Ion Propulsion Flight Experience, Life Tests, and Reliability Estimates

J. H. MOLITOR*
Hughes Research Laboratories, Malibu, Calif.

The application of low-thrust ion propulsion systems to space missions requires long duration (~5,000–20,000 hr) component operation. Thus, components must be developed with wearout mean times to failure in excess of these required mission times. Furthermore, chance failures which occur during the useful life of components must be minimized. This paper makes an assessment of both early and wearout failure modes of ion propulsion systems by examining the results of existing developmental and long duration testing. Estimates of chance failure rates of system components are also presented along with design concepts which maximize total propulsion system reliability.

I. Introduction

BECAUSE of the long time operation required for most space applications, ion propulsion system reliability is a major concern. For this reason, the long duration and space testing of critical components, subsystems, and systems associated with primary and auxiliary ion propulsion systems are of great importance. It should be recognized, however, that although a large number of long duration tests have been conducted to date on ion propulsion systems, subsystems, and components, these tests have been part of a technology development phase and are not to be considered statistical reliability testing of fully developed systems. Thus, even though space-qualified ion propulsion systems are becoming available, system reliability can only be inferred from the results of past and present developmental testing and estimated by analytical techniques.

II. Flight Experience

Since 1962 there have been eight space tests of ion engine systems culminating in the flight of the Space Electric Rocket Test (SERT) II launched on Feb. 3, 1970 into a near polar orbit of 1000 km. A chronological listing of these ion engine space tests is shown in Table 1. As noted, the two types of thrusters tested to date are the cesium-contact and the mercury-bombardment ion engines.

Ion engine flight testing started with a series of three ballistic flights launched with the Scout rocket. These three flights took place on Dec. 18, 1962, Oct. 29, 1964, and Dec. 21, 1964. The engines tested were Electro Optical Systems' 2 mlbf contact ion engines which were provided approximately 40 min for operation above 100 nm. The first failed through high voltage arcing because of venting of the battery compartment into the power conditioning unit. The second functioned according to plan as verified through electrical measurements. The third flight gave only 10 min of operation because of a launch failure, but all measurements indicated normal operation up to that time.

The SERT I was launched from Wallops Island on July 20, 1964 on a four-stage Scout rocket. It followed a ballistic trajectory for 47 min and had the distinction of being the first successful flight of an ion engine. The Hughes cesium-contact ion engine was unable to be operated because of a high-voltage short circuit. However, the NASA Lewis Research Center mercury electron bombardment thruster operated for 31 min at a thrust level of 4.5 mlbf providing data for ion beam neutralization, thrust level, radio communication interference, and differences in performance between ground and space testing. Thrust was measured by determining changes in spin rate. From these measurements, it was determined that complete beam neutralization was achieved in agreement with vacuum chamber tests. In addition, there was no interference with radio communication.

The next flight test was the SNAP-10A orbital flight test of a 2 mlbf contact ion engine system. Electrical incompatibility between the ion engine and spacecraft resulted in an interruption

Table 1 Ion propulsion space tests

Date	Flight
Dec. 18, 1962	Air Force ballistic flight to test an Electro-Optical Systems cesium-contact ion thruster system (No. 1)
July 20, 1964	NASA SERT-I ballistic flight to test a Hughes Research Labs. cesium-contact ion thruster system and NASA Lewis Research Center mercury-electron bombardment thruster system
Oct. 29, 1964	Air Force ballistic flights to test an Electro-Optical Systems cesium-contact ion thruster system (No. 2)
Dec. 21, 1964	Air Force ballistic flights to test an Electro-Optical Systems cesium-contact ion thruster system (No. 3)
April 3, 1965	Air Force SNAP-10A orbital flight with Electro-Optical Systems cesium-contact ion thruster as an auxiliary experiment
Aug. 10, 1968	NASA ATS-4 gravity gradient stabilized synchronous satellite containing two Electro-Optical Systems cesium-contact ion engines for station-keeping
Aug. 12, 1969	NASA ATS-5 gravity gradient stabilized synchronous satellite containing two Electro-Optical Systems cesium-contact ion engine for station-keeping
Feb. 3, 1970	NASA SERT-II orbital flight to test two NASA Lewis Research Center 15 cm mercury-electron bombardment ion engines during a six month polar orbit

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* Manager, Ion Physics Department. Member AIAA.

Table 2 SERT II 15-cm mercury bombardment thruster and component life tests (1967–1970)

Component	Hours	Reason for termination
Experimental thruster (No. 1)	5000[1]	End of experimental thruster program
Experimental thruster (No. 2)	5480	End of experimental thruster program
Prototype thruster (P-3)	1000	End of 1000-hr demonstration test
Prototype thruster (P-5)	208	Neutralizer vaporizer passing liquid Hg
Prototype thruster (P-5)	3200	Increase in main discharge current
Prototype thruster (P-5)	1240	Replace main cathode with final flight design
Prototype thruster (P-10)	5412[2]	End of program (no failures)
Neutralizer, grids, body	6331[2]	End of program (no failures)
Grids and body	7346[2]	End of program (no failures)
Prototype thruster (P-20)	779[2]	Internal arc damage to PPU; no T/S failure
Prototype thruster (P-20)	6787[2]	Main propellant tank emptied
Neutralizer, grids, body	7994[2]	Main propellant tank emptied
Various prototype thrusters (on prototype SERT II spacecraft)	2400 hr[3] in ~50 tests	One T/S failed due to facility failure
Neutralizer cathode in bell jar	12,979[4]	Facility failure exposed hot cathode to air, heater failed on restart attempts

of telemetry during its initial operation and for this reason, it was subsequently not activated.

The two 5–20 μlbf cesium-contact ion engines launched on ATS-4 in 1968 functioned normally in all respects for the two month period prior to its decay from orbit. This engine is identical to the one used in ATS-5. The engines on ATS-5 were never turned on because of a malfunction of the nutation damper which caused the spacecraft to go into a flat spin.

Since cesium-contact thrusters are no longer under development in this country, the results of the SERT II are the most significant of all ion engine flight tests to date. It is appropriate, therefore, to emphasize the accomplishments of SERT II and to discuss further the failure modes of the thrusters as well as the subsequent development efforts which have eliminated these failure modes from present-day mercury bombardment ion engines.

