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ABSTRACT

The results of an experimental program directed toward determining some of the operational characteristics of the nitrogen tetroxide–hydrazine propellant system under oxidizer-rich and fuel-rich conditions are reported. Data are presented for the mixture-ratio ranges of 0 to 0.55 at a nominal chamber pressure of 300 psia and characteristic chamber length of 250 in., and at mixture ratios of 5.9 to 13.0 at nominal chamber pressures of 400 and 500 psia and characteristic chamber lengths of 100 and 3900 in.

The relationship of the actual performance data obtained in each region with those predicted from thermochemical performance calculations is presented. From this comparison, it is concluded that in neither region are equilibrium conditions obtained and that utilization of performance data obtained from assumptions of equilibrium will lead to serious errors. In the oxidizer-rich region investigated, temperatures considerably below those predicted occur throughout the range of mixture ratios investigated because of the lack of exothermic dissociation of the nitrogen oxides. In the fuel-rich region, below a mixture ratio of 0.3, temperatures considerably higher than those predicted occur because of the lack of endothermic dissociation of ammonia.

I. INTRODUCTION

Liquid-propellant systems utilizing compounds which are readily storable at normal ambient temperatures enjoy considerable current interest for both military and space-vehicle applications. A superior bipropellant combination of this type is the system nitrogen tetroxide–hydrazine. Considerable research and development effort has been expended in defining the optimum theoretical performance of this system, its operational characteristics and vagaries, and in demonstrating its use in rocket engines and propulsion systems (Refs. 1, 2, 3, 4). The general interest in utilizing this propellant combination has also, quite naturally, produced interest in and requirements for information and data regarding the operational characteristics of the system for auxiliary applications. Typical applications of this type include turbo-pump energy sources, generated-gas tank pressurization systems, vernier rockets, midcourse propulsion systems, and attitude-control rockets.

In general, in the interest of obtaining simplicity and flexibility in an auxiliary system, it is desired that such devices operate at combustion-temperature levels considerably below those obtained at conditions of optimum performance. In addition, because of the peculiarities and uniqueness of a particular auxiliary application, it may be required that the products of combustion be specifically reducing or oxidizing in nature. The foregoing qualifications are most readily obtained by simply operating the bipropellant system in an off-mixture-ratio condition.

Specifically, the experimental program described in this Report was initiated to obtain oxidizer-rich data for use in a feasibility demonstration of a bipropellant gas generator operating on nitrogen tetroxide–hydrazine for use in a generated-gas pressurization system for the propellant tanks of the 6000-lb-thrust upper stage *Vega* vehicle (Ref. 5). Upon subsequent cancellation of the *Vega* system, the development of the generated-gas tank pressurization system was discontinued. The experimental effort on the combustion device, however, was broadened to include both the oxidizer-rich and fuel-rich regions in the interests of obtaining data applicable to propulsion auxiliaries in general.

A search of the literature indicated few theoretical or experimental performance data for the nitrogen tetroxide–hydrazine system in the high-mixture-ratio range, i.e., above a mixture ratio of 4 (4 parts by weight oxidizer to 1 part fuel) (Ref. 6), and only a single reference to theoretical performance in the low-mixture-ratio range, i.e., below 0.6 (Ref. 7).

It was recognized that the prediction of operating performance might be difficult in both the oxidizer-rich and fuel-rich regions with this particular propellant system because of the presence of chemical species which resist kinetically the approach to chemical equilibrium. In the oxidizer-rich region, such species consist of the nitrogen oxides. Nitric oxide is known to be a relatively stable compound, whose rate of decomposition has been reported by Wise and Frech to be small even near temperatures of 1500°K (Ref. 8). In addition, earlier calculations by Altman and Penner regarding the decomposition of NO during expansion through a rocket nozzle at temperatures below 2700°K have suggested that the nitric oxide equilibrium cannot be maintained because of the rapid temperature drop (Ref. 9). In the fuel-rich region, the presence of ammonia certainly would be expected to lead to nonequilibrium conditions. It is well known that under conditions such as are obtained during the monopropellant decomposition of hydrazine, chemical equilibrium is not obtained and that the kinetics of the ammonia dissociation can be manipulated through the use of appropriate catalysts to obtain reaction exit temperatures over an approximate temperature range of 1300 to 2200°F, as desired for a particular application (Refs. 10, 11). From such knowledge, it was recognized that the effects of the kinetics of ammonia dissociation would certainly be expected to be of significance in some portion of the low-mixture-ratio operating range and that thermochemical equilibrium computations would not yield data suitable for comparison with the experimental data.

