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Space Shuttle Bipropellant RCS Engine

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The Space Shuttle RCS Engine requirements include high performance/reliability and minimum weight with 100-mission life with emphasis on reusability and minimum maintenance/servicing. Engine component selection is based on analyses and test data. The Bell columbium, fuel-vortex-cooled, flight-type engine has demonstrated, at 600 lbf thrust and at a chamber pressure of 200 psia, the performance goal of 295 sec vacuum specific impulse with maximum insulated wall temperatures below 2100°F. Pulse performance, duty cycles and endurance test data are presented with long engine life demonstrated. The Shuttle RCS Engine has successfully demonstrated 10,411 sec operation including 6377 firing cycles with no engine maintenance. Based on the engine performance and multimission environmental (salt water spray, sand and dust, vibration, and humidity) and hot fire cycles demonstrated, this candidate engine will meet all the Space Shuttle requirements and goals including reusability with no maintenance.

Introduction

DIFFERENT types of "Reaction Control System" rocket engines have been developed to provide space vehicle attitude control and orbit adjustment and docking maneuvers. Bipropellant and monopropellant designs have met the requirements for single mission manned and unmanned spacecraft. The Space Shuttle requires RCS engines with similar capabilities but adds the requirements for reusability and high reliability for 100 missions over a 10-yr service life. The Shuttle requirements have prompted NASA to award technology contracts to define the current state of the art of RCS engines to meet those requirements. The technology questions include the ability to translate current design experience to larger thrust levels and the ability of the engine to survive multiple mission cycles with minimum servicing.

The technology contract effort for bipropellant RCS engines using nitrogen tetroxide/monomethylhydrazine is summarized in this paper. The work is being conducted by Bell Aerospace Co. under NASA Contract NAS 9-12996 with N. Chaffee as NASA-JSC Technical Monitor.

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Requirements and Goals

Table 1 presents the RCS engine technology program requirements and Table 2 presents the goals. However, several additional characteristics must also be met. The engine technology must be scalable over a thrust range of 400–1100 lbf. The pulse mode operation is further constrained by the need to minimize chamber pressure overshoot or spiking during the start transient and to

Table 1 Design requirements

Vacuum steady-state thrust (F_∞)	600 lb
Exhaust nozzle area ratio (ε)	40
Steady-state chamber pressure (P_c)	200 psia
Engine steady-state mixture ratio (O/F)	1.6
Propellants	
Oxidizer	N_2O_4 (MIL-P-26539C MON-1)
Fuel	MMH (MIL-P-27404A)
Pressurant	Helium
Propellant feed conditions	
Static pressure	300 ± 6 psia
Dynamic pressure	290 ± 10 psia
Temperature	75 ± 35°F
Valve voltage	28 ± 4 vdc
Minimum impulse bit (nominal conditions)	30 ± 10 lb-sec
Maximum pulse frequency	5 cps
Maximum single firing	600 sec
Maximum firing time per mission	1000 sec
Maximum number pulses per mission	2000
Engine life	100 missions 10 yr 100,000 sec 200,000 pulses
Stability	
High frequency	Dynamically stable recovery in 20 msec
Low frequency	Oscillations-± 5%
Thrust vector alignment	0.5°
Maximum outer wall temperature	800°F
Engine environment	
Temperature	-20°F to $+300$°F
Launch/re-entry temperature (5 min)	2000°F at nozzle exit 1500°F at throat
Pressure	S/L to 10^{-13} torr
Acceleration (twice/mission)	3.5 g for one min
Vibration	*
Shock	TBD
Rain (per mission)	0.5 in./hr for 0.5 hr
Sand (per mission)	140 mesh-500 FPM (4 hr)
Salt atmosphere (per mission)	Coastal areas at 75 ± 20°F for 30 da
Humidity (per mission)	0–100% RH for 30 da
Leakage of propellants or combustion gases	None

* Contract definition
Random vib (one min per axis/Mission)
20–90 Hz 0.1 g^2/Hz
90–180 Hz + 12 db/octave
180–350 Hz 1.6 g^2/Hz
350–2000 Hz − 6 db/octave

Sinusoidal vib (6 min/axis/mission)
5–23 Hz 1 g
23–40 Hz 0.036 in. DA

minimize plume contamination. Steady-state and pulse-mode operation at wide variation of feed pressures, blockage of injector orifices, and pressurizing gas entrainment or bubbles in the propellants will not cause damage to the engine. Specific impulse and mixture ratio shift should be minimized for off-nominal propellant feed pressure and temperatures. The engine design must lend itself to postflight inspection and maintenance. Postflight servicing and maintenance will be minimized. All of the requirements are to be met with an engine design which has minimum weight and envelope.

