Development of a Five-Pound Thrust Bipropellant Engine

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The subject program advanced the technology for high-performance, rapid response, long-lived bipropellant engines in the 5-lb. thrust class. The results of design and development activities and the fire testing of fast response (0.005 sec), high-performance (300 sec I_{sp}), 5 lbF bipropellant (N_2O_4/MMH) engines to obtain performance and life durability data are presented. Data were obtained from over 17,000 sec of firing duration and over 400,000 engine starts. Steady-state and pulsing performance, and engine response data were obtained over a range of tank pressures and at environmental temperatures ranging from 20-120°F. Thermal characteristics and heat rejection rates were experimentally evaluated as injector and chamber designs were varied. Simulated duty cycles were demonstrated with three engines. These data were utilized in the forecast of engine life and reliability and the assessment of exhaust plume contamination.

Background

PRIOR to 1963, few spacecraft missions required the injection of a payload into orbit. Furthermore, there was little need or room for an onboard propulsion system as part of an orbiting package. During the period 1963 to 1968, payload weights increased and a need for station-keeping developed as mission goals became more ambitious. Reaction control systems employed catalytic decomposition of hydrogen peroxide and/or cold gas jets. The instability of the hydrogen peroxide under storage and the need for pressure relief valves made the reliability of this system inherently low. Cold gas systems, although much more reliable, provided very low performance.

Monopropellant reaction control systems utilizing hydrazine were evaluated for station-keeping missions starting in 1967. By 1973, such systems enjoyed an undisputed industry acceptance and had performed well in a wide range of applications. However, the demands being placed on these reaction control systems are becoming more and more stringent and there are indications that the requirements may soon exceed monopropellant system capabilities. Future military space missions, such as space defense and reconnaissance, are likely to have requirements in excess of those of the commercial systems, itemized in Table 1, which typically illustrates the evolution to these new demands.[1]

A large number of engine cold starts and very high total impulse have been shown to degrade the response, repeatability, and performance of monopropellant thrusters. This is caused by gradual degradation of the catalyst bed. Figure 1, taken from Ref. 1, summarizes the demonstrated capabilities of hydrazine monopropellant engines in 1973. The cold start quantity of less than 700 for 5-lbF class engines is not adequate for anticipated missions and illustrates a need for technology improvement. Efforts to correct this limitation by using bed heaters and improved catalysts and bed designs are in progress and have shown limited success.

A bipropellant system is a logical advance in technology which eliminates the problems associated with catalyst beds and monopropellant systems. Storable bipropellants, such as N_2O_4/MMH, have long been employed in larger engines (greater than 20-lbF), performing reaction control functions with considerable success and a high degree of reliability. In addition to the 25% improvement in steady-state specific impulse performance over monopropellant systems, bipropellant systems offer a potential for the following: 1) longer life and nearly unlimited thermal cycling with the performance decrement and impulse variation attendant to catalyst bed degradation entirely eliminated; 2) higher pulse mode performance with particular performance advantages obtained on cold starts; 3) more predictable response over the engine life and lower power consumption resulting from the elimination of catalyst bed heaters; 4) lower propellant freezing temperatures; and 5) improved handling and reliability resulting from the elimination of a catalyst bed which is sensitive to contamination and damage.

In some applications, the use of bipropellant engines allows the attitude control system to be integrated with the propellant feed system of the larger bipropellant engines onboard the spacecraft. This results in a system weight advantage which is additive to the performance advantage.

Those areas which have historically proved troublesome to small bipropellant engines were addressed in the technology work reported herein. These included the following: 1) poor combustion efficiency and performance due to the very low propellant flowrates and limited number of injection elements (usually 2 or 3 orifices); 2) failure to achieve uniform propellant combustion which is free from wall-damaging hot streaks; 3) inadequate nozzle cooling and unacceptable heat soaks over a wide range of duty cycles; 4) a relatively large dribble volume resulting in residual propellants which degrade performance, aggravate ignition spike problems, and increase plume contamination levels; and 5) exhaust plume contamination resulting from ejection of propellant droplets caused by incomplete combustion.

Technology Goals

The goals of this effort to advance the state-of-the-art were to develop and demonstrate the technology required to provide a high-performance, long-lived, fast response, 5-lb thrust bipropellant engine capability for the future Air Force requirements indicated in Table 2. The propellants employed in the demonstration were nitrogen tetroxide (N_2O_4) and monomethyl hydrazine (MMH).

Most noteworthy are the 300 sec steady-state specific impulse and 240 sec pulsing specific inpulse at impulse bits of 0.05 lb-sec, the 3:1 tank pressure ratio for a blowdown system, the 20-120°F propellant supply temperature range, and the general life and reliability requirements. Additional

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requirements which resulted from a survey of spacecraft manufacturers were a need for buried engine operating capabilities (adiabatic wall) and a limitation of the engine-spacecraft thermal coupling.

Design Approach

Analyses indicated two parameters were essential to the attainment of nearly all of the pulsing and steady-state performance goals and proper chamber cooling. These were a multi-element axisymmetric injector pattern and a very small volume of residual propellants downstream of the valve seats. Parametric analysis defining the relationship between element quantity, chamber length, and contraction ratio (Fig. 2) revealed that no fewer than 3 unlike-doublet type elements would be required to attain the 300 sec steady-state specific impulse goal. The use of 4-6 such elements would allow a shorter chamber length and more uniform nonstreaking flowfield.

