi or 35% of PC) "pops" (Figure 13). These pops damped quickly (approximately 4 msec.), well below the CPIA 655 requirement damp time of less than 21 msec.

The chamber ablation rate was slightly higher than desired for the Scorpius® launch vehicle family booster and sustainer engines. In particular, there was too much erosion in the barrel portion of the chamber. The barrel ablation rate is high in part because the film cooling injection orifices were not configured for the inclusion of the acoustic cavity. The inclusion of the acoustic cavity forced the FFC to "jump" the cavity inlet before creating a fuel film along the wall; the location of the cavity lip should have been considered when the FFC spray angle was set. The barrel material density and fuel film cooling injection features are being adjusted to enhance compatibility between the chamber and the triplet injector. This should minimize the erosion rate. The barrel wall could also be made thicker to increase life.

Three narrow field radiometers were mounted to measure the exhaust plume emission (Figure 14). Data comparisons with model plume calculations showed good agreement. The model plume calculations are a critical part of the vehicle base heating analyses.

Summary

The Scorpius® family of rocket engines has achieved a development milestone. A 20K lb_f O-F-O triplet injector was tested with flight-type silica phenolic ablative chambers. The calculated ISP ranged from 281 to 288 lb_f-s/lb_m with an uncertainty of ±4 lb_f-s/lb_m. A total of 12 tests were run, with a maximum run duration of 30 sec. A 1T instability (2800 Hz) occurred early in the test series with the hardwall chamber. Nine subsequently tests in an ablative chamber incorporating a ¼-wave acoustic cavity showed good stability characteristics. The ablation rates were slightly higher than desired. Minor design changes to the injector and the ablative liners are being performed to complete the 20K lb_f engine development.

References

[1] Conger, R.E., Chakroborty, S., Wertz, J.R. and Kulpa, J.; "The Scorpius Expendable Launch Vehicle Family and the Status of the Sprite Mini-Lift", 20th AIAA International Communications Satellite Systems Conference (ICSSC), Montreal Canada, 13-15 May 2002

[2] Chakroborty, S., Wertz, J.R. and Conger, R.E.; "The Scorpius Expendable Launch Vehicle Family and the Status of the Sprite Small Launch Vehicle", AIAA-LA Section/SSTC Responsive Space Conference, 2003

[3] Muss, J.A., “Advances in the Understanding of Combustion Characteristics of LOX/Hydrocarbon Rocket Engines”, Liquid Propellant Rocket Combustion Instability, V. Yang and W. Anderson Ed., AIAA Progress in Aeronautics and Astronautics V 169, 1995

[4] *Safe Use of Oxygen and Oxygen Systems,* H.D Beeson, W.F. Stewart and S.S. Woods Editors, ASTM Manual 36, 2000

[5] *Guidelines for Combustion Stability Specification and Validation Procedures for Liquid Rocket Engines*, CPIA Publication 655, Jan 1997

Table 2: Summary of Test Data

Test Number	Injector	Chamber	PC (psia)	O/F MR	Steady-State Duration (s)	C* (ft/s)	ηC* (%)	Estimated ISP (lbf-s/lbm)
001	S/N 002	Hardwall	-	?	-			
002C	S/N 002	Hardwall	235.5	2.772	0.2			
003	S/N 002	Hardwall	243.1	2.310	1.8	5479	93.3%	285
010A	S/N 002	Ablative #1	265.7	2.626	0.7	5737	98.8%	301
011	S/N 002	Ablative #1	257.2	2.182	2.2	5472	93.1%	283
012	S/N 002	Ablative #1	257.7-258.3	2.23-2.27	5.2	5484	93.3%	284
						5547	94.4%	288
013	S/N 002	Ablative #1	255.9-257.2	2.26-2.29	10.2	5464-5546	93.0-94.4%	284-288
014	S/N 002	Ablative #1	252.6-53.4	2.23-2.24	10	5432-5517	92.2-93.8%	281-286
016	S/N 001	Ablative #3	257.6	2.266	1	5459	92.9%	283
017	S/N 001	Ablative #3	257.2-258.0	2.33-2.26	30	5472-5482	93.3-93.4%	285-286
018A	S/N 001	Ablative #2	390.2	2.237	1	5510	93.7%	286
019	S/N 001	Ablative #2	390.6-392.4	2.30-2.34	10	5487-5497	93.3-93.6%	2845-286