Table 2 Performance goals

Steady-state vacuum specific impulse (I_{sp_∞})	295 sec
Pulse vacuum specific impulse (I_{sp_∞})	220 sec ($I_T = 30$ lb-sec)
Engine start time (0.90% P_c)	50 msec
Engine decay time (100–10% P_c)	50 msec
Specific impulse shift	Minimum
Mixture ratio shift	Minimum
Off limits operation	Capability to withstand

The total requirements present an exacting task of engine configuration selection which provides the maximum potential of compliance with all requirements and goals over the engine life. That selection process includes analyses and tests.

The technology program has evolved with a comprehensive program of design, analyses and tests. The initial engine design phase with supporting tests was completed in late 1972 demonstrating four missions on a single injector and five missions on a single thrust chamber. Two 600-lbf thrust engines have recently completed the flight-type demonstration program. One engine accumulated ten missions‡ of hot fire testing while the other engine demonstrated ten missions cycles of environmental exposure (salt water spray, sand and dust, vibration, and humidity) and subsequent hot fire after each cycle to assess the effects of the exposure.

Engine Design Approach

General

The Bell flight type engine is a film-cooled coated columbium alloy integral thrust chamber and nozzle extension externally insulated with a direct acting torque motor bipropellant valve. A columbium injector is welded to the thrust chamber. The injector is a doublet configuration providing fuel vortex film coolant to the thrust chamber. A bipropellant torque motor valve is bolted to the injector with a titanium standoff for valve thermal protection. The titanium standoff is welded to the columbium injector. The external insulation is a low-density blanket encased in foil and mechanically attached to the engine. The selection rationale for the engine configuration follows.

Fuel Vortex Cooling

Several years of effort have been devoted to methods of rocket engine gas side film cooling with N_2O_4 and amine fuels. The objective of the investigations was to establish design techniques which would provide a uniform wall thermal environment at predictable values with minimum engine performance loss. Various types of fuel and oxidizer film cooling were evaluated primarily by test firings, and fuel vortex film cooling was determined to provide the desired characteristics.[1,2] Fuel vortex film cooling is provided by tangential injection of fuel near the chamber wall and the development of a fuel vortex film at the head end of the chamber.

Most of the testing was conducted with 3.4-in.-diam hardware at an L^* of 14 in. at a chamber pressure of 175 psia yielding a thrust of 640 lb ($\varepsilon = 40$). Radiation-cooled, coated columbium thrust chambers were used with pyroscanners to obtain continuous thermal profiles of the engine during test. This allowed a direct measurement of temperature uniformity and maximum wall temperature for a completely insulated thrust chamber. The engine was also operated with external insulation to confirm the computed values of gas driving temperatures. Wall temperatures were measured by embedded thermocouples in these tests. Close agreement was obtained between the derived driving gas temperatures and measured wall temperatures.

The maximum wall temperature which occurs near the throat is a function of the ratio of barrier flow to total propellant flow (ρ) for a given configuration. A statistical multiregression correlation analysis for the 640-lb engine thermal data was employed to assess the significance of the various operating parameters on the maximum steady-state wall temperature.

$$T_{\max} = A\rho + B(r_o) + C(P_c) + N$$

where $T_{\max}$ is maximum steady-state wall temperature, r_o is overall mixture ratio, ρ is the ratio of barrier flow to total

‡ The 10,411 sec represents 52 missions based on RI/SD definition of 200 sec/mission.