Further analyses and experimentation, summarized in Fig. 3, revealed the total residual propellant volume downstream of the valve seats should be in the 0.0005-0.0008 in.^{3} range if the 240 sec pulsing specific impulse goal was to be attained at a 0.05 lbF-sec bit impulse level. By comparison, the residual propellant volume of the 5-lb thrust R6C engine noted in Fig. 3 is 0.0017 in.^{3}.[3,4] This volume limitation dictated that the valve and multi-element injector be an integral device. This, in turn, required that the thermal standoff conventionally located between the valve and injector be moved to the chamber to valve interface as shown schematically in Fig. 4, and the injector be self cooling.

Optimization of the chamber pressure on the basis of the requirements for 3:1 tank blowdown, maintaining subcooled liquid propellants within the injector orifices at 120°F and reliable hypergolic ignition at 20°F, indicated that all requirements were more readily achieved at higher pressures. Higher chamber pressure also improves performance and minimizes contaminate generation, as shown in Fig. 5. This was in contrast to the requirements of chamber cooling which showed higher pressures to have a mildly adverse influence. This, however, could be overcome by the generation of a lower temperature fuel rich thermal barrier environment at the chamber wall.

The criteria for the optimized engine design resulted in a tank blowdown from 300-100 psia with a corresponding chamber pressure and thrust decay of 170 to 75 psia and 5.0 to 2.2 lbF, respectively. Figure 6 shows the tank pressure vs thrust and chamber pressure relation of one of the engines fire tested.

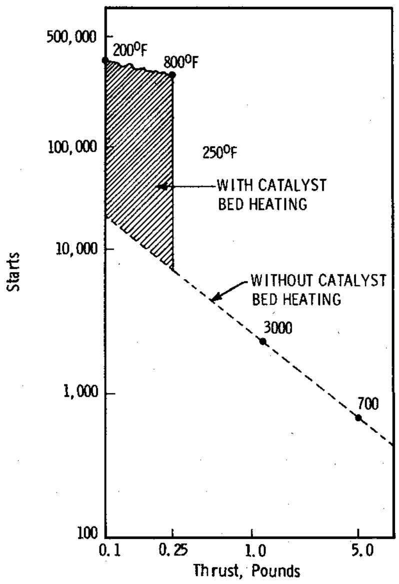

Fig. 1 Demonstrated monopropellant thrust capabilities.

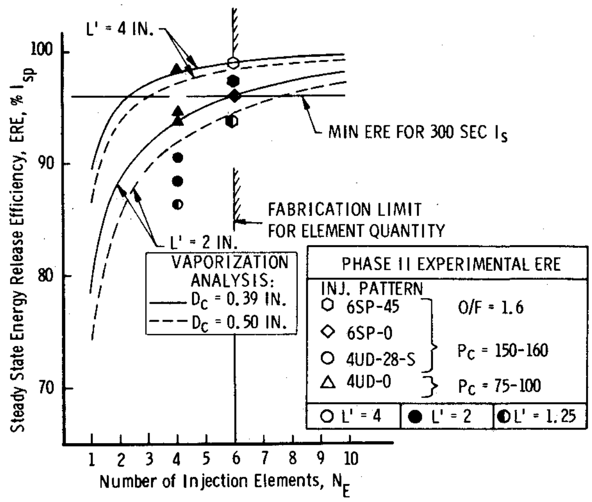

Fig. 2 Element quantity and chamber interaction.

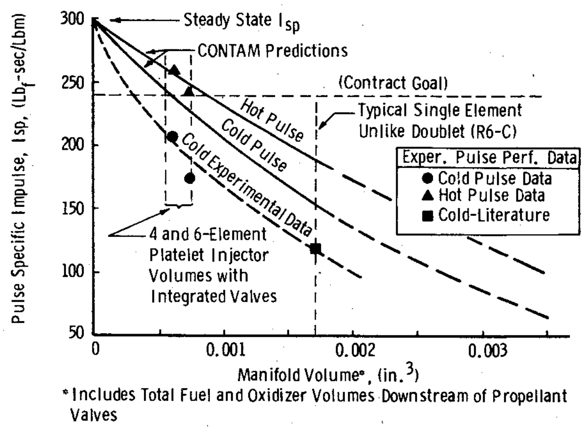

Fig. 3 Manifold volume effect on pulse performance at 0.05 lb_f-sec minimum bit impulse (EP_w = 0.010 sec).

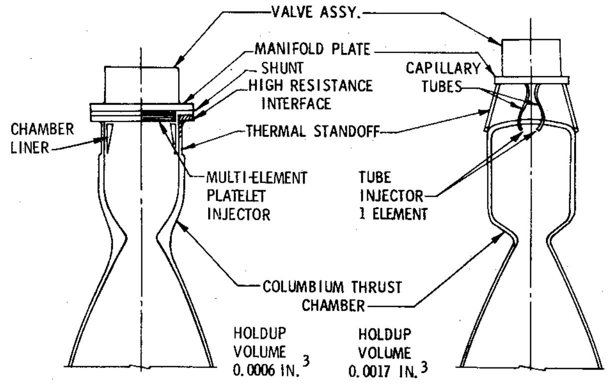

Fig. 4 Comparison of low residual volume and conventional engine designs.

Engine Design

Two 5-lbF thrust class engines, designated AJ 10-181-1 and -2, whose designs were based on the CONTAM computer model[4] and on experimental verification data, were fabricated and subjected to simulated duty cycle testing. The -181-1 is barrier-cooled and capable of operating for unlimited duration in a buried installation; the -181-2 is radiation-cooled and of slightly higher performance. Each utilizes a multi-element injector (6 elements in the -2 design, and 4 elements in the -1 design) integrated with a flight qualified torque-motor actuated linked bipropellant valve manufactured by Moog, Inc. Linked actuation assures synchronous valve operation over the life of the engine, thus assuring maximum pulsing performance and eliminating plume contaminants which could otherwise result from timing drifts. The two engines tested had total manifold volumes (dribble volume) of 0.00054 and 0.00065 in.[3], respectively. The volume was evenly divided between propellant circuits. This assured very fast response and high pulse mode performance with a minimum of plume contaminate generation.

