High-Thrust Throttleable Monopropellant Engine System

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Recent Mars soft-landing studies show that a propulsion system is required to accomplish the terminal landing sequence. Relatively high thrust levels and throttling ratios are required and engine weight is extremely significant since it comprises nearly one-third of the propulsion system weight due to the relatively small total impulse which is required. To overcome potential problems with scaling conventional catalytic thrusters, a pilot-chamber-initiated, thermal decomposition reactor concept has been studied. The pilot chamber operates catalytically on 5% to 15% of the total flow. The remainder reacts thermally in the main chamber which is void of catalyst. The characteristics of such a chamber have been experimentally studied and it shows high potential for further development. Operation at design values of $L^* = 150$ in and $G = 0.1$ lbm/in.^{2}-sec appears completely feasible if a suitable heat sink "flame holder" is utilized.

Introduction

THE development of the Shell 405 iridium-based spontaneous catalyst has given rise to a wide variety of flight applications[1–3] for engines operating in both the pulse and steady-state modes over fixed and variable thrust profiles in the range of less than 1.0 to 300 lbf of thrust. However, this technology has not to date been qualified at higher thrust levels because monopropellant systems are usually not weight-competitive for applications which require more than 100,000 to 300,000 lbf-sec of total impulse. In general, these low total impulse applications do not require thrust levels as high as 300 lbf.

Recent studies[4,5] of Mars soft-landing spacecraft in the 1000 to 2000 lbm range have resulted in a requirement for a propulsion system to accomplish the terminal descent portion of the landing sequence. This requirement results from tradeoff studies in which the weight of such a terminal descent rocket system was optimized against the weight of the parachute required to reach an acceptable terminal velocity and the weight of the energy absorbing medium required to attenuate the landing shock to values tolerable by the instruments. Such optimizations show that a propulsion system should be used but that only a relatively modest total impulse (~30,000 lbf-sec) is required. When the additional requirements of ultra reliability, heat sterilization and prevention of planetary surface contamination with water or carbonaceous exhaust products are added, use of a monopropellant thruster is suggested. However, while the total impulse required is modest, the time available for its application is restricted by guidance considerations with the result that thrust levels in the range of 400 to 1500 lbf are required. In addition, throttling ratios of 5:1 to 10:1 are required of the thruster.

In recognition of these prospective requirements, an independent research and development study was undertaken at TRW in 1967 to demonstrate advanced concepts which would yield competitive engine weights when scaled-up to thrust levels greater than 600 lbf. Engine weight is particularly significant in this application since it accounts for one-third or more of the total system weight. As a starting point for this study, conventional catalytic chamber design techniques were considered. Problems noted in extending this technology to higher thrusts were: 1) a loss in performance at low thrust due to excessive ammonia dissociation, 2) a structurally inefficient catalyst bed geometry which yields excessive chamber shell weight, and 3) a large amount of catalyst is required (weight, cost and availability problems).

As a fixed-area catalyst bed is throttled, the flow rate per unit cross-sectional area G decreases, increasing the effective residence time of the exhaust gases in the chamber. The increased residence time yields increased dissociation of the ammonia produced from the hydrazine decomposition process. Plots showing the increase in ammonia dissociation X due to variations of G in a fixed geometry catalytic reactor, are presented in Ref. 6. A typical sample from this source which illustrates this point is shown in Fig. 1. The effect of the increased ammonia dissociation (an endothermic process) on the performance variables C^* and I_{sp} is shown in Fig. 2. The performance decreases 5% when the ammonia dissociation changes from 40 to 80%. Such a change in ammonia dissociation may be anticipated when a well designed reactor is throttled over a 10:1 flow range.

When monopropellant reactors are designed according to established relationships,[6,7] a cylindrical bed results with a length-to-diameter ratio which decreases as thrust level increases. As thrust level increases, this phenomenon leads to shallow thrust chambers of excessive diameter. Figure 3 shows a conceptual design of a 500-lb-thrust hydrazine engine. This figure illustrates the "pancaking" effect. The

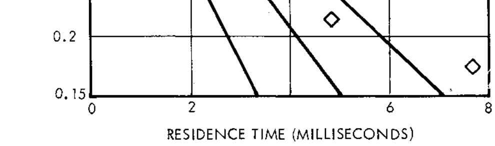

Fig. 1 Effect of bed loading on ammonia dissociation fraction (X) for a 5 lbf engine (P_c = 225 psia).