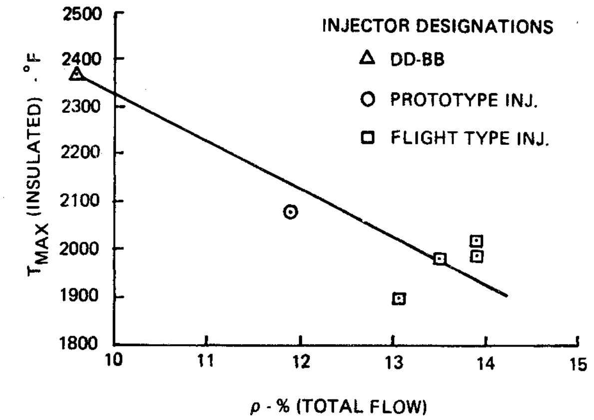

Fig. 1 Temperature—barrier flow correlation.

propellant flow, and A, B, C, N are constants. It was found that over the range of parameters investigated (that is, $0.08 < \rho < 0.14$, $125 < P_c < 250$ psia, $1.4 < r_o < 1.8$), ρ is the only significant parameter and the data for a given geometric configuration are correlated well by the relationship

$$\rho = (N^1 - T_{max})/A$$

where N^1 is an averaged constant. Figure 1 is a plot of T_{max} vs ρ for tests with ρ above 9%. Good agreement is indicated.

An analytical model was developed to incorporate the effects of combustor geometry and thrust on the relationship between T_{max} and ρ. The basic hypothesis underlying the analytical model is that the ratio of film coolant flow rate and the heat rejected to the barrier film, as it moves from the point of injection to the throat, is a constant. A fuel vortex cooling parameter ρ^* has been developed from that hypothesis. The comparison of test results is shown in Fig. 2, with the engine configurations in Table 3. These results include data from flight-type and nonflight type engines. At barrier flows below 13%, the flight-type engine data indicates somewhat higher temperatures than nonflight type engines. The higher thrust data is associated with the work at Bell for the Space Shuttle Orbit Maneuvering Engine (OME) under Contract NAS 9-12803. The data show that the requirement of cooling concept scalability has been demonstrated.

Table 3 Engine configuration

Thrust	L-in.	D_c-in.	L^*-in.	D_t-in.	P_c-psia
600	3.9	3.4	14	1.45	200
5500	12.8	8.7	26	6.10	100
6000	10.8	10.0	26	5.80	125

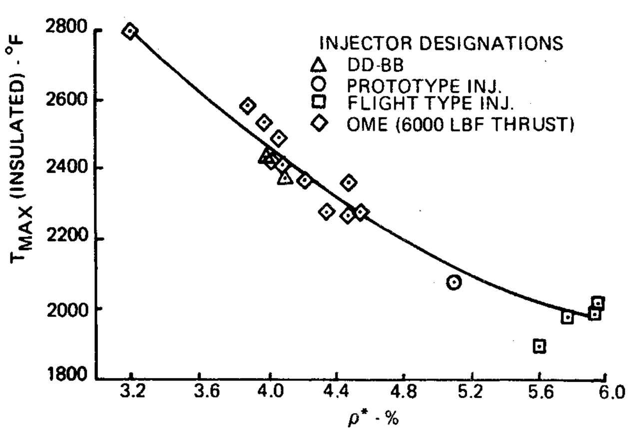

Fig. 2 Vortex cooling parameter correlation.

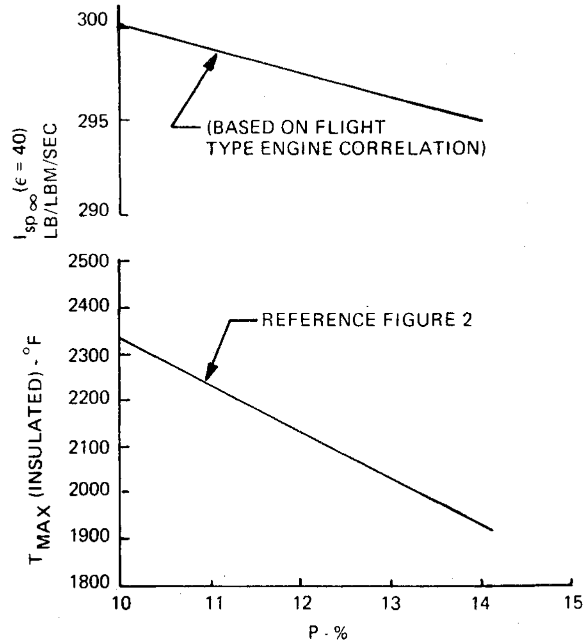

Fig. 3 Effect of barrier flow on performance and wall temperature.