The multi-element injection patterns provided axisymmetric mass and mixture ratio distributions tailored for each of the particular applications. This was demonstrated by cold flow testing in which an element to element flow uniformity of $\pm 6\%$ for the 4-element injectors and $\pm 12\%$ for the 6-element injectors was measured. This results in excellent injector-chamber compatibility to insure a long-lived, non-duty-cycle-limited, thrust chamber capability.

The thrust chambers demonstrated on both engines were mechanically attached, silicide coated, columbium alloy (FS85) units which provided a 50:1 nozzle area ratio and had a weight of 0.18 lb. The weight of a complete engine consisting of valve, injector, and thrust chamber is 1.2 lb. The use of 100:1 and 150:1 area ratio nozzles has negligible impact on overall weight and adds 7 and 9 sec of specific impulse to the designs tested.

In conventional engine designs, the normally hot injector and thrust chamber are thermally uncoupled from the valve by use of a high-resistance mount and small diameter, thin wall tubes which transport propellants from the valve to the injector. A typical tube contains a volume which is considerably larger than the 0.0003 in.[3] single circuit volume for 4- and 6-element designs required to attain the goals of an advanced engine.

The elimination of the tubular stand-offs required that the injector be propellant cooled and that a series of high-thermal-resistance paths be provided between the chamber and valve-injector assembly. The first such resistor in the thermal path from chamber to valve was the thrust chamber itself. The chamber is thinned to a 0.017 in. wall for a 0.50 in. length such that the thermal coupling of the flange and hot portion of the chamber is minimized. The thin wall zone is shielded from the combustion gases by a coated columbium flame liner which is cooled by the secondary impingement of unvaporized propellant.

The chamber flange was further isolated from the valve by a thermal dam which consisted of multiple 0.001 in. thick stainless steel laminated shims designed for minimal contact area. The limited heat which penetrated this thermal barrier was transferred to a copper shunt which conveyed the heat to a suitable sink precluding heat flow to the valve. The maximum valve temperature acceptable was defined by the saturation temperature of the oxidizer at the lowest expected feed pressure. Figures 7 and 8 are schematic presentations of the engine design.

Operational Characteristics

Response

Figure 9 provides two typical successive cold engine thrust traces with each pulse delivering 0.05 lb-sec impulse. The response data reported in Table 3 were obtained over an operational box bounded by tank pressures of 300-100 psia and temperatures of 20-120°F. The valve response of 0.0023-0.0026 sec from signal to start of valve travel (at 28v) showed an insensitivity to variances of both temperature and pressure. The engine response was 0.0056 ± 0.0006 sec from start signal to 90% thrust at all operating conditions. The ± 0.0006 sec variance is due mainly to the variation of tank supply pressures in the blowdown system. Thus, with full tanks and accompanying high supply pressure (300 psia), response is 0.005 sec. This increases to 0.0062 sec when the tank pressure is decreased to 100 psia. The pulse-to-pulse response repeatability of these data is within ± 0.0002 sec for a given set of operating conditions. Response data were obtained with

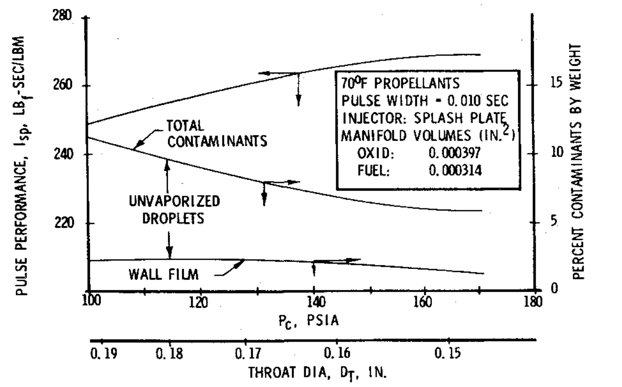

Fig. 5 Nozzle throat diameter and chamber pressure effect upon pulse performance and contamination.

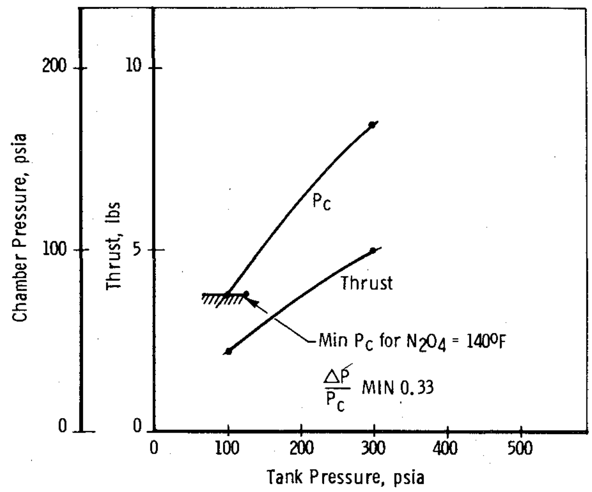

Fig. 6 Blowdown mode operation.