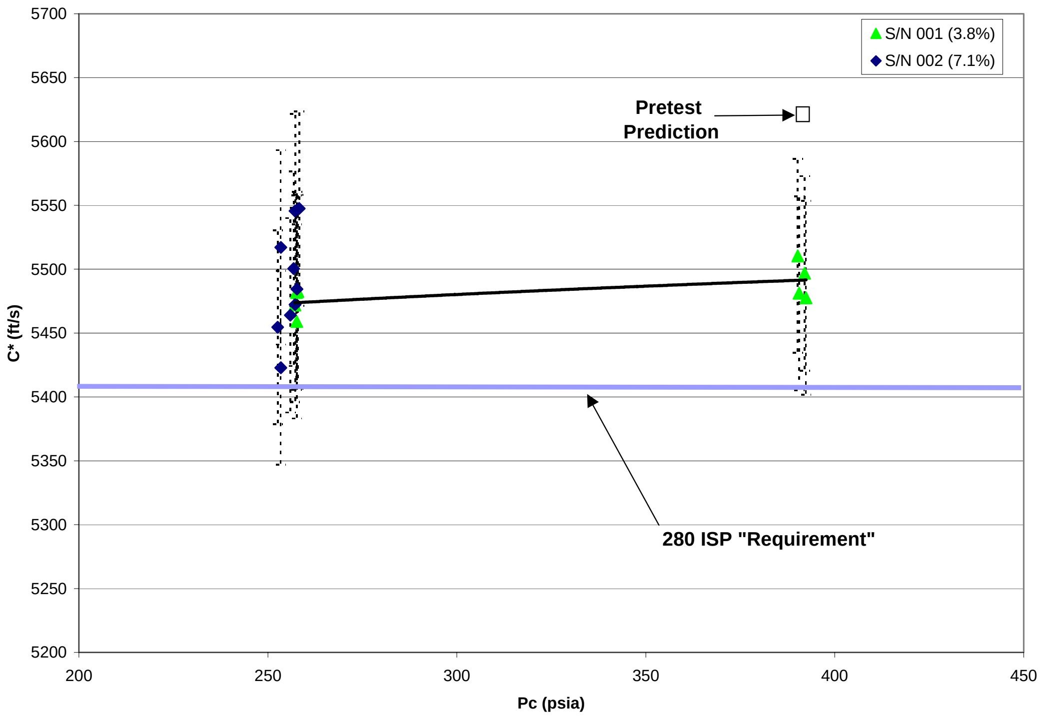


Figure 9: Trends of C* with Chamber Pressure and FFC Percentage, Uncertainty Included