Injector

The specific impulse goals of the Space Shuttle RCS engines require an injector providing high steady-state and transient combustion efficiency in conjunction with the type of chamber cooling employed. Various injector configurations were tested as part of the film cooling tests described in the preceding section, and the steady-state performance goal has been demonstrated on the established injector configuration. Injector design characteristics for transient operation include minimization of propellant manifold volumes and provision for the desired propellant sequencing. The injector must damp high-frequency combustion disturbances and operate without excessive low-frequency pressure oscillations. The injector material must provide long life in its operating and nonoperating environments. The injector design for the preceding requirements will be reviewed in this section with supporting test data. The additional engine transient operation will be discussed in a later section.

The injector configuration incorporates two concentric rows of unlike doublets with the oxidizer orifices on the outside of each row. Six cross-feed holes provide fuel vortex film cooling from the fuel manifold. The injector has acoustic slots to dampen high-frequency combustion disturbances. The injector is fabricated from a columbium (1% zirconium) alloy.

The selection of the design point of the injector requires a tradeoff between the amount of fuel vortex film cooling and the performance achieved within the constraints of maximum wall temperature. Figures 3a and 3b relate the operation to wall temperature and performance for operation to the requirements of Table 1. The plots based on test data indicate that the 2200°F maximum insulated wall temperature can be achieved with $\rho = 13.5\%$ generating a performance of 295 sec vacuum specific impulse. The performance goal has been demonstrated with two flight-type engines (SN RDV-2B and SN FT-2A) which are shown in Fig. 4.

High-frequency damping is provided by acoustic slots designed to dampen the first tangential mode of high-frequency combustion instability. Bomb tests have demonstrated dynamic stability on three injectors with pressure recovery within approximately 1 msec.

In addition, high altitude (250,000 ft) start tests have indicated a maximum spike of 740 psia based on 110 tests with three flight-type injectors. This occurred with 50 msec tests with cold (40°F) hardware and propellants with or without helium saturation.

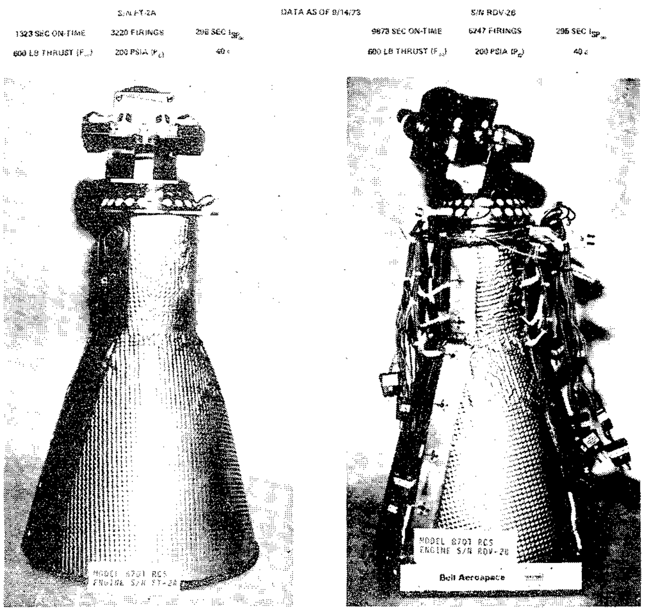

Fig. 4 Bell space shuttle flight type engines.