First, SERT II provided long-term operational experience with solar-powered, space-qualified ion thrusters. This experience has established the viability of long duration operation of ion thrusters in space. Second, both thrusters, in general, performed the same in space as in ground tests. This correspondence between laboratory and space performance provides confidence in the specification of thruster operating parameters prior to launch for future missions. The first thruster operated continuously for 3781 hr before an electrical short caused an early shutdown with less than a month remaining to complete the goal of six months' operation. The second engine also failed because of an electrical short after 2011 hr.

The failure mode (as established in ground tests) of the SERT II thrusters was a high voltage short between the screen and accelerator electrodes. This short was caused by the erosion of the accelerator grid (in a localized area under the neutralizer) resulting in the bridging of the electrode gap by a piece of the grid webbing. This particular failure mode has since been eliminated by the repositioning of the neutralizer (e.g., as proved by subsequent long duration testing of various sized mercury bombardment thrusters). With the elimination of this failure mode, projected SERT II thruster lifetime is 15,000–20,000 hr.

A final significant result of the SERT II flight was the fact that during the past $2\frac{1}{2}$ years, all cathodes and vaporizers have remained functional and have been periodically ignited and tested, giving over 180 restarts of the cathode subassemblies.

III. Life Tests

During the past eight years, a substantial number of long duration tests (i.e., >500 hr) have been carried out on mercury bombardment thrusters. The majority of these tests have been carried out as part of the development phase of ion thrusters and, therefore, should not be considered in the realm of statistical testing of fully developed, quality controlled, flight qualified devices. However, these tests did serve to uncover wearout failure modes which through subsequent development were eliminated, to add to the confidence in those component designs which repeatedly survived, and, finally to provide data on the fundamental and limiting failure modes of ion thrusters. Since in many cases the test periods approached the operating times required by typical mission applications, these latter data allow projections of ultimate thruster life to be made with reasonable accuracy.

During the course of the development of the 15-cm mercury bombardment thruster, which was ultimately to be flight tested on SERT II, numerous long duration tests of thrusters, subassemblies, and components were carried out. A summary of these tests is given in Table 2. The testing associated with the two Experimental Thrusters was performed during the early development phases of the thruster. Thus, although the thruster bodies were operated for an accumulated time of 5000 hr and 5480 hr, respectively, the cathodes and grids were replaced with advanced designs when and if failures occurred. Based on the results of these tests, a flight thruster design was generated. Evaluation of this design was carried out primarily during the "hands off" testing of Prototype Thrusters P-10 and P-20. These two tests, both of which were completed without failures, lasted for 5412 hr and 6787 hr, respectively. (Actually, most of the thruster subassemblies including the neutralizer, grids, and body

Table 3 Potential wearout mechanisms (based on SERT II thruster testing)

Component/wearout mechanism	Comments	Projected life
Hollow cathode		
Tip wear	Changes orifice geometry	20,000 hr
Insert material depletion	Loss prevents restart	15,000 hr
Heater corrosion	None seen	Indefinite
Electrode system		
Accelerator grid erosion		
1) Direct beam ions	None seen	Indefinite
2) Charge exchange ions	Uniform erosion	20,000 hr
3) Neutralizer ions	Localized erosion	2,000 hr[a]
Screen grid erosion from Discharge chamber ions	Minimal effects	50,000 hr
Metal flakes from Discharge Chamber shorting grids	None seen	Indefinite
Grid insulator shorts	None seen	Indefinite

[a] Could occur anytime after ~2,000 hr when and if an eroded web becomes trapped between electrodes.

Table 4 5-cm mercury bombardment thruster and component life tests (1972–1973)

Component	Hours	Mode	Reasons for termination
Thruster (SIT-5II No. S/N 101)	9715[5]	1 Shutdown	Electron vector grid element severed
Translating vector grid	5027[5,6]	Accumulated	Tests completed
Electrostatic vector grid	7688[5]	Continuous	Vector grid element severed
Propellant reservoir	13,500[7]	Accumulated	
Thruster (SIT-5II S/N 204)	927[8]	Continuous	End of test
Main cathode		594 cycles	Vaporizer heater failure
Neutralizer		1009 cycles	
Cathode isolator-vaporizer assembly			
S/N 109	13,700[5]	1 exposure many restarts	Removed from SIT-5 life test
Neutralizer-vaporizer assembly			
S/N 111	9715[5]	1 exposure many restarts	Removed from SIT-5 life test
Misc. components			
First gen. E.V. grid	1000[9]	Continuous	End of test—inspection
1-axis E.V. grid	1367[10]	Accumulated	End of test—inspection
LeRC cathode	10,000[11]	184 cycles	
LeRC cathode	2150[11]	943 cycles	
Pulse ignition cathode		10,000 cycles[13]	End of test—inspection
LeRC cathode	2650[11]	59 cycles	More study required
HRL cathode	1104[12]	Continuous	End of test

were pretested giving them, in one case, an accumulated test time of approximately 8000 hr.)

Several observations, based on the SERT II thruster development and life testing, can be made concerning potential mercury bombardment ion thruster wearout phenomena. Initially, the most critical components in terms of inherent wearout mechanisms were expected to be the hollow cathodes and the screen and accelerator electrodes. The life tests of the SERT II thrusters confirmed these expectations. A list of the potential wearout failure mechanisms associated with these components is given in Table 3 along with an evaluation of their significance as basic life-limiting phenomena. As presented in Table 3, the life-limiting wearout mechanism of the SERT II thrusters was shown to be the erosion of the accelerator grid in a localized area beneath the neutralizer. (Actual failure caused by this erosion did not occur in ground tests because gravity prevented eroded accelerator grid webs from shorting out the electrode system.) Investigation of other potential wearout phenomena indicated mercury-bombardment thruster lifetimes of 15,000–20,000 hr.