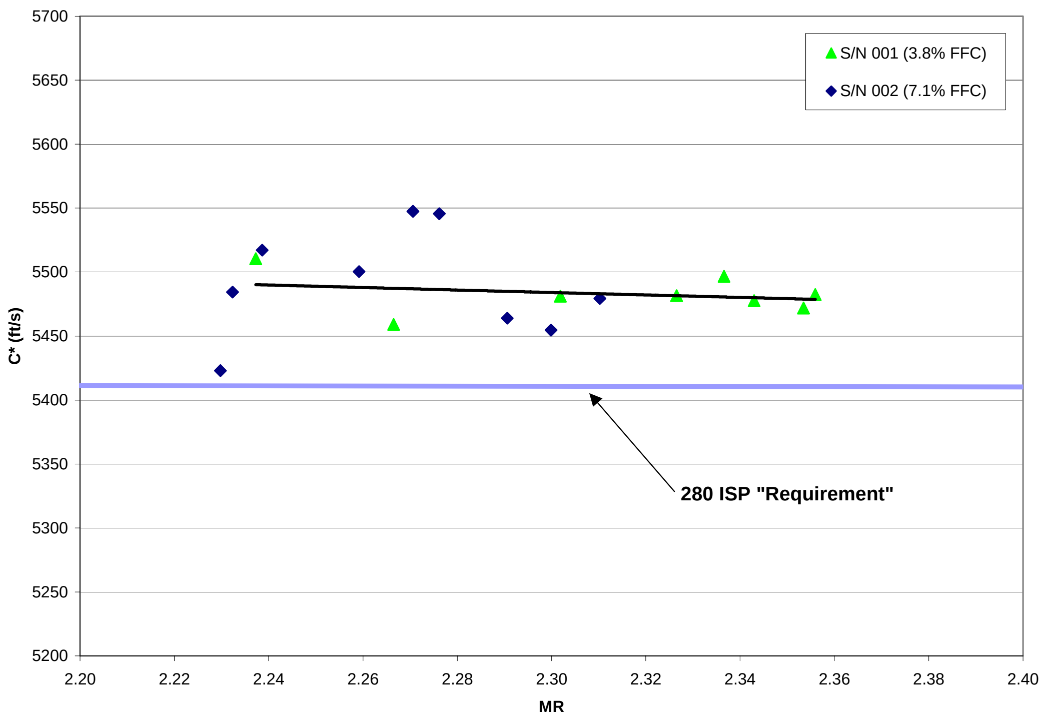


Figure 10: Trends of C* with Mixture Ratio and FFC Percentage

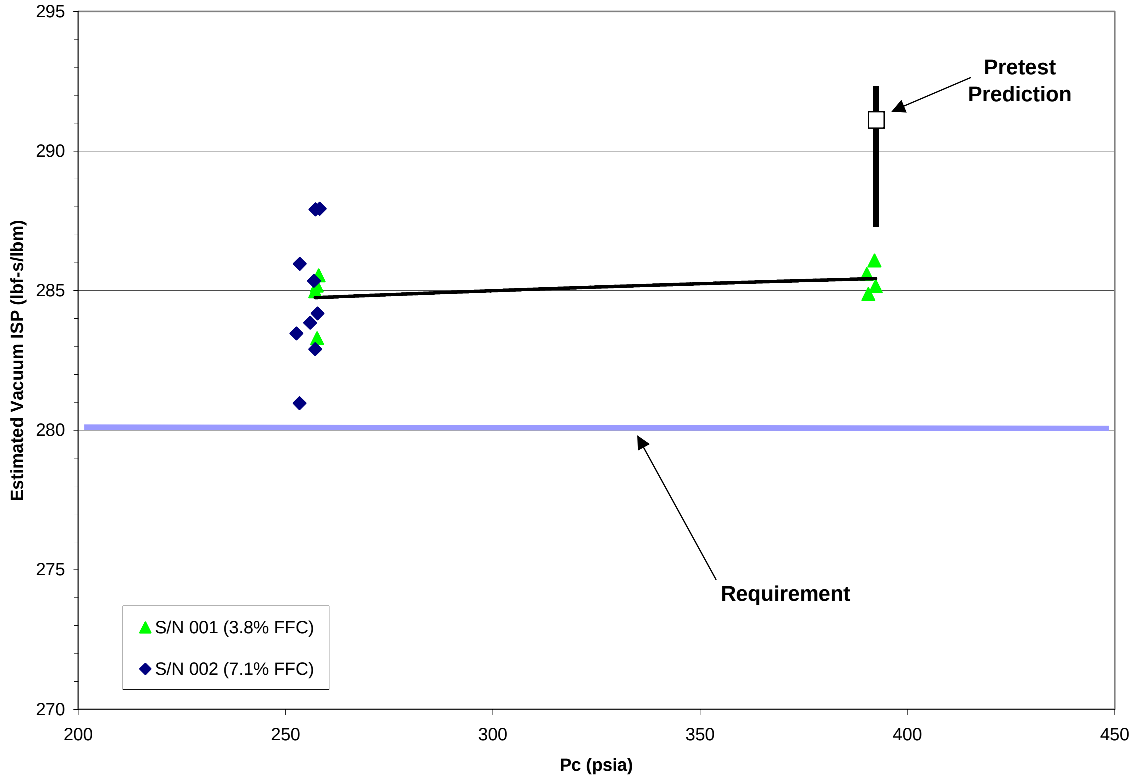


Figure 11: **Trends of Delivered ISP with Chamber Pressure and FFC Percentage**
Pretest ISP Prediction with Dispersions and Requirement value (280 lb_f-s/lb_m) included for reference.