The selection of columbium (1% zirconium) alloy for the injector is based on material tests which show the alloy to be unaffected by exposure to propellants and the nitric acid that can be formed by N_2O_4 and moisture as well as its excellent resistance to marine environment. The fabricability of the alloy is excellent; over 6000 RCS engine columbium injectors have been manufactured at Bell including over 5500 for the Minuteman III PSRE. Tests to verify negligible injector gas embrittlement for the operating life of the Space Shuttle have been reported.[3]

Thrust Chamber Assembly

Evaluation of the RCS engine requirements indicates that regenerative cooling is not a viable concept for the application; similarly the requirements of life, reusability, and dimensional (thrust) vector control eliminates ablative chambers from consideration. Therefore, the uncooled insulated or an inter-regeneratively cooled chamber must be used. The preceding section defined the tradeoff between maximum wall temperature and performance which indicated that the vacuum specific impulse goal can be met with a maximum wall temperature of 2200°F. The thrust chamber material selected must operate at the maximum temperatures with the capability of surviving the firing environment as well as operational requirements and exposures. Included in the selection criteria is propellant compatibility, resistance to formation of propellant residues, compatibility with coolant barrier gas species (that is, N_2, H_2, NH_3), thermal cycle stability, creep resistance and fatigue life at temperature, resistance to salt spray, sand and dust and humidity, fabricability, and weight.

The spectrum of potential materials was re-evaluated. Molybdenum, copper, TD Ni, TDNiCr, and Beryllium are attacked by propellants or nitric acid formed by moisture and N_2O_4 or salt spray. The maximum temperature of 2200°F near the throat does not allow the use of superalloys (L605, etc.) because of insufficient thermal margin. Tantalum is a candidate but it is heavy. The temperature limits of columbium alloys are consistent with the requirements and the material is compatible with all the environments (that is, propellants, nitric acid, salt spray, sand and dust, humidity, etc.). Resistance to combustion gas embrittlement and re-entry oxidation has been demonstrated with silicide-coated columbium alloys. Silicide-coated columbium has demonstrated steady-state oxidation-erosion life in excess of 212 hr at 2200°F and greater than 10,000 thermal fatigue cycles at 2200°F.[4] Tests at Bell have shown that coated columbium is unaffected by propellants or nitric acid exposure. A coated columbium alloy thrust chamber can be joined by welding which eliminates hot gas mechanical joints and so offers high reliability and reusability. The columbium thrust chamber assembly (including injector) is backed by over 6000 engines of this type produced by Bell at thrust ranges from 1 lbf to 6000 lbf, including over 5500 engines produced for Minuteman III PSRE.

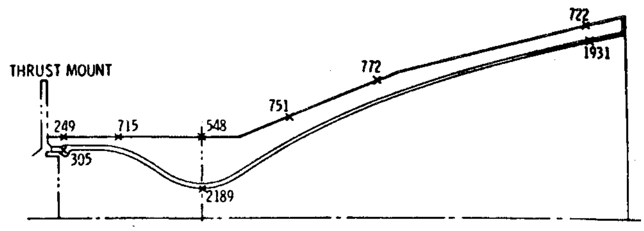

Fig. 5 Steady-state temperature distribution—flight type engine (Phase III).

The selection of the specific thrust chamber alloy and coating includes past experience and projection of current life data to the 100,000 sec (27.8 hr) life. The principal candidates are SCb-291, FS-85, C129Y, and C103 columbium with R512A or R512E silicide coatings. Tests by various agencies indicate that the R512E coating is superior to R512A for the total RCS engine environmental exposure.[5,6] SCb-291 was selected for use on the NAS 9-12996 program because of maximum creep life. Creep property data for the abovementioned four alloys are being investigated.

The selection of 12 lb/ft^3 Dynaflex insulation with a foil wrap has been made on the basis of 100 missions.[5] Insulated engine testing at Bell has been carried out with that insulation without detectable change of insulation or coated external surface of the thrust chamber.

Valve Assembly and Mount

The rapid operation, cycle life, and leakage requirements make a poppet valve the most attractive type for RCS engine application. Various kinds of poppet valves were studied and compared by weight, cost, envelope. power consumption, dribble volume, decontamination characteristics, reliability, scalability, and design maturity. The valves compared had nonmetallic seats, response times consistent with engine operation (pulse mode) over a thrust range of 400–1100 lbf. The following types of valves were considered: direct acting torque motor valves, two-stage bipropellant-actuated bipropellant valve, two-stage mechanically linked pilot-operated valve, pressure-balanced valves, and two-stage propellant-actuated individual valves.