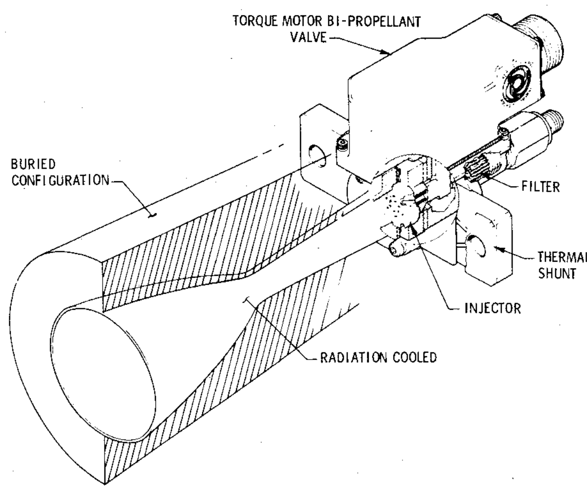

Fig. 7 5-lb thrust bipropellant engine schematic.

three different valves of the same design and revealed a variance between valves of less then ± 0.00035 sec.

Minimum Impulse Bit (MIB) and Repeatability

The minimum electrical pulse demonstrated in engine firings was 0.010 sec although the valve provided a capability to respond to electrical pulses as small as 0.003 sec. The 0.010 sec pulse results in a nominal 0.05 lbF-sec impulse bit with the tanks fully pressurized at 300 psia. The impulse bits demonstrated when the tank pressure decayed to 100 psia were nominally 0.02 lbF-sec. These were found repeatable within

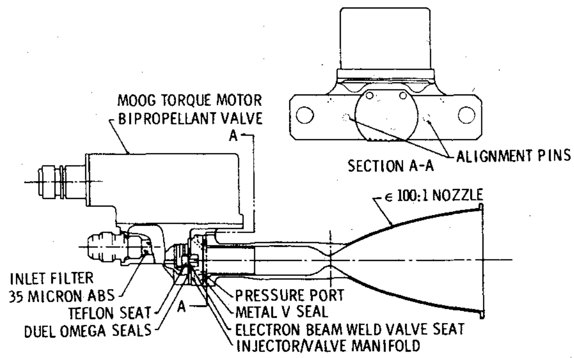

Fig. 8 5-lb thrust bipropellant engine assembly; __first pulse, --- second pulse; 0.3 sec between firings.

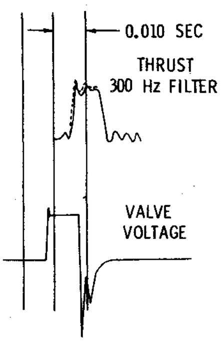

Fig. 9 Typical successive cold engine thrust traces.

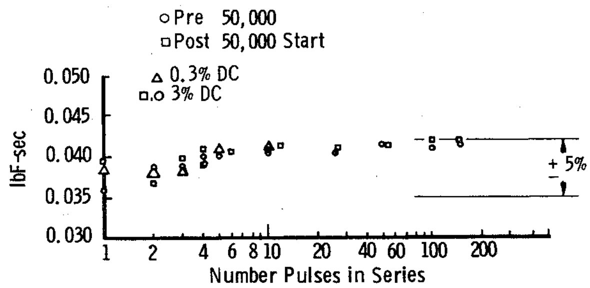

Fig. 10 Bit impulse repeatability, AJ10-181-1.

±3% in long pulse trains. The potential MIB capability of this design is in the 0.005-0.01 lbF-sec range. Table 4 provides statistical data showing the repeatability of individual 0.010 sec pulses at the minimum impulse bit (MIB) demonstrated for the AJ 10-181-1 engine. This table compares the bit impulse of a new engine with that obtained following a 50,000 restart life simulation test, and also compares a 3.0% duty cycle with a 0.3% duty cycle. Each pulse train starts with the engine cold. Since the measurement accuracy for individual pulses is approximately ±2%, more meaningful data are obtained considering groupings of ten pulses. The impulse bit performance shown by these data is noted to be highly repeatable over the life of the engine and independent of duty cycle. The same data are shown graphically in Fig. 10.

Table 1 Evolution of communication satellite thruster requirements

Parameter	SYNCOM	ATS-4	INTELSAT IV A	Advanced Spinner	Advanced Spinner
Start Quantity	100	50	700	4,000	40,000
Total Impulse, lb-sec	6,000	10,000	72,000	75,000	75,000
Predictability % (Total Error)	± 40	± 30	± 20	± 15	± 15
Life in Orbit, Years	1	3	7	10	10
Propellant System	H_2O_2	N_2H_4	N_2H_4	TBD	TBD

Table 2 Projected requirements for future 5-lb thrust bipropellant engines

Parameter	Blowdown or Regulated Feed System
Maximum Vacuum Thrust, lbf	5.0 (nominal)
Chamber Pressure, psia	170 to 75
Feed System Pressure, psia	3 to 1 Blowdown Range
Expansion Ratio	100 to 150
Minimum I_{sp} (at max thrust), sec	
Steady State	300
Pulsing	240
Minimum Impulse Bit, lbf-sec	0.05 ± 0.005
Total Impulse Delivery Capability lbf-sec	30,000 to 100,000
Number of Ambient Starts	100 to 1,000
Total Number of Restarts	175,000 to 300,000
Total Firing Life	2 to 10 hr
Total Mission Life	5 days to 10 years
Valve Response, ms	
Signal to full open	< 5
Signal to full close	< 5
Valve Leakage, scc/hr	
GN_2 at ΔP = Feed System Pressure	< 2.5
Mixture Ratio	1.6 (nominal)
Propellant Inlet Temperature, °F	20 to 120
Storage Life Goal	10 years
Flightweight TCA Reliability Goal	0.999
Flightweight TCA Maintainability Goal	Zero Maintenance Over Storage Life