In addition to the 15-cm SERT II thruster development and life tests, a large number of test hours have been accumulated on mercury-bombardment ion engines, for example, the 5-cm, 15-cm, 20-cm, and 30-cm devices under development for auxiliary and primary propulsion applications. Tables 4 and 5 list the long duration testing performed on these thruster sizes and their critical components. Most notable (from a thruster life standpoint) of the tests listed in these two Tables are the 9715 hr test of a 5-cm thruster and the 1100 hr test of a 30-cm thruster. The significance of these two tests is that they established that the enhanced localized erosion of the accelerator grid beneath the neutralizer has

Table 5 15-cm, 20-cm, and 30-cm mercury bombardment thruster and component life tests

Component	Hours	Mode	Reasons for termination
15-cm thruster, reservoir, and PPU (1966)	500[13]	Continuous	End of planned test
15-cm thruster (1967)	1000[14]	Continuous	End of planned test
20-cm thruster (1964)	4000[15]	Continuous	End of planned test
20-cm thruster system (1971) (including 3 thrusters, 2 transistorized PPU's, 1 reservoir, translator, and gimbals)	1500[16]	Continuous	End of planned test
30-cm thruster (1969) glass grid	500[17]	3 restarts	Glass coated grid failures
30-cm thruster (1970) dual grid (with center support)	450[18]	Continuous	End of planned test
30-cm thruster (1973)			
a) Dual grid (with insulated center support)	500[19]	Continuous	End of planned test
b) Dished grids (no center support); sheathed cathode heater	500[19]	Continuous	End of planned test
c) Dished grids; plasma sprayed cathode heater	1100[19]	Continuous	Vacuum facility failure
30-cm thruster (1973)	800	Continuous	End of planned test
30-cm thruster (1974)	2000	Continuous	Continuing
30-cm thruster components			
Cathode A	1600[20]	Continuous 5 restarts	Test completed
Cathode B	1980[20]	Continuous 6 restarts	Test completed
Cathode C	3880[20]	Continuous 19 restarts	Test completed
Cathode D	3950[20]	Continuous	Continuing
Neutralizer	1400[20]	Continuous	Test completed
Isolator	1800[21]	Continuous	Continuing
Dished grid A	1552[22]	Accumulated	Sporadic tests
Dished grid B	1580[22]	Accumulated	Sporadic tests

been completely eliminated and that they confirmed the projected lifetime (based on wearout mechanisms) of 20,000 hr for mercury bombardment thrusters was attainable. This conclusion has been further supported, of course, by the numerous additional long duration tests of critical components and subassemblies such as mercury propellant reservoir, electrode systems, translating vector grids, cathode-isolator-vaporizer assemblies, neutralizers, and cathodes. In many cases, these tests have exceeded 10,000 hr, thereby, giving additional credibility to the estimates of ultimate wearout lifetimes of greater than 20,000 hr for mercury bombardment thrusters.

Two life tests of 30-cm thrusters are planned during the 1974–1975 time period. The first test is a long duration run of a development model thruster. The second (and most significant) is a 10,000 hr test of a fully developed Engineering Test Model (ETM). Successful completion of this latter test will establish the availability of long life, flight qualified mercury-bombardment thrusters.

One final test to be noted is the modularized thrust subsystem listed in Table 5, which employed 20-cm mercury-bombardment thrusters. This system consisted of a three thruster array (two operating and one standby) including array translator and thruster gimbals, two flight-type transistorized power processor units, and a liquid mercury reservoir. The system test also included simulation of spacecraft command and control system functions and of spacecraft dynamics with control loops closed to the thrust vector control system (i.e., translator and gimbals). The test was run for 1500 hr with two thrusters (two of which accumulated 1200 hr and one 600 hr), as well as the two PPU's and single reservoir operating simultaneously. During the 1500 hr period two failures occurred; one was the loss of a thruster cathode, the other was the loss of a discharge supply. The latter was a system (i.e., not a PPU) failure mode caused by a transient interaction between the computer and PPU and was subsequently remedied by a software modification.

IV. Subsystem Failure Rates (Reliabilities)

Since it is unlikely that the amount of testing required to generate the classical failure rate curve (such as shown in Fig. 1) for ion propulsion systems will take place in the near future, a different approach must be taken to satisfy potential users that these systems will survive to perform their required functions. For example, if it is assumed that the early and wearout failures are eliminated and that the chance failure rate λ_c in the interval (T_E, T_W) is constant, the reliability of a device is given by the well known exponential law

$$r(t) = e^{-\lambda_c t} \tag{1}$$

This exponential law is derived on the assumption that the failures are random (i.e., chance failures) and that their statistics are given by the Poisson process. The "elimination" of early and wearout failures can be accomplished by proper preflight checkout and by limiting required operating times to an interval

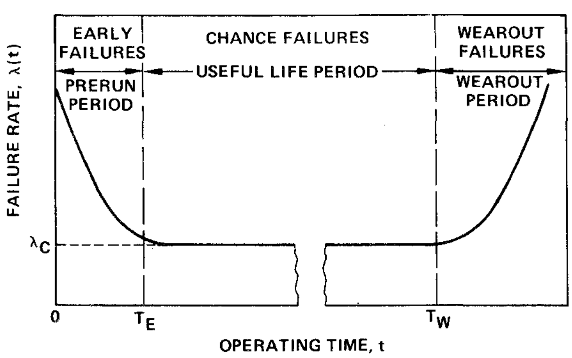

Fig. 1 Typical component failure rate vs operating time curve.

less than the established wearout life of the major subsystems (i.e., by lifetest). The estimating of failure rate values can then be limited to random or chance failures that might occur during the useful life of the system. Thus, the approach to be taken here is the following.

1) *Early Failures:* Indicate potential early failure modes, establish proper checkout procedures, and specify required preflight test periods (i.e., determine time T_E).

2) *Wearout Failures:* Indicate life-limiting wearout failure modes and, based on existing life-test data, project minimum wearout life expectancy (i.e., determine time T_W).

3) *Chance Failures:* By analysis of the various component parts that make up an ion thruster, a propellant reservoir, and a power processing unit, estimate the chance failure rates of these three major ion propulsion subsystems (i.e., estimate subsystem failure rates λ_c).