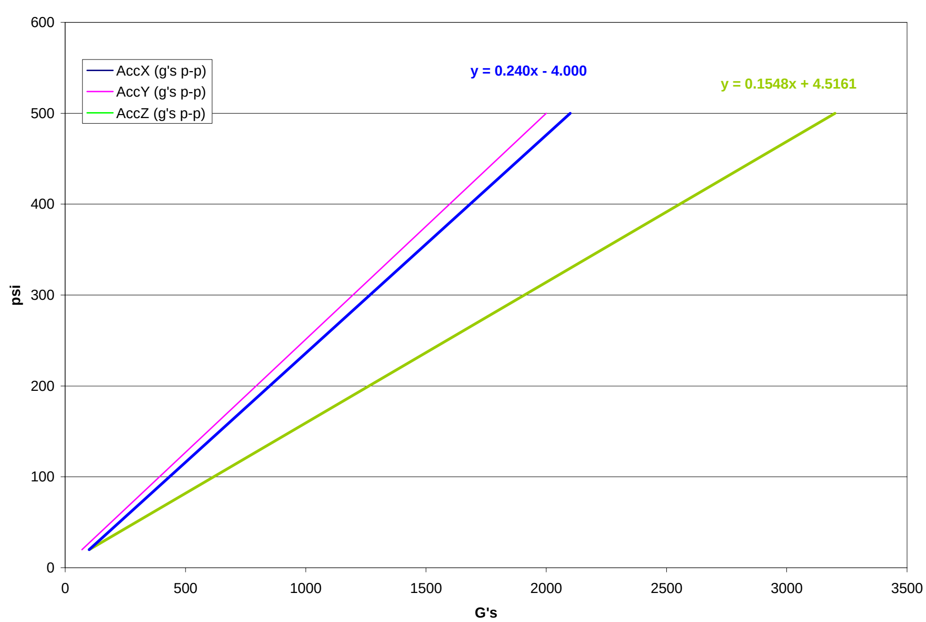


Figure 12: Amplitude Correlation of Vibration Level to