Table 3 Typical engine response characteristics

	Propellant Supply Condition			
Tank Pressure, psia	300-400		100-150	
Propellant Temp, °F	22	118	22	18
Start Signal to 90% P_c sec	0.0051	0.0050	0.0061	0.0062
Stop Signal to 10% P_c sec	0.0055	0.0052	0.0075	0.0065
Signal to Valve Open sec	0.0026	0.0025	0.0023	0.0025
Signal to Valve Close sec	0.0025	0.0027	0.0028	0.0025
Valve Travel Open sec	≈0.0005	≈0.0005	≈0.0005	≈0.0005
Valve Travel Close sec	≈0.0005	≈0.0005	≈0.0005	≈0.0005
P_c Decay sec	0.0025	0.0025	0.0047	0.0040

The major factor influencing the bit impulse variation is the heating of the chamber wall during the first 10 pulses. Figure 10 shows this variable to be highly repeatable. Pulses 11-20 and 100-109 are noted to have standard deviations in the 0.5-1.5% range; thus, with proper allowance for temperature effects, impulse bits should be predictable within a few percent.

The one sigma repeatability for a 100 pulse sample size including the first 10 pulses relative to impulse bit size is shown in Table 5.

Engine impulse linearity as a function of electrical signal is shown in Fig. 11. The ratio of impulse to fire duration for 11 tests of 0.010-0.10 sec duration resulted in a coefficient of 4.188 with a standard deviation of 1.45%.

Performance – Steady State and Blowdown

During the initial hot-fire injector and chamber length evaluations, 6-element injectors optimized for maximum performance and tested in 4-in. long chambers provided steady-state specific impulses of 312-314 sec when the test data was extrapolated to a nozzle expansion ratio of 100:1. High chamber heat loads at this performance level limited steady-state firing durations to less than 10 sec.

Table 4 AJ10-181-2 bit impulse repeatability. A = predurability test data, 3.0% duty cycle 0.010 sec pulse; B = post 50,000 pulse data, 3.0% duty cycle 0.010 sec pulse; C = Predurability data, 0.3% duty cycle 0.010 sec pulse

Pulse No.	A	B	C	Pulse No.	A	B	Pulse No.	A	B
1	0.0361	0.0397	0.0386	11	0.0421	0.0419	100	0.0411	0.0419
2	0.0386	0.0369	0.0386	12	0.0413	0.0411	101	0.0417	0.0417
3	0.0389	0.0403	0.0387	13	0.0410	0.0425	102	0.0413	0.0415
4	0.0400	0.0408	0.0391	14	0.0412	0.0420	103	0.0415	0.0414
5	0.0402	0.0406	0.0410	15	0.0423	0.0410	104	0.0417	0.0416
6	0.0395	0.0407	0.0413	16	0.0415	0.0414	105	0.0417	0.0419
7	0.0421	0.0436	0.0416	17	0.0419	0.0422	106	0.0418	0.0417
8	0.0403	0.0430	0.0416	18	0.0402	0.0426	107	0.0413	0.0421
9	0.0409	0.0428	0.0400	19	0.0421	0.0424	108	0.0409	0.0416
10	0.0408	0.0418	0.0409	20	0.0419	0.0419	109	0.0413	0.0416
Mean	0.0397	0.0410	0.0401		0.0416	0.0419		0.0414	0.0417
Standard Deviation	0.00163	0.00194	0.00128		0.00064	0.00057		0.00029	0.00021
% Deviation	4.11	4.73	3.19		1.54	1.35		0.72	0.51
Mean		0.0403			0.0418			0.0416	
Standard Deviation		0.00066			0.00021			0.00021	
% Deviation		1.65			0.51			0.51	

Table 5 One sigma repeatability for a 100 pulse sample

Impulse (lbF-sec)	1σ Repeatability
0.400	$\pm 1\%$
0.100	$\pm 2\%$
0.04	$\pm 2.4\%$
0.02	$\pm 3.0\%$

In the subsequent engine design optimization the chamber length was reduced to 2 in. and the oxidizer spray angle modified to provide a fuel rich environment at the chamber wall. This resulted in the demonstration of a 300 sec (@100:1) specific impulse engine (AJ 10-181-2) which has the capability of operating for long durations in the radiation cooled mode. A universal design (AJ 10-181-1) having the capability of unlimited duty cycles in the buried mode (adiabatic wall) was achieved by further increasing the barrier cooling. The specific impulse of this engine is 283 sec @100:1.

The evaluation of these designs under simulated tank blowdown conditions provided the data shown in Fig. 12. Each design can provide its maximum performance over the full life of a mission if a pressure regulated system is employed. The same figure also shows data obtained from several injector patterns which did not meet the goals of the 5-lbF class engine due to cooling problems at high chamber pressures, but did exhibit very high performance and good durability when throttled to ≈ 100 psia and a nominal 2.5 lbF thrust level. Further development work based on these injector designs could result in an improvement in the blowdown performance of the -1 and -2 engine designs.

Performance – Pulse Mode

Pulsing performance was evaluated at electrical pulse widths (EPW) of 0.010, 0.025, 0.050, and 0.100 sec. Performance values were calculated using data from the first 150 pulses of each pulse train. The pulsing frequency was 3 pulses/sec. Two major factors influencing pulsing performance are pulse length and chamber wall temperature. The shorter pulses are less efficient due to the fixed startup and shutdown losses relative to the burn time. Conversely, a hot chamber wall aids in propellant vaporization and suppresses the losses due to wall film. The wall film losses are significantly suppressed when the wall temperatures exceed the fuel saturation temperature of ≈ 375°F.