II. EXPERIMENTAL PROGRAM

This investigation had the following objectives: (1) to determine the operational characteristics of the propellant system at oxidizer-rich mixture ratios of 5.0 to 14.0; (2) to determine the operational characteristics of the propellant system nitrogen tetroxide–hydrazine at fuel-rich mixture ratios between 0 (monopropellant-hydrazine operation) and 0.5; (3) to compare the actual operational data obtained with the theoretically predicted data from thermochemical calculations.

The choice of mixture-ratio ranges to be investigated was admittedly arbitrary in nature. However, it was generally felt that (1) in the case of the oxidizer-rich region, the limits chosen would bracket temperature levels most suitable for auxiliary propulsion applications (2) in the fuel-rich region, a mixture ratio of 0.5 appeared to be an appropriate upper limit for investigation because of the existence of theoretical and operating data in that region and above (Ref. 7) and because of the anticipated temperature level of approximately 3500°F obtained at that mixture ratio.

The following discussion is divided into two parts, one describing the high-mixture-ratio portion of the investigation and the other the low-mixture-ratio portion. Each part includes a description of experimental equipment used, the results of the experimental investigation, and a discussion relating the theoretical and experimental information pertinent to that region.

A. Oxidizer-Rich Region (High Mixture Ratio)

1. Experimental Effort

Two combustion devices were utilized in the experimental investigation of the oxidizer-rich performance of nitrogen tetroxide–hydrazine. Initially, a series of tests were conducted utilizing existing experimental hardware which had previously been developed and employed in an advanced-development generated-gas tank pressurization system version of the *Corporal* missile. The system had utilized a monopropellant-hydrazine gas generator to pressurize the fuel tank and an oxidizer-rich bipropellant gas generator using red fuming nitric acid and hydrazine to pressurize the oxidizer tank. This technique was quite successful; details of the development effort can be found in Refs. 12 and 13. A sketch of the bipropellant combustion device as used in the present investigation is shown in Fig. 1. The unit was 5 in. in diameter.

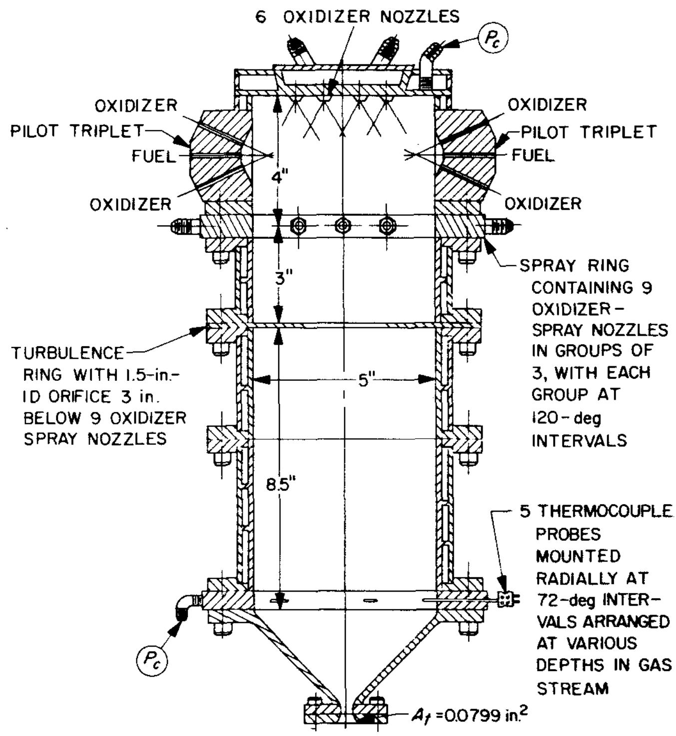

Fig. 1. Oxidizer-rich gas generator, $L^* = 3900$ in.

The top section of the generator included a pilot flame produced by two triplet injectors (two oxidizer jets impinging on a central fuel jet), mounted diametrically opposite each other on the chamber wall. Six spray nozzles were mounted in the head of the generator and injected highly atomized oxidizer through the pilot-flame region. Immediately below the top section of the generator, nine additional spray nozzles were mounted in a downstream diluent spray ring. In order to obtain a relatively uniform temperature profile across the chamber, a turbulence ring with a 1.5-in.-diameter orifice was mounted in the generator 3 in. below the diluent spray ring. The over-all characteristic length L^* of the unit as used in this investigation was 3900 in., approximately 2150 in. of which was located below the turbulence ring.