Chamber Pressure

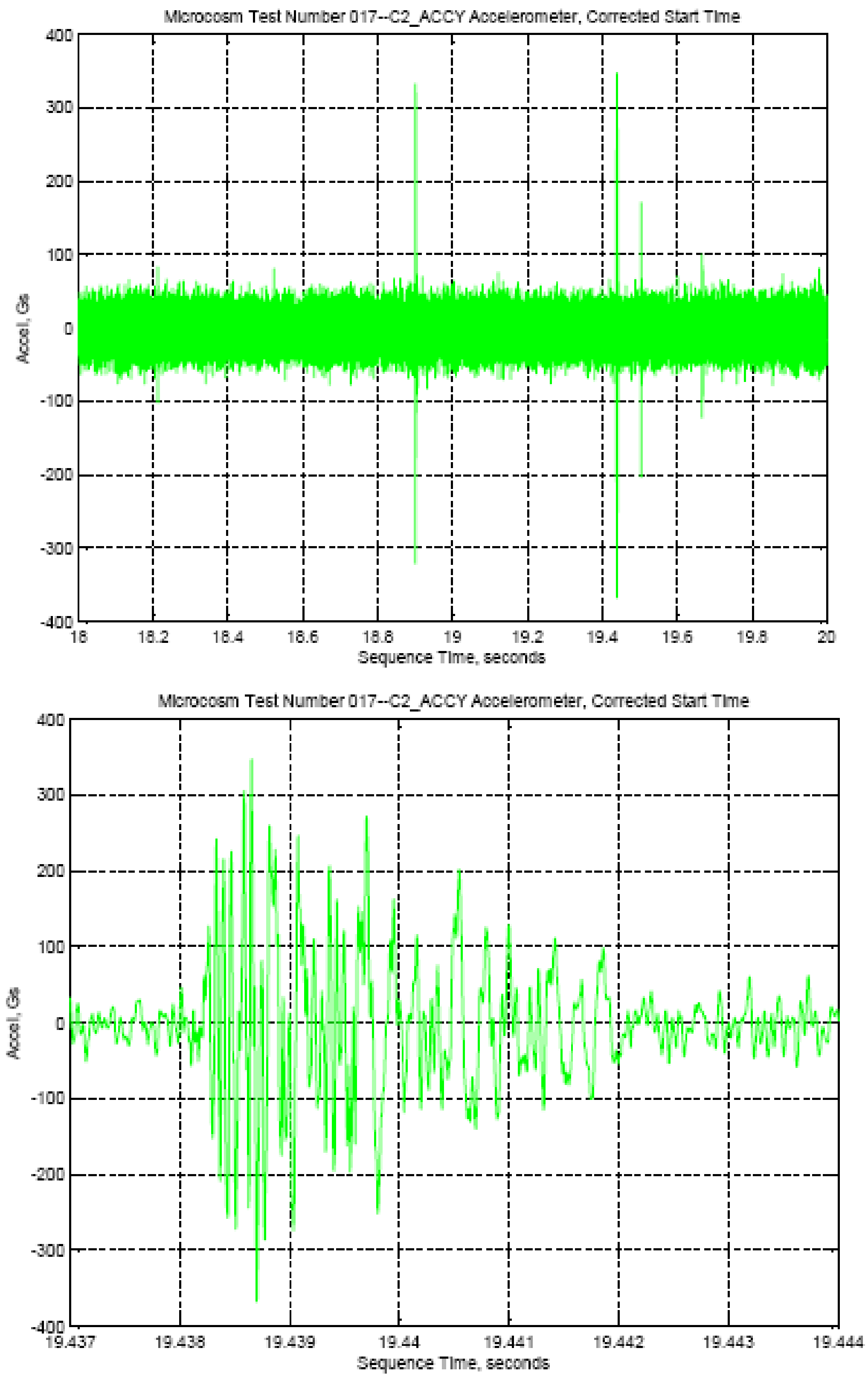
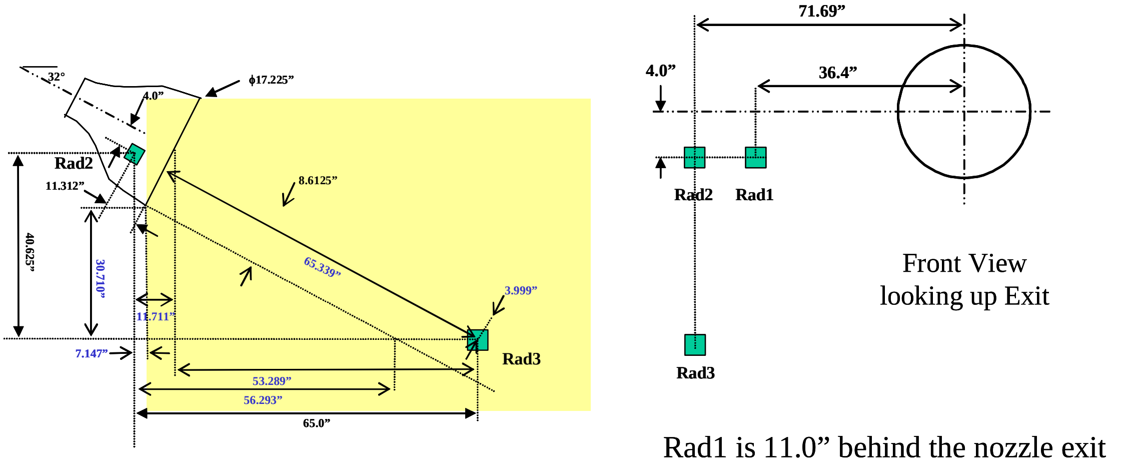