Figure 13 provides pulsing performance data for the -2 radiation cooled engine. The cold chamber pulse performance is based on the average of the first 4 pulses.

The items of major significance in these data are: 1) the goal of a 240 sec specific impulse for a 0.05 lbF-sec impulse bit was attained; the measured 246 sec I_{sp} provides a 6-sec margin; 2) there was no change in specific impulse over the

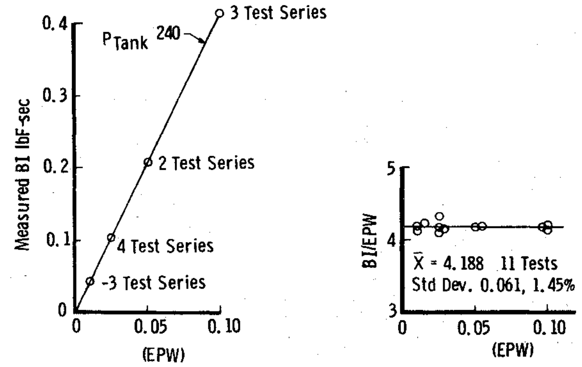

Fig. 11 Linearity and repeatability of engine and test facility.

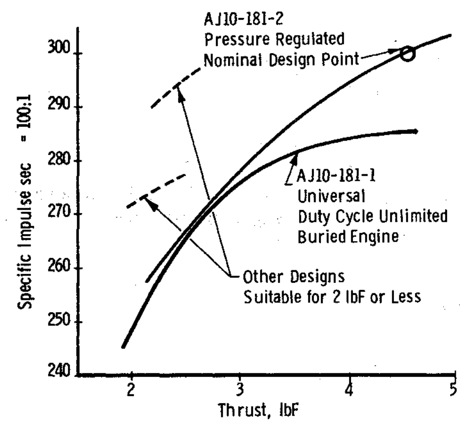

Fig. 12 Comparison of blowdown performance capabilities.

300,000 restarts of the engine; 3) environmental temperatures ranging from 20 to 120°F had no significant influence on steady-state performance and response. The merging of the data for hot and cold pulses at the larger impulse values is a result of the heating of the chamber during the longer duration pulses.

Figure 14 shows comparable data for the lower performing (-181-1) adiabatic wall engine. There was no variation in performance during or following the 50,000 restarts and 6300-sec continuous burn accomplished with this design. This engine is capable of operation in both radiation cooled and adiabatic wall modes. Operation in the radiation cooled mode results in lower wall temperatures which tend to slightly reduce the pulse mode performance.

Thermal Characteristics

Steady-State Firings

Temperature profiles along the axis of the engine, starting at the thrust mount and extending through the combustion chamber and into the supersonic nozzle, are shown in Figs. 15 and 16 for the (-181-2) radiation cooled and (-181-1) buried, barrier cooled designs. The low-flange temperature followed by the rapid rise to the midpoint of the 2 in. combustion chamber is the result of the chamber's thermal liner, the thin wall at the chamber's forward end, and the thermal shunt acting in concert to property manage the heat flow to provide a highly efficient hot wall for vaporization without excessive heat input to the valve or sink. The heat rejection through the shunt ranges from 11 watts at the 100 psi pressure level to 75 watts at the highest operating pressures.

The shapes of the temperature profile plots are explained by the impact of unvaporized fuel on the convergent nozzle, the partial mixing of the fuel rich vapors passing through the throat, and the subsequent expansion process in the nozzle. The large increase in temperature of the -181-2 engine between 105 and 125 psia is a result of a change from blow apart to

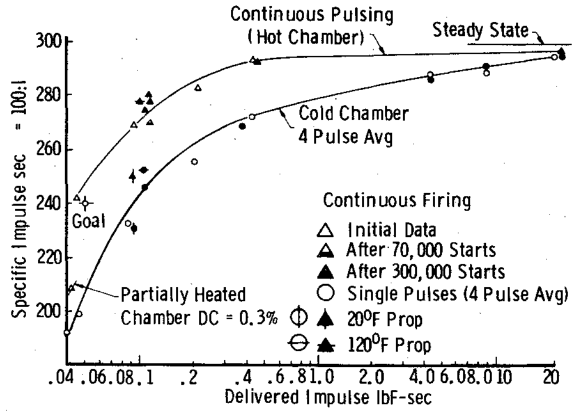

Fig. 13 AJ10-181-2 engine radiation cooled regulated pulsing performance evaluation.

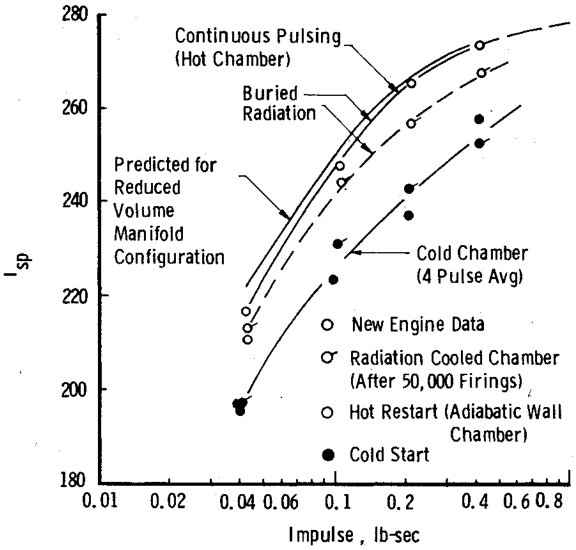

Fig. 14 AJ10-181-1 engine pulsing performance.

spray penetration of the propellant flow of the unlike-doublet type elements. The significantly lower wall temperatures can be attained for only a small loss in performance as shown in Fig. 17. The -181-1 design does not exhibit this effect and is thus highly suitable to blowdown operation.