The gas generator was mounted for these tests as shown in Fig. 2. The top section of the generator was water-cooled; the remainder of the generator was uncooled. Five thermocouple probes were mounted at uniform radial increments in the chamber wall near the aft end of the generator, approximately 8 in. below the turbulence ring. These probes consisted of commercially available chromel–alumel sheathed thermocouples, with the junction exposed to the gas stream. The thermocouples were inserted to varying depths into the gas stream. The mean gas temperature reported is the arithmetic mean of the five measurements. These thermocouples were considered adequate because of the relatively low temperatures expected; however, a high thermocouple mortality rate was anticipated due to the oxidizing atmosphere. Chamber pressure was monitored at both the generator injector head and at the same station as the thermocouple probes. Chamber-pressure data from the nozzle entrance station were used in the calculation of performance. Flow rates were determined through the use of orifice plates and differential pressure transducers. In general, tests of 12- to 14-sec duration were made, and the data reported were obtained during the last second of operation. All of the tests were made at a nominal chamber pressure of 500 psia.

After testing of the aforementioned unit was completed, and utilizing the data and operating techniques developed from its operation, a gas generator suitable for use as a flight prototype model was designed and fabricated. As indicated in Section I, it was intended initially that this unit would be used to demonstrate the feasibility of a dual generated-gas tank pressurization system in conjunction with an advanced phase of the *Vega* program. Particular emphasis in the design was placed upon obtaining a suitable injector. A sketch of the gas-generator injector is shown in Fig. 3. This design consisted of a central fuel-inlet tube with a deflector baffle attached to the tip, thus forming a circumferential slot capable of discharging a high-velocity radial fan of liquid. The radial gap between the end of the tube and the baffle was 0.005 in. Surrounding the tube was a concentric oxidizer annulus which produced a lower-velocity, concentric spray. The width of the annulus was 0.050 in. Calibration tests using water indicated that where the central tube projected 0.235 in. from the injector face, the radial fan of fuel from the central tube and the hollow cone spray of oxidizer from the outer annulus intersected at a position at which one-half of the cross-sectional area of the chamber was outside the impingement circle and one-half inside it. This position resulted in the optimum experimental performance as determined by preliminary firing tests. All test data reported with this injector apply to the condition at which the tube extends 0.235 in. For testing purposes, this injector was mated to an uncooled cylindrical stainless-steel chamber of 1.5-in. outside diameter and 0.065-in. wall thickness. The over-all length of the chamber was 12 in. A turbulence ring having a ¾-in.-diameter orifice was placed in the chamber approximately

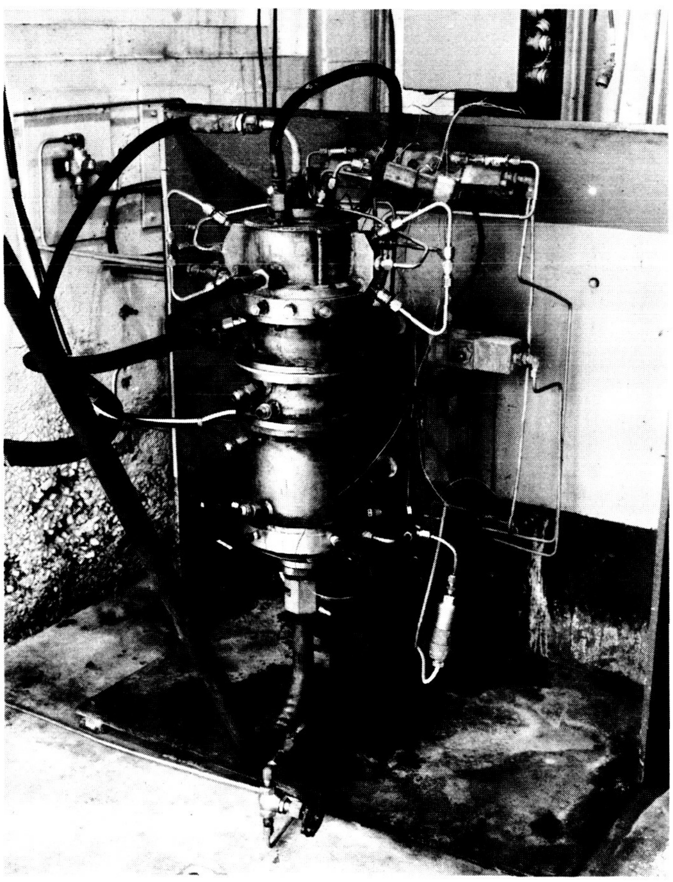

Fig. 2. Experimental test setup of high-mixture-ratio survey gas generator, $L^* = 3900$ in.