Early Failures

Since the potential early failures associated with the power processing unit are typical of space-qualified electronic systems, only the thruster and reservoir subsystems will be analyzed here. First, any ion thruster (or reservoir) considered for a flight application will have been based on designs that have passed normal flight qualification tests. Furthermore, quality control procedures governing fabrication and assembly of thruster systems will have been established during development programs to minimize the probability of both early and random failure modes. However, since complete elimination of these failure modes would require an excessive expenditure of time and money on quality control testing for each unit produced, some preflight testing is required.

Most early failure modes have been experienced in developmental testing and a substantial background of data is available for detection of potential early failure by proper prelaunch testing of the thruster system of subsystems. The details of this testing procedure should be tailored to fit the requirements of the mission application. However, a format similar to that used in the SERT II program is recommended. The schedule of prelaunch testing under that program proceeded as follows.

1) Preliminary operation of assembled thruster system for calibration of operating parameters over full operating range with laboratory power processor.

2) Preliminary operation of assembled thruster with flight power processor to calibrate thruster and telemetry system.

3) Vibrational and thermal-vacuum test of thruster system mounted on spacecraft.

4) Operation of all thruster and system components while installed on spacecraft to accumulate about 50 hr operating on each subsystem.

The amount of operating time logged on each thruster or thruster system under the preliminary testing phase will depend on the test facility available. Operation should be kept to a minimum if testing must be done in a small vacuum facility where the ion beam impacts and sputters a metallic collector during test. It may not always be practical to install the assembled spacecraft in a vacuum facility for operation of all thruster systems in situ following the vibrational and thermal-vacuum tests. If the final test with the entire thruster system in situ is not possible, then in order to minimize early failures, the thruster system must be demountable in such a manner that minimal disconnection of wiring and propellant lines is necessary. Using these guidelines and a catalog of "normal" thruster parameters, it should be possible to virtually eliminate early failures.

Wearout Failures

Based on the thousands of hours of accumulated developmental and life testing, it has been concluded that at this time the fundamental or limiting wearout mechanisms of an ion propulsion system (i.e., thruster, reservoir, and power processor unit) are associated with three critical components of the thrust device. These components and their wearout failure modes are

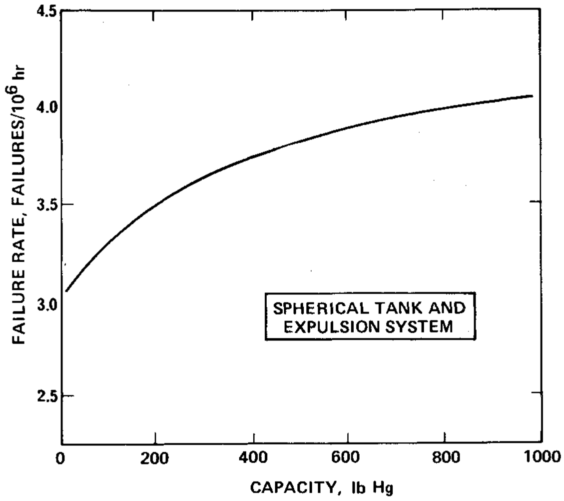

Fig. 2 Mercury propellant reservoir failure rates.

orifice erosion and barium depletion of the discharge chamber and neutralizer cathodes, charge exchange erosion of the accelerator grid, and screen/accelerator grid system short due to metal flakes from discharge chamber erosion. Estimates of the wearout life (i.e., the value of T_W) of present-day thrusters range from 15,000 to 20,000 hr. These estimates are based on straightforward physical extrapolation of the erosion and depletion phenomena indicated above.

It should also be noted that research and development efforts devoted to increasing thruster life beyond 20,000 hr are being pursued vigorously. It is expected that by increasing propellant mass utilization efficiency, reducing cathode starting and operating temperatures, reducing discharge voltages, and using low sputtering yield materials, thruster life can be extended to values much greater than 20,000 hr in the near future. It is, of course, important that life tests of fully developed thrusters be conducted for periods of 20,000 hr or more so that thruster life can be established without reliance on extrapolation. Furthermore, because of the potential deleterious effects of life testing in ground-based test facilities (e.g., backsputtering vacuum tank material), laboratory tests should continue to be augmented by space testing.

Chance Failures

In order to obtain an estimate of the chance (or random) failure rates for ion thrusters, propellant reservoirs, and power processor units without the benefit of statistical testing, an analysis of each of these subsystems was made on a component level. In the case of the thruster and reservoirs the resulting total failure rates were essentially the sum of the failure rates of the components and their assemblies. The values obtained in this manner for the thrusters should be considered a probable minimum since the thermal and electrical stresses associated with operation were not considered. The reservoir failures, however, can be considered representative. Furthermore, in the case of the power processor units, where individual component failure rates are directly available and where circuit designs including redundancy are known and amenable to reliability analysis, the failure rates (and reliabilities) should be reasonably accurate.

Thruster failure rates

Two ion engines, both of which have been developed to a flight qualification state, were analyzed. These are the 5-cm and 30-cm mercury-bombardment thrusters. The failure rates of these thrusters were calculated by determining the failure rates of their component parts and certain construction features (such as welds).

Since reliability data for many of the components used in an ion engine do not exist, the component failure rates of similar equipment used in "airborne" applications were selected.

Table 6 Thruster failure rates

Subassembly	Unit failure rate, failure/10^9 hr 5-cm thruster	30-cm thruster
Thruster shell and structure	2,475	5,988
Optics and supports	2,698	3,300
Cathode isolator, vaporizer	2,995	3,178
Neutralizer, vaporizer	2,408	2,525
Main isolator, vaporizer	...	1,356
Total	10,576	16,347

Because components are subject to far greater stress in an airborne environment than in the space environment (other than the relatively short boost phase), the failure rates were adjusted by a K factor of $\frac{1}{200}$. This factor, which is suggested in Martin Company's "Reliability Data Handbook" to convert data from airborne to space environment, brings the NEDCO I and II failure rates into good agreement with the "generic" failure rate range of the Martin data. Furthermore, experience by the HAC Space and Communication Group has shown that observed failure rates are in good agreement with the Martin data, scattered randomly between the specified upper and lower limits. In final analysis, the NEDCO I and II data were used because they are more extensive than any other compilation of mechanical reliability data available at present.