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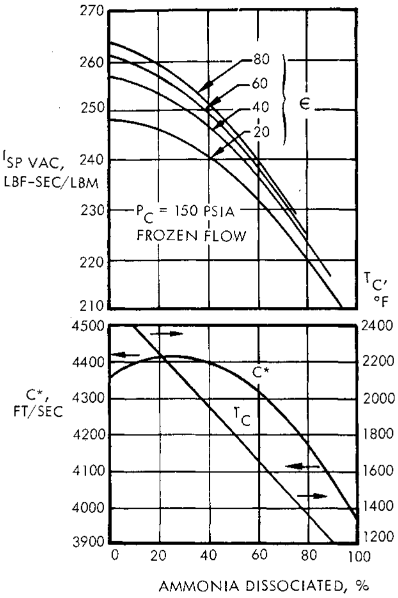


Fig. 2 Characteristic velocity, adiabatic gas temperature, and vacuum specific impulse of monopropellant hydrazine.

weight of the catalyst chamber becomes excessive because the large diameter of the cylindrical portion of the chamber requires a thick and heavy pressure shell and the relatively flat plates used above and below the catalyst bed are poor structural members.

For higher thrust, the problem becomes more acute. Figure 4 shows a design study of a 1500-lb thruster. In this case, the plates upstream and downstream of the catalyst bed have been slightly domed, but the weight predicted for this thrust chamber is still excessive. For comparison, the weight of a bipropellant engine for the same application would be expected to be less than one-half the thrust chamber weight quoted. Moreover, if three engines like the one shown in Fig. 4. were to be used on a landing vehicle, the amount of catalyst required would be approximately 40% of the annual production rate of Shell 405.[8] The cost of this catalyst would be ~\$100,000. Although neither of these figures is prohibitively large, their magnitude is such as to be a serious consideration in the eventual use of such an engine.

Design Concept and Testing

To overcome these problems, a pilot-chamber-initiated, thermal decomposition reactor concept was suggested that

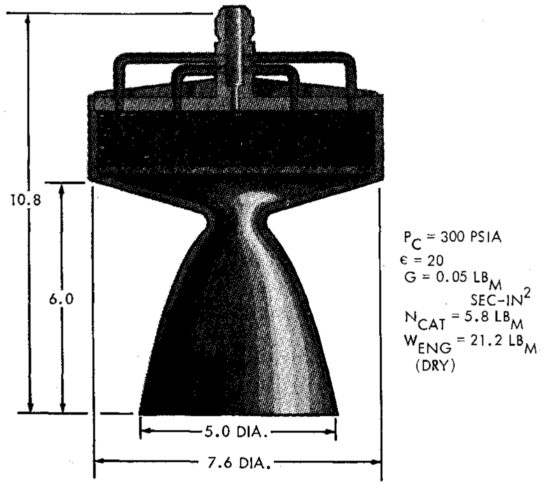


Fig. 3 Conceptual design for a catalytic 500 lbf thrust chamber for N_2H_4.

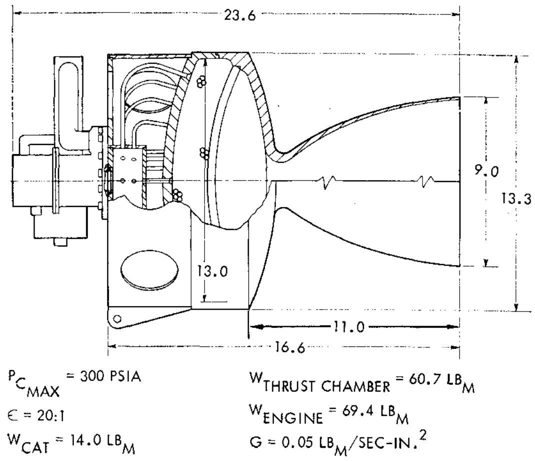


Fig. 4 Design study, 1500 lbf catalytic hydrazine engine.

retains the advantage of operating spontaneously, reduces the quantity of the catalytic material and yields a more compact chamber (Fig. 5). The pilot chamber operates conventionally in the catalytic mode on a portion (5–15%) of the total flow. The remainder of the flow reacts thermally in the main chamber which is void of catalyst. Experiments have shown that with proper design, the main thermal decomposition chamber can be operated at mass flow rate loadings (G's) four to five times those currently feasible in catalytic chambers.