Figure 13: Test 17 Y-Accelerometer Output Showing Mean output (T) and Pop Detail (B).



Rad1 is 11.0” behind the nozzle exit
Rad2 is 11.312” behind the nozzle exit

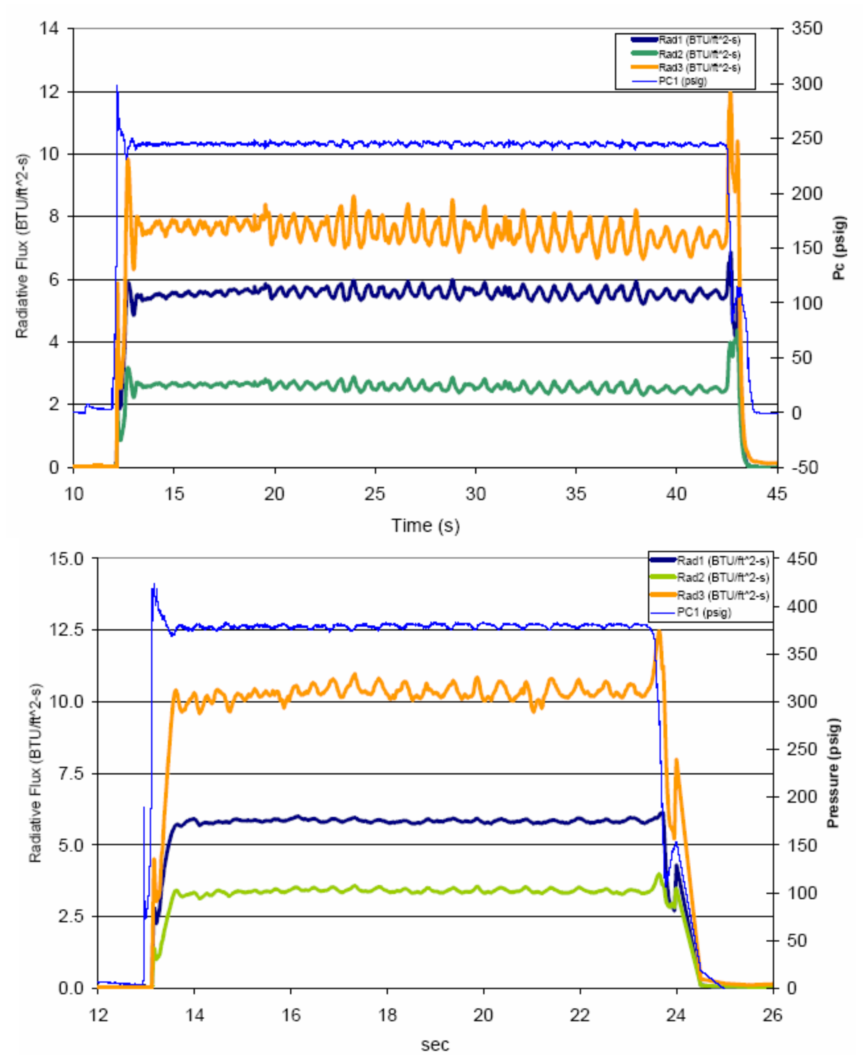


Figure 14:Radiometer Setup (T) and Data for Tests 17 (M) and 19 (B)