The small mechanical loads due to pressure and thrust and the low stresses resulting from thermal gradients result in the FS 85 columbium alloy providing a large structural design margin at temperatures as high as 3200°F. A 3000° wall translates to a stress rupture lift of approximately 2400 hr.

The life of the columbium chamber is, however, limited by the allowable operating temperatures of the $CbSi_2$ and Cb_5Si_3 compounds which provide the required oxidation protection for the base material. A typical requirement for a 100,000 lbf-sec total impulse capability for a single engine dictates a nominal 5.5 hr at maximum temperature conditions. If a safety factor of 4 is applied to this duration, i.e., a 22 hr operating requirement, the maximim operating temperature should be limited to 2700°F. The use of a tantalum base alloy could add approximately 400°F to the allowable temperatures and allow a nominal 5-sec specific impulse improvement. Increasing the nozzle area ratio from its nominal value of 100 to a 150-200 range also provides several seconds performance improvement.

Pulsing Operation

Each engine was subjected to pulse mode duty cycles of 0.3, 3, 7, 10, 15, and 30% to obtain the equilibrium temperature of each major component; no limiting operating conditions

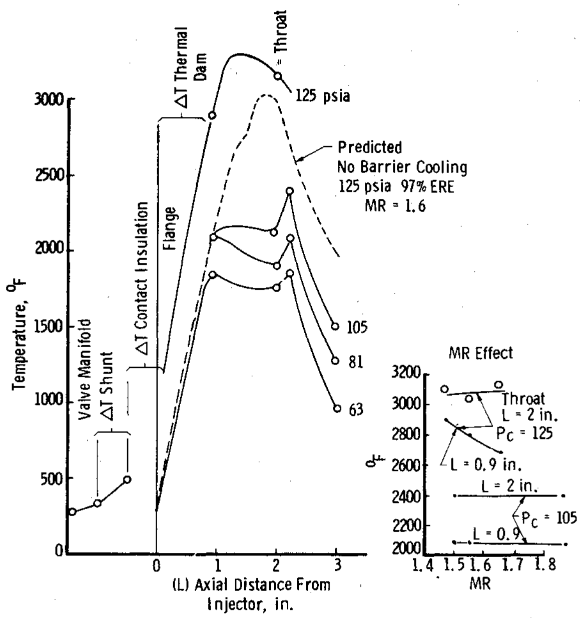

Fig. 15 AJ10-181-2 engine axial temperature profiles.

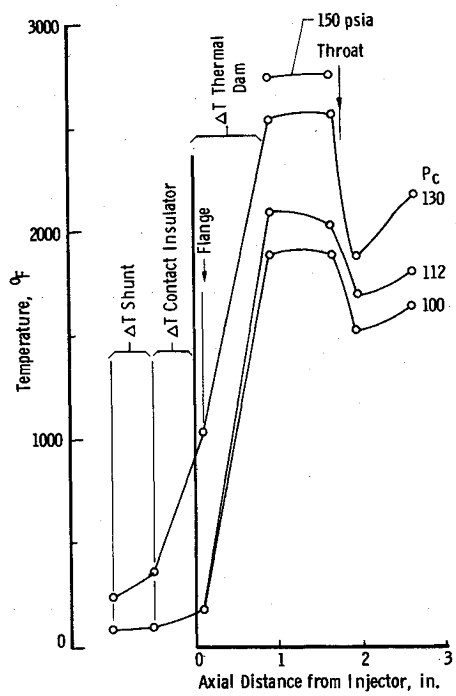

Fig. 16 AJ10-181-1 engine axial temperature profiles.

were encountered over this duty cycle range. Data from these tests, shown in Figs. 18 and 19, demonstrate that steady-state operation results in the most adverse thermal environment on all components of the radiation cooled design and most components of the buried engine with the exception of the injector face and valve manifold. The latter temperatures peak out at approximately 300°F in the 15-30% duty cycle range. This temperature does not impact response and is favorable for performance. The 300°F maximum operating temperature of the stainless steel injectors assures a virtually unlimited firing and restart capability.

Operation at duty cycles less than 100% extends the life of the silicide coated columbium chamber by lowering the chamber temperature. For example, at a 30% duty cycle the maximum temperature of the buried engine (AJ-10-181-1) is 2100°F compared to 2550°F @ 100%. At 30% on-time, it would require 73 hours of continuous pulsing to deliver

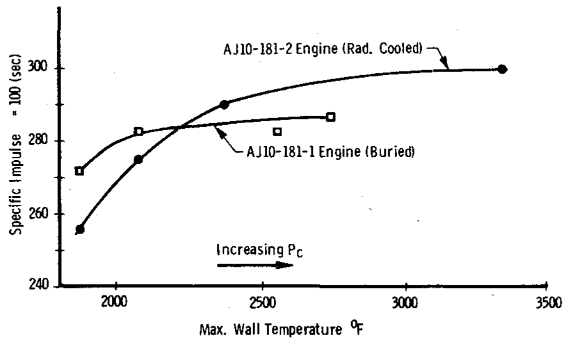

Fig. 17 Continuous pulsing duty cycle thermal characteristics, AJ10-181-2 regulated-radiation cooled engine.