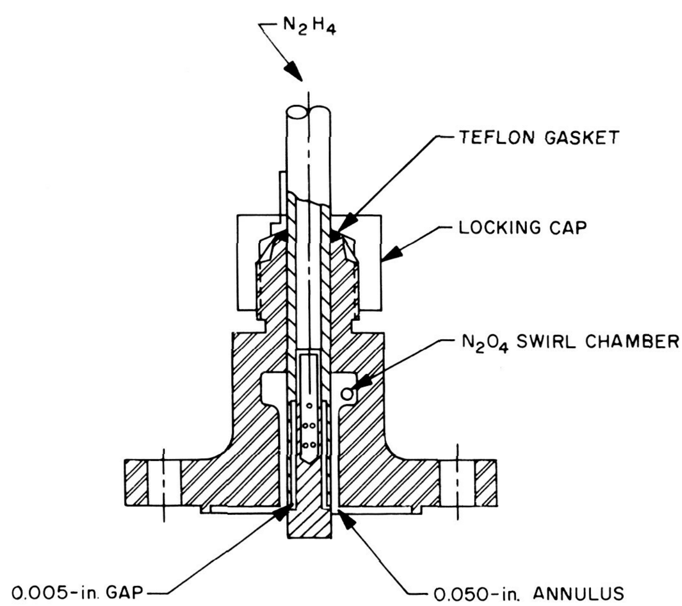

Fig. 3. Single-element concentric-spray gas-generator injector

4¾-in. below the injector face to obtain a near-uniform temperature profile. The over-all L^* of the unit was 100 in., with 62 in. of this total below the turbulence ring.

The test setup of this gas generator is shown in Fig. 4. Instrumentation on the combustor included three commercially obtained sheathed thermocouples and probes with exposed junctions. The thermocouples were located 1⅝ in. from the aft end of the generator and were inserted to different depths into the gas stream. The mean temperature reported is the arithmetic mean of the three measurements obtained. Two chamber-pressure transducers were utilized, one near the head of the generator and one near the aft end at the same station as the thermocouples. Chamber-pressure measurements reported are those obtained from the transducers mounted at the aft end of the chamber. Orifice plates and differential-pressure transducers were utilized for liquid-flow measurements. Tests were run at chamber pressures of 400 and 500 psia. Test durations of 5, 10, and 15 sec were used in this series, with data being reduced from operation during the last second of operation.

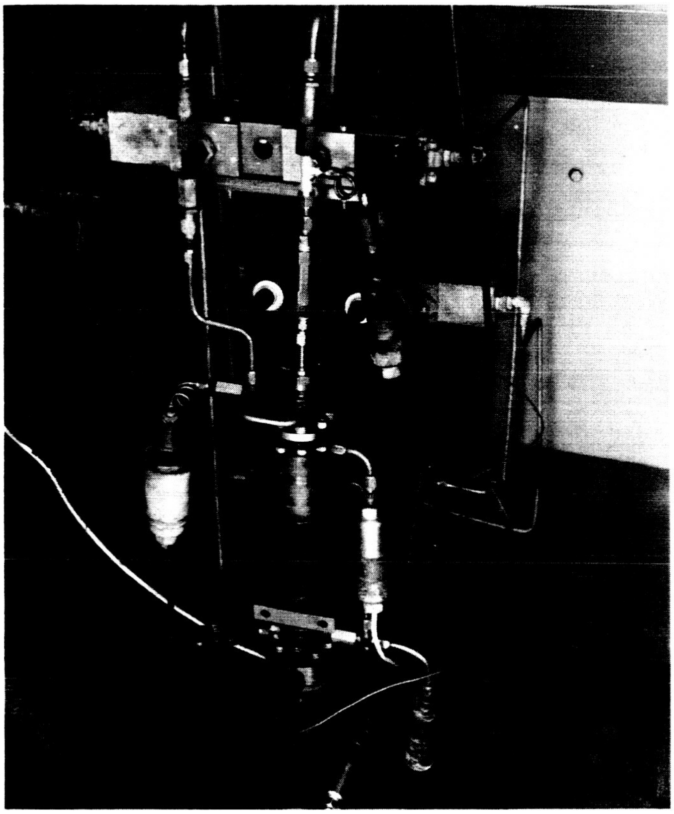

Fig. 4. Experimental test setup of high-mixture-ratio gas generator, $L^* = 100$ in.

2. Results

A total of 21 tests were made with the two oxidizer-rich bipropellant gas generators in which data satisfactory for reduction and presentation were obtained. Utilizing the initially described combustor of 3900-in. characteristic chamber length, a series of 13 tests were made over a mixture-ratio range of 5.9 to 13.0. The data from these tests are shown in Table 1. Data from the eight tests using the unit of 100-in. characteristic length, covering a mixture-ratio range of 7.49 to 10.16, are reported in Table 2.