Valves were evaluated for potential problems. The following items were considered: use of sliding fits/seals; 2) sensitivity to fatigue; 3) repeatability of flow transients; 4) leak paths (internal); 5) cleaning and flushing capability; 6) materials exposed to propellants; 7) overheating susceptibility (self-induced); 8) seat design; 9) contamination susceptibility; and 10) weights.

The direct-acting torque motor valve was selected on the basis of the study to meet the Space Shuttle RCS engine requirements.

The valve and engine mount subassembly must provide structural support of the engine and valve for all operating and nonoperating loads (vibration, etc.) and maintain thermal protection from the engine. The thermal protection role is aided by a low head end chamber wall temperature (<400°F) provided by vortex film cooling. Titanium (6A1-4V) alloy feed tubes and support are selected for low conductivity and minimum weight. The assembly technique of columbium-to-titanium welds has been defined by the Minuteman III PSRE engines.

Engine Characteristics

Three major engine characteristics will be discussed in this section—thermal analyses, pulse performance, and engine

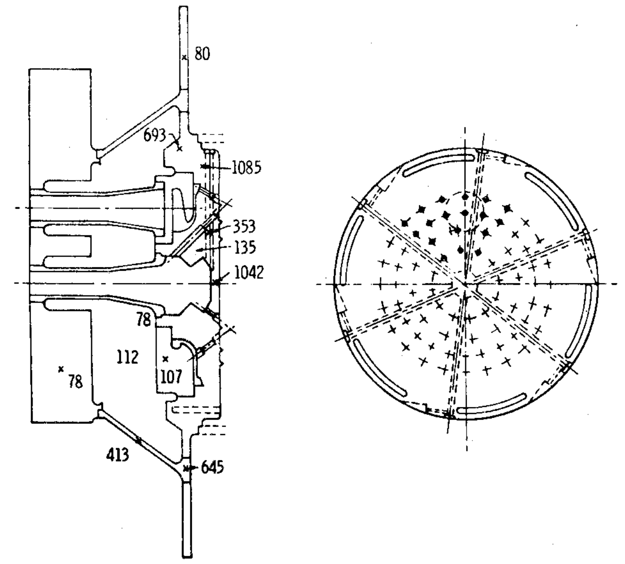

Fig. 6 Steady-state temperature flight type injector—phase III all welded engine.

durability. The weight of the flight-type 600 lbf thrust engine is 12.25 lb (measured) including 4.5 lb of valve weight.

Thermal Analyses

An 82-node thermal model of the flight-type engine was constructed to define the engine thermal characteristics. The coated columbium emissivity was set at 0.8 which is based on the R512E coating. The outer surface of the engine was constrained by the required 800°F temperature. The throat temperature was established as 2200°F. The thermal input of valve electrical power was considered in the analysis.

Figures 5 and 6 show the calculated steady-state temperature distribution for the flight-type engine. Figures 7 and 8 present the calculated effects of temperature on various parts of the engine for worst-case duty cycles. Test data indicates the chamber section is operating at 350°F, the throat at 2000°F, and the nozzle at 1600°F. The injector valve mount is operating at 114°F and increases to 140°F (82°F initially) during the worst-case heat soak. The test data shows the conservatism of the analytical model.

Durability

The flight-type engine was subjected to worst-case mission duty cycles and maximum endurance testing to assess the engine capabilities. The engine successfully completed the following: 1) worst-case mission duty cycle (965 sec firing time and 120 firings including 9–90 sec firings and pulses); 2) worst-case mission duty cycle with helium-saturated propellant; 3) worst-case mission duty cycle at maximum conditions (O/F = 1.7, $P_c = 220$ psia, $T = 110$°F, helium-saturated propellants); 4) maximum endurance test—600 sec continuous; 5) thermal duty cycle (steady-state firings and pulses); 6) thermal duty cycle with helium-saturated propellants; and 7) thermal missions (steady-state and pulses).

The performance variation range on a worst-case mission (965 sec firing time and 120 firings) or maximum endurance test is less than 0.7%. The worst-case mission duty cycle at maximum operating conditions reached a maximum throat temperature of 2320°F compared to the normal maximum of 2000–2100°F. Based on these results the engine has a large thermal margin since the R512E coating capability is 2900°F.