High performance is attained over extended throttling ranges, because the effective ammonia dissociation is limited in a thermal reactor regardless of reactor length. The dissociation of the ammonia in a thermal reactor does not proceed in a truly heterogeneous manner, so that discrete values of the dissociation fractions can not be correlated with reactor length or effective residence time. While the kinetics of hydrogen generation in a thermal reactor are not well understood, it has been postulated[8,9] that the reaction proceeds slowly through a complex series of dissociations and recombinations of NH radicals producing an equivalent ammonia dissociation of 25 to 40%. Performance is not independent of reactor length, however, because the heat losses at a given thrust level become appreciable with high values of reactor L/D. This explains why thruster performance for thermal decomposition reactors is a much weaker function of thrust level than for catalytic reactors.

The design requirements and parameters for Fig. 5 were identical to those used for the thruster in Fig. 4. A major portion of the weight advantage for the pilot/thermal engine is caused by the reduction in chamber diameter.

Thermal decomposition of hydrazine antedates its use in catalytic chambers (i.e., pre-1950[10]). Electrical heating elements and hypergolic slug starts were used to promote ignition. Most of the early experiments were conducted with large L^* values (~400 to 500). However, several experimenters were able to demonstrate stable operation using L^*'s in the range of 150 to 250 with chamber mass loadings ranging up to 0.2 to 0.25 lbm/in.^{2}-sec. While a pilot chamber concept is certainly not novel in terms of prior use with monopropellants, this is the first known combination utilizing the spontaneous catalyst. In addition, the application of this approach to the Mars lander mission is viewed as a contribution to the art, since it has the potential to fulfill the unique requirements of this mission.

Feasibility experiments were run to determine whether stable operation could be sustained at design conditions which would make the concept attractive ($L^* \simeq 150$ in., $G \simeq 0.1$ lbm/in.^{2}-sec) for the Mars lander and to determine the requirements for initiation of the reaction. Nineteen tests were completed and operation at $L^* = 325$ in. and $G = 0.07$

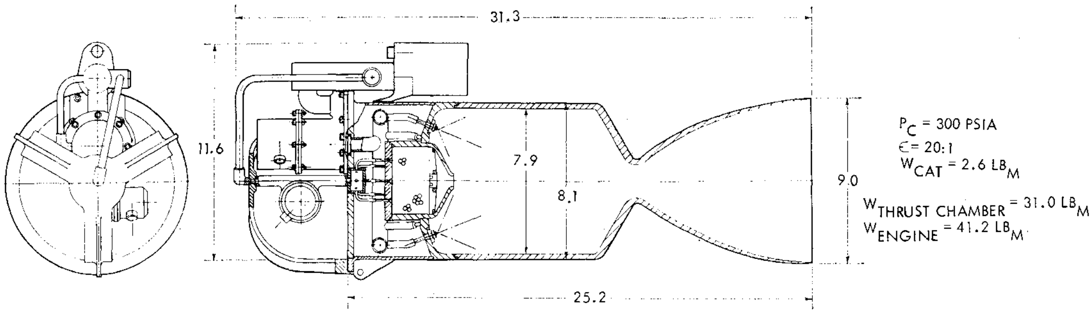


Fig. 5 Design study, 1500 lbf thermal decomposition thruster with catalytic pilot.

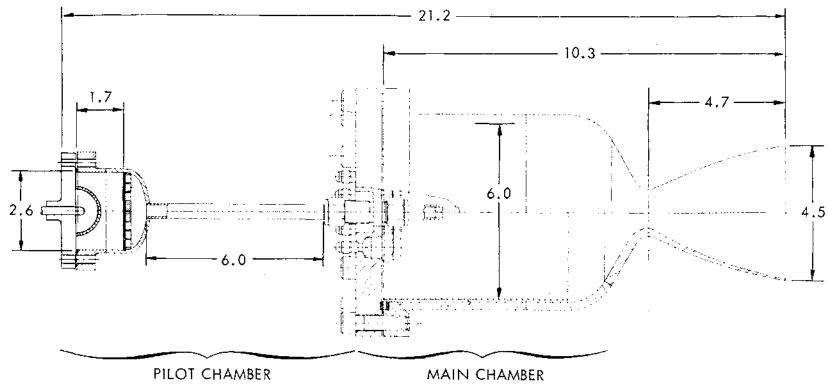


Fig. 6 400 lbf test thruster assembly.