The failure rates for the major subassemblies of the 5-cm and 30-cm thrusters are tabulated in Table 6. As indicated previously, the total failure rate of each subassembly was obtained by determining the failure rates of the parts (or of similar parts) which make up the subassembly and the length or number of TIG welds, spot welds, electron beam welds, and brazes required for assembly. The failure modes considered were:

1) structural weld failure; 2) sealing weld failures; 3) insulator failures; 4) structural failures; and 5) heater failures.

Reservoir failure rates

The reservoir failure rate was calculated for a single spherical tank as a function of its capacity in pounds of mercury. The system includes expulsion systems, filling values, and welded joints and assumes a safe operating pressure of four atmospheres with a 43% safety factor. The failure rates for component parts were obtained from data in the sources listed in the previous section on thruster failure rates. Again the unit failure rates were adjusted by a K factor of $\frac{1}{200}$ as recommended by the Martin Company's "Reliability Data Handbook" to convert data from airborne to space environment. Total estimated reservoir system failure rate versus mercury propellant capacity is shown in Fig. 2.

Power processor unit failure rates

The power processor unit is the collection of power circuitry and control logic required to operate and control an ion thruster

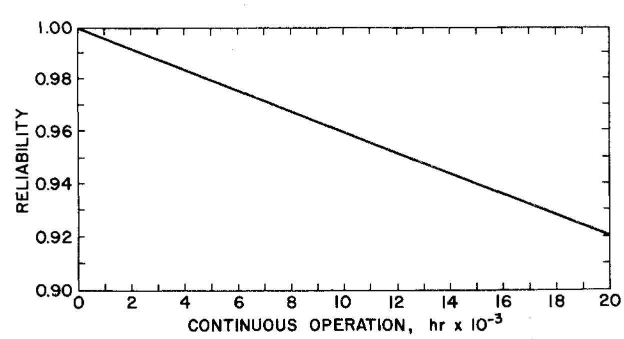

Fig. 3 30-cm thruster power processor reliability.

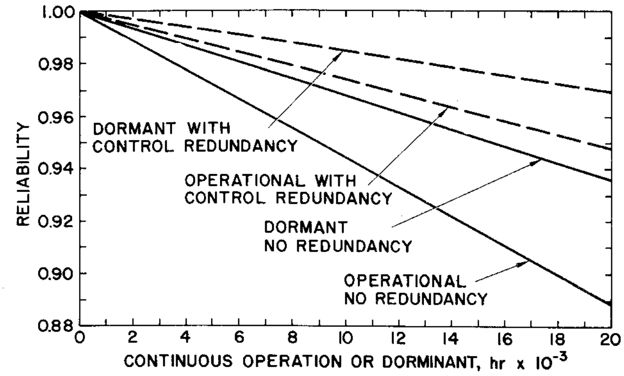

Fig. 4 5-cm thruster power processor reliability (for operational and dormant conditions).

from a solar panel bus, for example. At the present time, power processor units for both the 5-cm and 30-cm mercury-bombardment thrusters are under development. Availability of flight versions of these units is scheduled for late 1975.

The reliabilities of the 30-cm and 5-cm ion thruster PPU are given in Figs. 3 and 4. Figure 3 presents the 30-cm thruster PPU reliability with the screen and discharge inverter redundancy. Figure 4 on the other hand, shows the reliability of the 5-cm thruster PPU with no redundancy (i.e., the reliability based simply on parts count). However, also shown is the impressively large increase in system reliability (from a circuit standpoint) by the simple addition of a low-weight redundant control module.

The results of the previous analyses of ion thruster PPU units have shown that by the use of partial redundancy techniques relatively high subsystem reliability can be obtained with nominal weight penalties.

V. Effect of Reliability Considerations on Propulsion System Designs

As indicated previously, the failure modes associated with early and wearout failures of the major subsystems of an ion propulsion system are at this time fairly well understood and thus predictable. This understanding allows for the design of burn-in testing procedures which can eliminate the concern of infant mortality of flight-qualified sytems. In addition, by proper design of both mission and propulsion system, required operating times can always be made less than the wearout life limitations of any of the system components. Furthermore, the chance failure rates and thus reliabilities of the power processor subsystem (because it is generally made up of statistically tested electronic components and its circuitry is amenable to failure mode analysis) and the reservoir subsystem (because it is relatively simple and passive) are obtainable with adequate accuracy. The major remaining uncertainty in evaluating total ion propulsion system reliabilities is the exact value of the random or chance failure rates of ion thrusters. Since the establishment of thruster reliability at high confidence levels via testing would be both time consuming and costly (especially since 5,000–20,000 hr of operation is required for most space missions), other methods for increasing total propulsion system reliability such as redundancy must be considered. Redundancy of course, can lead to severe system mass penalties. Since increased mass is undesirable in any space system, it is important that these penalties be minimized.

Solar-Electric (Primary) Propulsion

An approach or methodology[23] based on extensive analyses and system design considerations has been developed for the purpose of designing prime solar electric propulsion (SEP) systems for space vehicles. It is based on the design concept of seeking a minimum mass system while maintaining the system reliability at or above a given level.

Propulsion System Designs and Reliabilities

Examples of the application of this methodology to a number of solar electric propulsion system designs are provided in Ref. 24. In this study eight potential SEP spacecraft missions were evaluated and optimum propulsion system designs for each mission were generated. The optimization procedure (as formulated in the EPSTOP computer program) used as inputs the low thrust trajectory data and provided as outputs specification of the optimum numbers (from a reliability-weight-power matching standpoint) of initially operating and standby thruster and PPU modules, the manner in which these modules should be switched on and off, and the required degree of thruster throttling. Since it was recognized that the chance failure rate of the ion thruster is not established, the sensitivity of the system reliability to a wide range of assumed thruster module failure rate was evaluated. The thruster module and PPU failure rates assumed for this study are given in Table 7. The thruster failure rates are presumed constant throughout their lifetime. Three different failure rates are indicated for the thrusters. The lowest failure rate represents the minimum which is expected. The largest failure rate is estimated to be an upper bound on the failure rate which might be experienced.