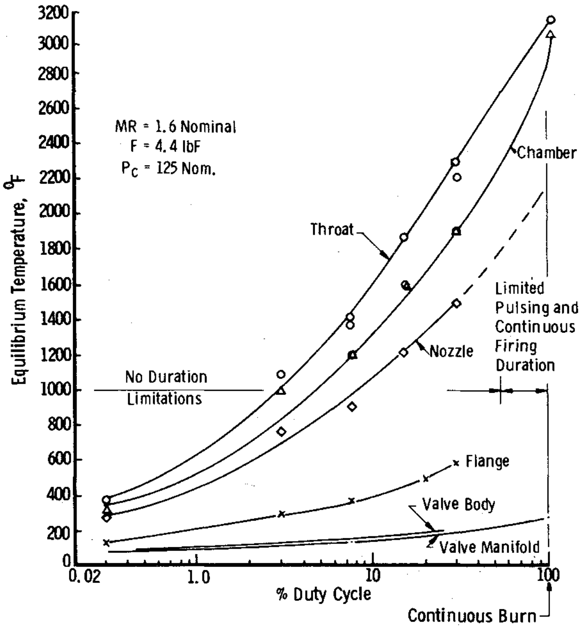

Fig. 18 Continuous pulsing duty cycle thermal characteristics, AJ10-181-1 buried blowdown engine.

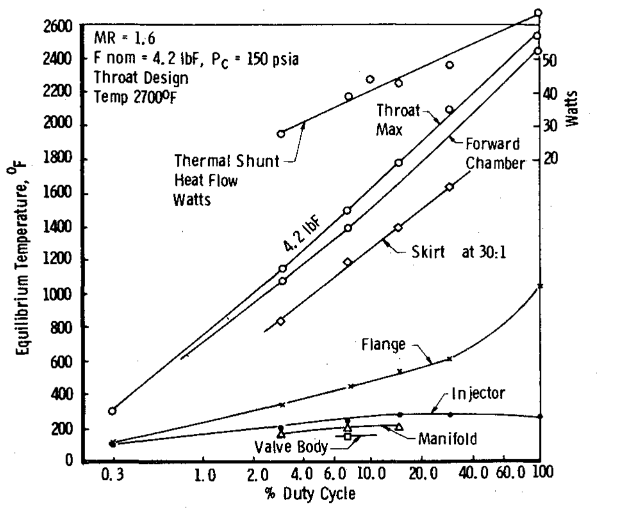

Fig. 19 Continuoug pulsing duty cycle thermal characteristics, AJ10-181-1 buried blowdown engine.

100,000 lbF-sec impulse. This compares to an estimated useful coating life of over 200 hr using the exposure-to-air data which are conservative. Below temperatures of 800°F, the columbium has sufficient oxidation resistance to operate without coating. This corresponds to duty cycles less than 1.0%.

A linear cumulative damage analysis predicts each engine to have a capability of in excess of 10^7 firings of 5 sec duration or less. The number of full thermal cycles, consisting of firings greater than 5 sec, is 10^6 while passing approximately 1000 lb of propellant.

Conclusions

A large improvement over the characteristics of current monopropellant and bipropellant engines in the 5-lbF thrust class has been demonstrated. This low dribble volume multi-element injector bipropellant engine offers: 1) unlimited duty cycle capability and practically unlimited accumulated and steady-state firing life (200 hr) without loss in performance; 2) rapid response, 0.0056 sec from signal to 90% thrust, without a need for thermal conditioning; 3) precise and highly repeatable impulse bits with nearly square wave thrust-time characteristics; 0.018 lbF-sec impulse bits were repeatable within $\pm 3\%$ and capabilities down to ≈ 0.005 lbF-sec are projected; 4) a delivered specific impulse (area ratio 100:1) of 300 sec where the engine installation allows radiation cooling or 283 sec for a fully insulated buried configuration; 5) pulse mode specific impulse in excess of 240 sec at bit impulse values down to 0.05 sec; and 6) very low plume contamination.[5]

Appendix: Recommendations

The technology demonstrated under this contract suggests the following avenues for further development and test evaluation.

Engine Durability Evaluation

The ultimate limitations of the materials utilized in the demonstrated 5-lb thrust engines could be established in a fire-to-destruction effort. This is estimated to require on the order of 10^6 full heating and cooling cycles while accumulating 20 hr burn time or between 200 and 2000 hr of continuous operation at full thrust with very few thermal cycles. The data obtained demonstrating the durability of the coated columbium alone would be highly useful to any of the long-lived columbium component engines now in development.

Engine Scaling Demonstration

Data obtained in the course of testing showed the unique characteristics of certain 4-element injector designs resulted in the deliverance of high performance down to the 2 lb thrust level. It is anticipated that a 1 lbF bipropellant engine which has nearly the same level of performance but a minimum impulse bit capability of a 0.005 lb-sec could be developed using the technology obtained on this program. Conversely, the same valve and injector integration approach can be employed on larger engines.

Propellant Change-over

Conversation with spacecraft primes indicates that the adaptation of the engine design to a N_2H_4/N_2O_4 propellant system would allow bipropellant engines in the 1.0-25.0 lbF range to be operated with very small (0.1-0.5 lbF) monopropellant engines which are fed from a common fuel system. The commonality of the fuel would lead to improved spacecraft propellant utilization, system simplification, and weight reduction.

Improved Engine Thermal Isolation

The engine design developed cannot be divorced from the spacecraft heat balance. Depending upon installation and duty cycle, there can be a heat flow of 10-75 watts to the spacecraft while the engine is firing. Provision can be made to transfer this heat to the flowing propellants or to have the heat flow substantially reduced or eliminated through an improvement in the chamber to valve interface design. A major advantage would result from the use of a more thermally resistive material, titanium, for the chamber forward end. The compatibility of titanium and combustion products was proved in this program. The durability and life cycle capability of titanium to columbium metallurgical joints remains to be demonstrated.

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