lbm/sec-in². was demonstrated. While only a few successful ignitions were obtained, the engine ran smoothly with exceptionally high performance on the successful runs. It was felt that most of the ignition problems were due to the heavyweight nature of the hardware used in these preliminary tests.

A program was initiated in the spring of 1968 to design, fabricate and test a 400-lb-thrust to engine determine performance at low L^*'s and high G's over a range of thrust levels. The engine (Fig. 6) was designed as a subscale test bed with $\epsilon = 20$ and $P_c = 300$ psia at full thrust. The pilot chamber had the same bed loading and injector as a previously developed 50-lb-thrust engine.

The engine was designed with a bolted flange in order that various internal configurations could be tested. The configurations included two internal liners of 5.6 and 4.65 in. in diameter in order to vary L^* and various heat-transfer configurations to improve performance and start-up time. In order to test the engine at $L^* < 150$ in., the engine was decreased in length approximately 3 in. midway in the test program.

Six commercial spray nozzles were used as the main flow injectors. Three nozzles utilized a 60° spray angle, while the others used a 45° spray angle.

Thirty-six tests were made with hydrazine over a thrust range of 50 to 400 lbf with L^*'s of 242, 169, 138, and 97.5 in. and G's from 0.01 to 0.1 lbm/sec-in.² The tests were conducted by first starting the pilot chamber and then initiating flow to a variable number of main injectors. Since the spray nozzles had fixed orifices, the low-flow atomization of the nozzles was undesirable. Therefore, the nozzles were connected so that one spray jet, two spray jets, or three spray jets could be used in any combination for main engine flow. The basic engine shown in Fig. 6 was initially tested both with and without its internal liner. The test schematic is shown in Fig. 7.

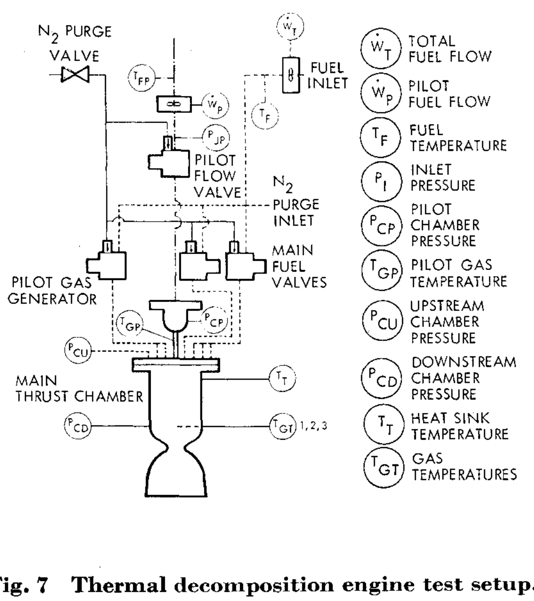


Fig. 7 Thermal decomposition engine test setup.

Test Results with Hydrazine

The initial 400-lbf hydrazine tests indicated that the engine has good performance at low L^*'s and high G's (Fig. 8).

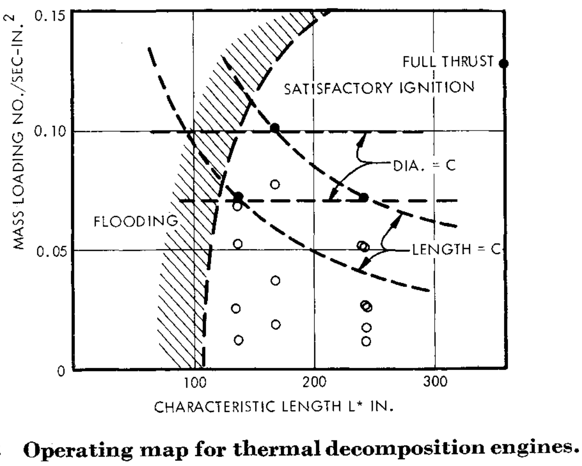


Fig. 8 Operating map for thermal decomposition engines.