Table 7 Component and subsystem failure rates (failures in 10^9 hr)

Thruster failure rate	30-cm thruster module
Minimum	6,400
Intermediate	25,600
Maximum	64,000
Power processor unit failure rate	$1{,}670 + 70\dfrac{T}{(10{,}000)}$
	where T = thrust time in hours

Table 8 Optimum thrusters and PPU system designs

Missions	Number of thrusters Total	Number of thrusters Maximum operating	Number of PC panels	Thruster and PC panel mass, kg	System reliability Thruster failure rate Minimum	Intermediate	Maximum
Eros rendezvous	6	4	5	105	1.000	0.989	0.885
Encke rendezvous	8	4	5	119	0.998	0.995	0.913
Mars orbit and return	12	4	5	147	0.994	0.991	0.845
Jupiter flyby/Saturn probe	5	4	4	85	0.987	0.974	0.914
Saturn orbiter	8	6	6	125	0.985	0.983	0.959
0.1 a.u. solar probe	13	8	9	204	0.996	0.993	0.857
Mercury orbiter	15	7	8	206	0.991	0.990	0.896
Geosynchronous orbit and return	8	7	7	144	0.997	0.995	0.942

Table 9 Summary of baseline propulsion systems characteristics

Propulsion system baseline design	Mission	Total power, kw	Propulsion weight,[a] kg	Thruster array: Total number	Thruster array: Maximum number operating	PPU system: Number of PPU panels	Thruster array and PPU system reliability[b]	Reservoir: Number	Reservoir: Reliability	Switching matrix: Reliability	Total system[c] (Including propulsion): Weight, kg	Total system[c] (Including propulsion): Reliability
No. 1	Eros	12	506	8	4	5	0.999	1	0.983	0.99950	648	0.982
	Encke	12	502	8	4	5	0.995	1	0.956	0.99950	644	0.951
	Mars orbit	12	706	8	4	5	0.962	1	0.954	0.99950	857	0.917
No. 2	Jupiter flyby	18	443	8	4	6	0.984	1	0.974	0.99935	599	0.958
	Saturn orbiter	18	576	8	6	6	0.983	1	0.968	0.99968	737	0.951
No. 3	1 a.u. solar probe	21	822	12	8	8	0.898	1	0.987	0.99927	1042	0.886
	Mercury orbiter	21	1584	12	7	8	0.923	1	0.977	0.99913	1830	0.901
	Geosynchronous orbit	18	1584	12	7	8	~ 1.0	1	0.985	0.99913	1830	0.984

[a] Propellant weights include 5% contingency.
[b] Reliability based on intermediate thruster failure rate.
[c] System weights and reliabilities do not include cabling or mechanisms.

As indicated, the over-all failure rate of the power conditioning panels is lower than that of the thrusters because of internal subsystem redundancy. This same redundancy also leads to a power conditioning system failure rate which is not constant but increasing in time.

The recommended baseline modularized thruster and PPU system designs for the trajectory profiles assumed are shown in Table 8. It should be noted that these designs provide a reasonably high probability of successful mission completion even if the maximum assumed thruster failure rate is experienced. In addition, these designs represent the minimum mass systems for the individual missions under consideration. Incorporation of additional thrusters or power conditioning panels for other reasons (e.g., commonality of designs between missions) would obviously be acceptable from a reliability standpoint.

A major objective of the study cited above was to find a limited number of baseline system designs (i.e., thruster/PPU combinations) which could adequately perform all the missions under consideration. Inspection of the optimum designs (e.g., see Table 8) suggest the definition of the three baseline designs defined in Table 9. Table 9 shows the reliabilities of the selected thruster array and PPU systems for each of the eight missions. Comparison of data in Tables 8 and 9, show that in some cases the baseline system has a higher or lower reliability than the corresponding "optimum." In these cases more or less redundancy (and, therefore, weight) is incorporated in an attempt to gain commonality.

To completely specify over-all propulsion system reliability, consideration must now be given to the circuitry required to switch the various thrusters and PPU's and to the liquid mercury propellant storage system. The switching of thruster modules and power conditioning panels during a mission is performed by a logical controller and a power switching matrix. The general operation of these subsystems is as follows. At the beginning of a mission the programmer assigns a power conditioning panel to each operating thruster (i.e., closes the appropriate switches in the power switching matrix), and commands the power conditioning panels to turn on the thrusters. The start-up sequence is provided by the panels. Panels and thrusters are then directed to switch on and off, or to throttle thrust as required during the mission. If an operating thruster fails, the failure monitor sends an operation interrupt signal to the programer which in turn shuts off the power conditioning panel for the failed thruster and opens the power switches. A standby panel and thruster are then activated or the panel previously in operation can be transferred to a standby thruster and brought on line. Power conditioning panel failures are also monitored and replacement panels, if available, are substituted for failed ones.

The weight and failure rate of the switching circuitry will be principally that of the power switching matrix. The logical controller will involve digital circuits that are complex but are inherently lightweight and can be made highly reliable through the use of redundancy.

The weight and reliability of the power switching matrix were estimated for all the missions in the three baseline designs. As a typical example of the switching requirements consider propulsion system baseline design No. 1. In accordance with the data in Table 9, the switching matrix couples eight thrusters with five power conditioning panels. Each of these missions begins with four operating thrusters and four standby thrusters. The four initially operating panels are connected to their respective operating thrusters, but also have the capability of operating any of the standby thrusters. One standby panel has the capability of operating any of the eight thrusters. These switching capabilities were assumed in the calculation of the thruster and power conditioning reliabilities. It should be noted that the switched off thrusters can be operated by any of the five panels but a switched off panel can only operate five of the eight thrusters. Therefore, the switched off panels are somewhat limited as standbys. This limitation is justified by the high panel reliabilities (they have internal redundancy) and by the increased switching complexity that would be required for its removal.

Each of the switches actually represents a parallel recombination of nine relays. It can be shown that there are 28 of the basic switching groups for each of the three missions in propulsion system baseline design No. 1. The number of groups in the various missions for baseline designs No. 2 and No. 3 have been calculated and can be shown to vary from 18 for the Saturn Orbiter to 54 for the Mercury Orbiter and Geosynchronous Orbiter. The resulting reliabilities (tabulated in Table 9) were calculated using a single operation reliability conservatively estimated at 0.999999, and an average total number of relay operations is twice the total number of relays.