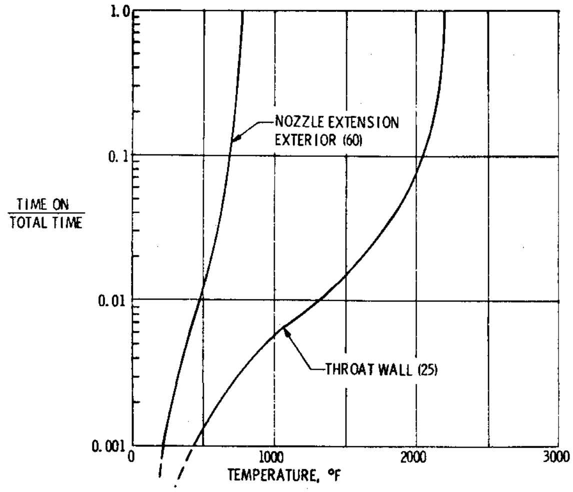

Fig. 8 Engine wall temperature worst duty cycle, phase III flight engine.

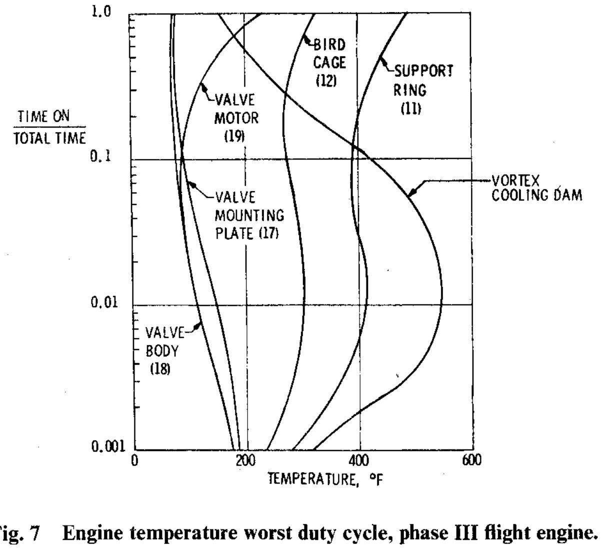

Fig. 7 Engine temperature worst duty cycle, phase III flight engine.

Pulse Performance

Characterization of pulse performance of the flight-type engine was conducted indicating a higher pulse specific impulse with helium-saturated propellants than without saturated propellants. Figure 9 shows the results. Tests were conducted to a minimum electrical pulse width of 28 msec.

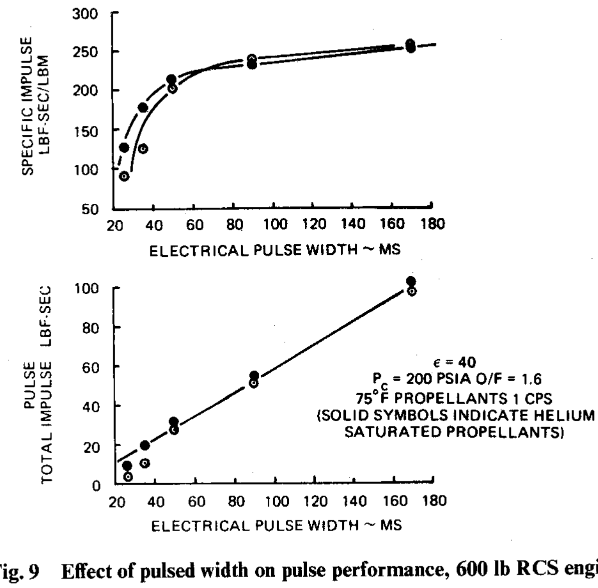

Fig. 9 Effect of pulsed width on pulse performance, 600 lb RCS engine.