Table 1 Test results for 400-lbf-thrust engine

Configuration[a]	Run no.	n_j[b] Start	t_p[c], sec	n_j[b] Start	$G \times 10^4$, lb/in.²-sec	P_c, psia	$\dot{W}_{\text{Pilot}}$, lb/sec	$\dot{W}_{\text{Main}}$, lb/sec	C^*, fps	T_{ct}, °F
A	135	2	25	2	189	78	0.0735	0.3837	4244	1804
	136	3	25	3	258	108	0.0727	0.5637	4314	1798
	137	6	25	1	525	222	0.0722	1.2210	4360	1957
	138	6	25	1	713	300	0.0634	1.6961	4334	2120
	139	1	15	1	124	N/A	0.0741	0.2324	N/A	1740
	140	1	10	1	126	N/A	0.0739	0.2365	N/A	1740
	141	1	10	1	126	N/A	0.0730	0.2364	N/A	1627
	142	3	10	3	262	110	0.0735	0.5727	4327	1762
	143	6	10	6	524	225	0.06743	1.224	4430	1976
	144	3	5	3	256	108	0.07373	0.5582	4345	1730
	145	3	3	3		Flooded				404
	147	1	25	1		Flooded				N/A
B	148	1	25	1	124	49	0.0749	0.2315	4076	1494
	149	3	15	3	259	109	0.07433	0.5637	4343	1859
	150	6	15	1	520	218	0.06843	1.213	4326	1799
	151	6	25	1	720	297	0.06354	1.712	4251	1832
C	152	1	25	1	184	N/A	0.0735	0.2394	N/A	1640
	153	3	25	3	378	108.5	0.0732	0.5670	4308	1799
	154	6	15	1	864	220	0.06701	1.2281	4319	2047
	155	6	25	1	1042	302	0.06396	1.706	4337	1924
Chamber reduced in length										
D	156	1	25	1	188	Flooded	0.0709	0.2486		1338
	157	1	25	1	188	Flooded	0.07102	0.2486		1372
E	158	1	25	1	185	Flooded	0.07588	0.2383		1269
	159	1	25	1	187	Flooded	0.07639	0.2402		1364
	161	1	25	1	189	Flooded	0.0753	0.2460		1429
F	162	1	25	1	126	49	0.0749	0.2363	4003	N/A
	163	3	15	3	259	105	0.0756	0.5641	4173	N/A
	164	6	15	1	526	221	0.0743	1.2239	4328	N/A
	165	6	15	1	720	305	0.0667	1.7122	4359	1781
	167	6	25	1	690	274	0.0809	1.707	4304	1776

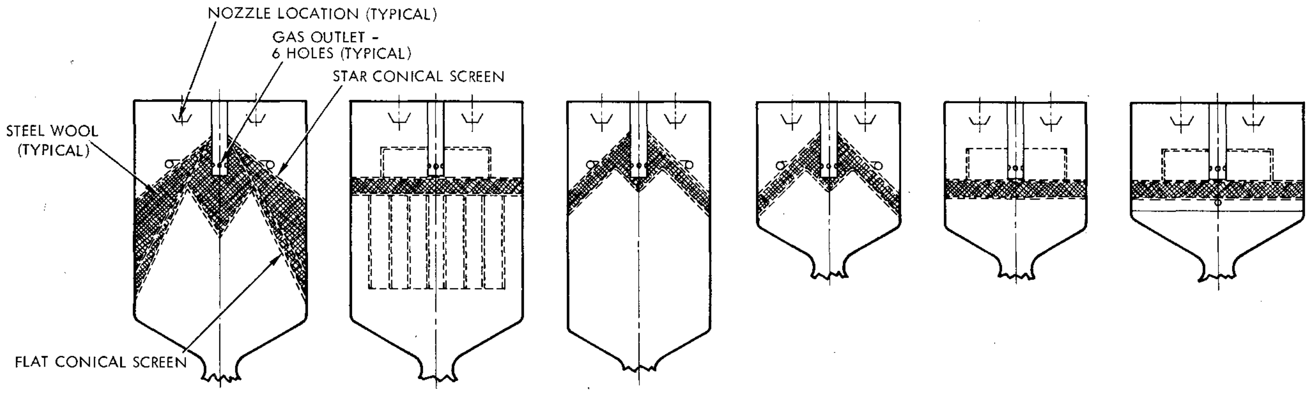
[a] See Fig. 9.
[b] n_j = number of jets.
[c] t_p = duration of pilot prior to main stage.