As stated in Ref. 23, a propellant reservoir weight-reliability optimization based on modularization is not as appropriate as it is for the thruster and power conditioning subsystems. The optimum reservoir system configuration (assuming no other constraints), therefore, appears to consist of a single tank with redundancy in its expulsion system. Furthermore, the number of valves should be kept to a minimum (possibly one) to carry out system operation (although redundant valves may be employed). Thus, in each of the baseline propulsion systems a single reservoir was employed and sized to the specific propellant requirement of the particular mission. The reliabilities of these various sized reservoirs were obtained from Fig. 2.

A summary of the weights and reliabilities of the ion propulsion systems for the eight baseline missions is given in Table 9. (Statements concerning items that have been included in these estimates should be carefully noted.) These results indicate that through redundancy techniques relatively high system reliabilities can be achieved for most SEP missions, even though long operating times are involved.

Satellite Control (Auxiliary) Propulsion

As in the case of SEP, numerous design studies on ion engine satellite control systems have been carried out. In general, the

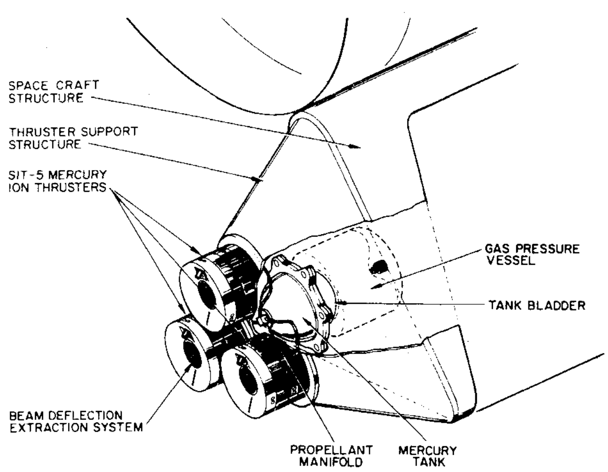

Fig. 5 ***AC/SK*** **thruster station.**

satellite control applications require system operating times up to twice that of the SEP missions (e.g., as much as 20,000 hr). For this reason reliability considerations are especially critical in the design of these systems.

Typical of ion propulsion satellite control applications is the attitude control and stationkeeping of synchronous satellites. For example, consider the design of a synchronous satellite attitude control and stationkeeping (*AC/SK*) system based on deflectable beam ion thrusters in combination with reaction wheels.[25] One control system configuration option which would satisfy all of the requirements of a synchronous satellite *AC/SK* system consists of three thruster stations. Two of these stations would be mounted at the base of the satellite with their nominal thrust vectors (parallel to the satellite longitudinal axis but offset from the center of mass) pointed north (or south) while the remaining station would be mounted on the body of the satellite so that it would nominally thrust through the center of mass in a west (or east) direction. North-south stationkeeping is unidirectional and performed during a 60° sector of the orbit at the time of maximum correction influence. This correction mode requires the two N-S thruster stations to operate in a continuous manner four hr per day. Thus, these thrusters must be turned on and off 365 times per year during which time they will accumulate 1460 hr of steady-state operation.

While providing the stationkeeping function, beam vectoring is used to produce three-axis control torques to allow reaction wheel energy to be reduced. Periodically during the orbit (but not during N-S stationkeeping), the E-W thruster is operated to compensate for solar pressure and earth triaxiality effects. This correction would occur five times per day for a total of three hours of operation.

Because of this cycling requirement, as well as the long duration of typical satellite control missions, reliability considerations have dictated that each thruster station be composed of three thrusters in a cluster. In this way, each control function has at least threefold thruster redundancy. The power conditioning unit associated with each thruster station is designed to operate a single thrust unit. In the event of a thruster failure, switching is provided to transfer the conditioned power to a standby. Power conditioning reliability is increased to the desired level by internal redundancy rather than by separate standby systems.

The thruster power processing unit (PPU) consists of three integrated units which share common housekeeping bias supplies, sequential circuitry and drive electronics in order to reduce total power conditioning weight. Between the PPU and the three thruster stations is a switching matrix mechanized with latching relays. This allows rerouting of power to a redundant thruster within a given thruster station and provides a redundant station power conditioning capability (only a maximum of 2 of 3 power conditioners are required to operate at a given time) in the event of a partial failure in the PPU. To minimize thruster power cabling weight, the thruster select switching would be located near its associated thruster station and the station select switching would be located with the PPU.

Individual thruster on/off commands and switch-state controls are issued from the command distribution circuitry. Thruster beam vector control signals from the attitude control electronics provide two-axis deflection analog reference signals for the closed-loop electrostatic deflection power supplies in the PPU.

As indicated, a total of three thruster stations are provided for attitude control and stationkeeping. Each station (see Fig. 5) is equipped with three separate ion thruster units which share a common liquid mercury feed system. Only one thruster per station is needed to fulfill mission thrust requirements, while two redundant thrusters per station ensure high propulsion reliability with minimal mass penalty since the mercury reservoir constitutes the major mass contribution of the thruster system.

VI. Conclusion

Based on the successes of SERT II, both the flight qualification and the long duration operation in space of ion propulsion systems have been established. Based on the extensive, developmental, laboratory life testing of ion propulsion systems, subsystems, and components, early failure modes are well understood (and preflight checkout procedures defined) and wearout lifetime projections of 15,000–20,000 hr are justified. Thus, the major remaining question appears to be that of system reliability during the required operating life of an ion propulsion system (i.e., an assessment of chance failure modes).

Because of the lack of statistical testing of fully developed systems, other approaches to determining chance failure rates must be considered at this time. Reasonable estimates of the chance failure rates of the three major ion propulsion subsystems—thruster, reservoir, and power processor unit—can be obtained by an evaluation of component part failure rates and with the use of accepted analytical techniques. For example, the over-all failure rate of the power processor subsystem can be readily obtained because these units are made up of well-documented space qualified electronic components and reliability analysis of electronic circuitry is well established. Similarly, the propellant reservoir subsystem failure rate can be determined with good accuracy because these units are relatively simple in design and passive in operation. Whereas ion thruster failure rates are more difficult to access, failure mode analyses, comparison with similar devices and components, and knowledge of construction features do lead to reasonable estimates.