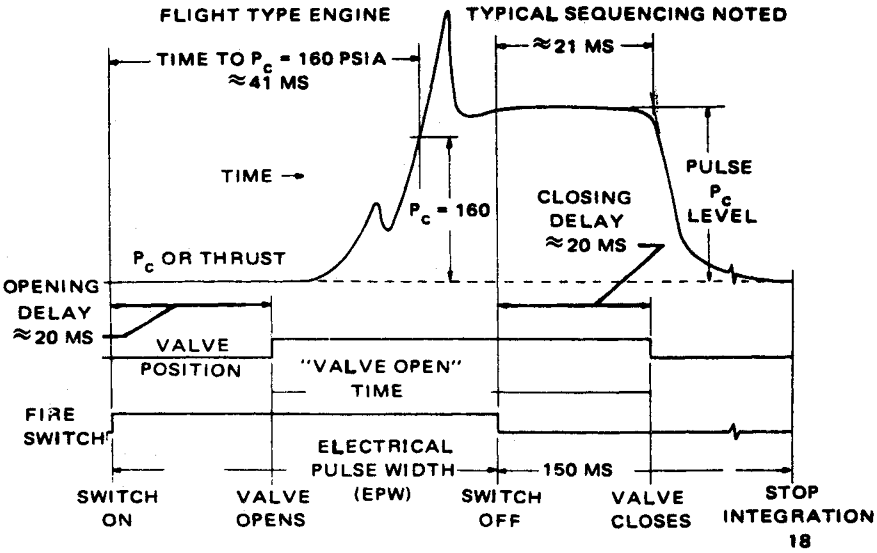

Fig. 10 Anatomy of a pulse.

All of the pulses fired during the tests were conducted in trains of 100 pulses maximum at altitude to obtain pulse specific impulse and tested at a constant electrical pulse width and constant frequency. Thrust was used for obtaining impulse bit, and a Propellant Metering System was used to measure total propellant during the pulse train. Figure 10 shows the anatomy of a pulse.

Maintenance and Servicing

The component design and material selections which have already been described have included factors associated with ensuring long service life, minimum maintenance and servicing, and ease of decontamination when desired. These factors include an injector designed for forgiveness to contamination and an engine assembly which has only one mechanical joint at the valve-to-injector valve mount interface.

The goal of minimum maintenance was explored by environmental test series. Ten environmental/hot fire cycles were demonstrated successfully with no maintenance with each cycle consisting of: 1) salt water spray per MIL-STD-810B for 30 min with the engine nozzle up at 45°; 2) sand and dust for four hours with engine nozzle up; 3) sinusoidal vibration for 6-min sweep plus dwell time per axis; 4) temperature and humidity for over 7 hr over a range of 40–160°F and 95% relative humidity; and 5) hot fire test—pulsing and steady state. The fire test data after each of the 10 environmental cycles indicated no change in performance (data within instrumentation accuracy), and no engine maintenance was conducted.

Early in the program, one engine demonstrated reusability with no maintenance after 3384 sec firing time and 7385 firings. One of the flight-type engines had accumulated 10,411 sec firing time and 6377 firings with no maintenance. So, the design has demonstrated excellent reusability with no maintenance.

Summary

Firing data shows that the selected method of engine cooling, "vortex" film cooling with a low-density external insulation, has exceeded the engine-cooling concept scalability goal. The injector and chamber material selections provide a thrust chamber assembly without mechanical joints and of minimum weight. Twenty-nine bomb tests over a range of operating conditions are reported with measured damp times of 1.2 msec or less. The bomb tests indicate that the dynamic stability requirements can be met. The injector/chamber-cooling concept selected exceeds the steady-state specific impulse goal (295 sec) with chamber temperatures (less than 2100°F) consistent with the long operating life requirement.

The valve and valve mounting selected also ensured high engine cycle life, minimum engine weight and an engine assembly capable of meeting the pulse mode I_{sp} requirement. The control of propellant injection provided by the valve/injector combination limits chamber pressure overshoot on start. Multiple worst-case mission duty cycles have been conducted, and a maximum throat temperature of 2320°F at maximum operating conditions (high P_c, high O/F, high T_{props}) was observed which suggests a large operating margin for the silicide-coated columbium chamber.

One flight-type 600-lb thrust engine accumulated 10,411 sec of firing time and 6377 firings with no maintenance or purging between runs. Ten environmental/hot firing cycles were also demonstrated without engine maintenance. The environmental cycles were conducted without engine nozzle closure and included salt water spray, sand and dust, and vibration.

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