Flooding of the engine occurred to the left of the line shown in Fig. 8. However, it is believed that with the proper internal thermal configuration lower L^* and higher mass loading can be obtained.

The thermal decomposition engine does require a time delay between the pilot and main flow in order to obtain proper ignition. For this first series of tests about 4 sec of pilot flow was required before main flow was easily ignited. However, it also appears that with increased pilot flow and proper design of the internal thermal configuration this start-up time can be decreased to 2 to 3 sec using hydrazine as the fuel, with a probable absolute lower limit of ~1 sec with much development.

The internal configurations that were tested (Fig. 9) were constructed of stainless steel screens. Steel wool inserted between screen layers was found to be instrumental in promoting fast ignition. More testing is required in order to obtain the optimum configuration.

Table 1 summarizes the data obtained from the hydrazine tests. At a few low thrust points C^* dropped, due to the deterioration of the steel wool used in the internal configuration, with the probable result that some N_2H_4 vapor exited



CONFIGURATION	STAR/CONICAL	FLAT/CYLINDRICAL	FLAT/CONICAL	FLAT/CONICAL	FLAT	FLAT
D_C, IN.	5.6	5.6	4.65	4.65	4.65	5.6
USED IN RUNS	131-147	148-151	152-155	156-157	158-161	162-167
L*, IN.	242	242	169	97.5	77.5	138

Fig. 9 400-lbf engine internal test configurations.

Table 2 Test data for run 176

Point	P_c, psia	Flow rate, lb/sec	Thrust, %	C*, fps (not corrected for At)
1	99	0.563	33.0	4470
2	151	0.853	50.3	4500
3	203	1.149	67.6	4491
4	219	1.244	73.0	4475

through the throat. In all further tests metal fibers of a high-temperature superalloy are recommended for use in place of the steel wool.

Testing with $N_2H_4/N_2H_5NO_3$ and Flow Control

After the hydrazine testing was completed, additional tests were conducted with a propellant containing 77.2% hydrazine, 22.7% hydrazine nitrate and 0.1% water and ammonia. Before testing the combined engine and flow control valve, the engine injectors were changed to six variable-area hollow cone spray jets.

The engine design tested was almost identical to the design as last tested with hydrazine (run 167), except that a flow control valve was added to the engine, and the injectors were changed to variable-area, hollow-cone spray jets. The internal configuration was the flat design similar to run 167, however, with stainless steel wool ($L^* = 138$ in.). Only three tests were conducted due to the unsatisfactory large spray angle of the injectors compared to those used earlier. The spray angle as measured in the water flow tests was 100°–110°.

For each test the propellant tank was pressurized to 475 psia. For tests number 174 and 175 the pilot chamber was started at $t = 0$; at $t = 30$ sec, the main chamber was started at 30% thrust; at $t = 40$ sec, the valve was opened to 100% thrust. No flooding was noted with the pilot chamber in operation during main chamber firing. No chamber pressure oscillations or roughness were noted. No system instability was observed. During test no. 176 the main chamber did not reach 100% full thrust due to excessive feed system pressure drop. Data from test 176 are shown in Table 2. It is felt that the large spray angle contributed significantly to the fact that sustained operation could not be maintained without the pilot. Future tests will employ injectors with a lower spray angle.

Conclusions

The pilot/thermal concept shows high potential for further development. Satisfactorily high performance was achieved over the throttle range.

Operation at $L^* = 150$ in. and $G = 0.1$ lbm/in.^{2}-sec appears completely feasible, but some form of "ignition-aid" heat sink or "flame holder" must be provided in the thermal chamber to enable operation at low L^*'s with reduced pilot lead times. The engine does not appear overly sensitive to the precise configuration of the heat sink material. A minimum pilot chamber lead time of about 4 sec was demonstrated. Optimal design may allow this lead time to be halved.

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