Furthermore, using the chance failure rates obtained for the major ion propulsion subsystems it has been shown that through the judicious use of redundancy, relatively high system reliabilities can be obtained for most primary and auxiliary ion propulsion applications without major weight penalties.

Thus, while statistical testing of fully developed, flight qualified, ion propulsion systems should be encouraged and while additional flight tests are desirable, relatively high system reliabilities can be predicted with confidence based on the use of good system design practices, available test data, and accepted analytical techniques.

References

[1] Rawlin, V. K. and Kerslake, W. R., "Durability of the SERT II Hollow Cathode and Future Applications of Hollow Cathodes," *Journal of Spacecraft and Rockets*, Vol. 7, No. 1, Jan. 1970, pp. 14–20.

[2] Kerslake, W. R., Goldman, R. G., and Nieberding, W. C., "SERT II: Mission Thruster Performance and In-Flight Thrust Measurements," *Journal of Spacecraft and Rockets*, Vol. 8, No. 3, March 1971, pp. 213–224.

[3] Rulis, R. J., "SERT II: Design Requirements for Integrating Electric Propulsion into Spacecraft," *Journal of Spacecraft and Rockets*, Vol. 8, No. 3, March 1971, p. 212.

[4] Rawlin, V. K., "A 13,000-Hour Test of a Mercury Hollow Cathode," TMX-2785, June 1973, NASA.
[5] Nakanishi, S. and Finke, R. C. "A 9700-Hour Durability Test of a Five Centimeter Diameter Ion Thruster," *Journal of Spacecraft and Rockets*, Vol. 11, No. 8, Aug. 1974, pp. 560–566.
[6] Lathem, W. C., "5000-Hour Test of a Grid-Translation Beam-Deflection System for a 5-cm Diameter Kaufman Thruster," TMX-68185, Jan. 1973, NASA.
[7] Nakanishi, S. "Durability Tests of a Five-Centimeter Diameter Ion Thruster System," AIAA Paper 72-1151, New Orleans, La., 1972.
[8] Hyman, J., "Performance Optimized Small Structurally Integrated Ion Thruster System," Final Repts., NASA Contract NAS 3-15483, May 1973, Hughes Research Labs., Malibu, Calif.
[9] Lathem, W. C., "1000-Hour Test of a Dual Grid, Electrostatic Beam Deflection Accelerator System on a 5-Centimeter-Diameter Kaufman Thruster," TM X-67907, Aug. 1971, NASA.
[10] Lathem, W. C., "A Single-Axis Electrostatic Beam Deflection System for a 5-cm Diameter Ion Thruster," TM X-68133, Sept. 1972, NASA.
[11] Hudson, W. R. and Weigand, A. J., "Hollow Cathodes with BaO Impregnated, Porous Tungsten Inserts and Tips," TM X-71417, 1973, NASA; also AIAA Paper 73-1142, Lake Tahoe, Nev., 1973.
[12] Zuccaro, D., "Mercury Vapor Hollow Cathode Component Studies," AIAA Paper 73-1141, Lake Tahoe, Nev., 1973.
[13] Molitor, J. H. and Olsen, R. N., "Development and Test of an Ion Engine System Employing Modular Power Conditioning," Final Rept., JPL Contract 951144, 1966, Hughes Aircraft Co., Malibu, Calif.
[14] "Discharge Chamber Studies for Mercury Bombardment Thrusters," Final Rept. NASA CR-72440, 1968, Hughes Research Labs., Malibu, Calif.
[15] King, H. J. et al., "Electron Bombardment Thruster Using Liquid Mercury Cathodes," *Journal of Spacecraft and Rockets*, Vol. 4, No. 5, May 1967, p. 599.
[16] Masek, T. D., "SEP Thrust Subsystem Development," JPL TR 32-1579, March 1973, Jet Propulsion Lab., Pasadena, Calif.
[17] King, H. J. and Poeschel, R. L., "A 30-cm Mercury Ion Thruster," *Journal of Spacecraft and Rockets*, Vol. 8, No. 4, April 1971, p. 420.
[18] King, H. J. and Poeschel, R. L., "An Ion Thruster Module for Primary Propulsion Systems," *Journal of British Interplanetary Society*, Vol. 26, No. 203, 1973, pp. 203–219.
[19] Collett, C. R., "Endurance Testing of a 30-cm Kaufman Thruster," AIAA Paper 73-10850, Lake Tahoe, Nev., 1973.
[20] Mirtich, M. J., "Investigation of Hollow Cathode Performance for 30-cm Thrusters," AIAA Paper 73-118, Lake Tahoe, Nev., 1973.
[21] Mantenicks, M. A., "Investigations of Mercury Isolators," AIAA Paper 73-1088, Lake Tahoe, Nev., 1973.
[22] Rawlin, V. K., "Studies of Dished Accelerator Grids for 30-cm Ion Thrusters," AIAA Paper 73-1086, Lake Tahoe, Nev., 1973.
[23] Russell, K. J. et al., "Electric Propulsion Design Optimization Methodology," *Journal of Spacecraft and Rockets*, Vol. 7, No. 2, Feb. 1970, pp. 164–169.
[24] Gilbert, J. and Guttman, C. H., "Evolution of the Common SEP Stage (COM-SEPS)," AIAA Paper 73-1122, Lake Tahoe, Nev., 1973.
[25] Gould, L. M., "Applications Technology Satellite Advanced Mission Study," Final Rept., NASA 3-14359, 1972, Hughes Aircraft Co., Malibu, Calif.

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Space Shuttle Bipropellant RCS Engine

M. L. CHAZEN* AND W. SANSCRAINTE†
Bell Aerospace, Division of Textron, Buffalo, N.Y.

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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* Program Manager/Technical Director, Shuttle RCS Engine.
† Chief, Space Propulsion.

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