Table 1. High-mixture-ratio bipropellant N_2O_4–N_2H_4 test data, $L^* = 3900$ in.[a]

Test No.	$\dot{w}_t$ lb/sec	$\dot{w}_o/\dot{w}_f$	p_c psia	c^* ft/sec	T_1 °F	T_2 °F	T_3 °F	T_4 °F	T_5 °F	T_{mean} °F	Test duration sec
322	0.73	11.48	565	1950	620	700	578	597	675	634	12
323	0.59	7.94	563	2410	1071	1079	1011	998	1033	1038	12
324	0.47	5.93	561	3030	1673	1728	1543	1553	1667	1634	12
325	0.70	12.90	505	1810	527	524	493	502	466	502	13
326	0.68	12.97	496	1822	563	559	528	528	497	535	13
327	0.66	12.28	498	1895	628	624	606	628	563	610	13
328	0.64	11.34	502	1982	716	710	692	710	632	692	13
329	0.60	10.19	503	2113	802	791	782	795	710	776	13
330	0.58	9.59	498	2170	877	863	856	863	778	847	13
331	0.54	8.59	495	2310	989	981	964	977	888	960	12
332	0.50	7.59	498	2515	1171	1158	1120	1137	1053	1128	14
333	0.46	6.69	500	2755	1372	——	1321	1346	1261	1325	13
334	0.42	5.91	495	2968	1609	——	1560	1591	1499	1565	12

[a]Nozzle-throat area for all tests = 0.078 in.^{2}

Table 2. High-mixture-ratio bipropellant N_2O_4–N_2H_4 test data, $L^* = 100$ in.[a]

Test No.	$\dot{w}_t$ lb/sec	$\dot{w}_o/\dot{w}_f$	p_c psia	c^* ft/sec	T_1 °F	T_2 °F	T_3 °F	T_{mean} °F	Test duration sec
360	0.82	7.50	385	2500	—	—	—	—	5
361	0.88	7.92	395	2405	1381	1036	1137	1185	5
362	1.03	9.92	407	2113	842	701	854	799	5
363	1.04	10.19	408	2095	820	697	825	781	15
364	1.04	10.04	413	2117	842	701	842	795	15
365	1.25	9.91	499	2137	859	735	863	813	10
366	1.25	9.84	501	2144	871	744	884	833	10
367	1.25	9.83	501	2144	877	752	871	833	10

[a]Nozzle-throat area for all tests = 0.166 in.^{2}

In order to observe the relationship of the experimental data to the data predicted on the basis of thermodynamic equilibrium, a series of thermochemical performance calculations of the nitrogen tetroxide–hydrazine system were performed. These calculations were made using an IBM 7090 computer program developed by and obtained from the Aeronutronic Division of the Ford Motor Company; the program consists of an extension of the Rand method for determining equilibrium compositions by free-energy minimization. This method is described by B. R. Kubert and S. E. Stephanou in Ref. 14. Cases were run over the mixture-ratio range of 5.0 through 15.0, for chamber pressures of 300 and 500 psia exhausting to 14.7 psia. The reaction-chamber temperatures obtained from these calculations were significantly higher than the experimentally observed chamber temperatures through the entire mixture-ratio range investigated, apparently indicating, as anticipated, a non-equilibrium condition. In addition, while the program indicated essentially only H_2O, N_2, and O_2 in the exhaust gas, with only trace quantities of NO or NO_2, in fact, the observed exhaust products from the gas-generator tests were virtually opaque and varied in color from brownish-red through brown, characteristic of the nitrogen oxides. Figure 5 indicates the magnitude of the difference existing between the theoretical equilibrium temperatures and the observed chamber temperatures. Since the dissociation of both nitrogen dioxide and nitric oxide is exothermic, it is evident that the lack of decomposition of the oxides would produce gas temperatures below those predicted for equilibrium conditions. It should be noted that the test data points for both the $L^* = 3900$-in. and $L^* = 100$-in. gas generators fall essentially along a common line, which would appear to indicate that even with gross excursion in residence time the approach to equilibrium conditions is not enhanced. The variation of characteristic exhaust velocity c^* with mixture ratio for both series of tests and its relation to the theoretical maximum value are presented in Fig. 6.

B. Fuel-Rich Region (Low Mixture Ratio)

1. Experimental Effort

The combustion device used in these tests was assembled from an existing uncooled heavy-walled gas generator originally designed and used in the monopropellant circuit of the *Corporal* advanced-development gas-

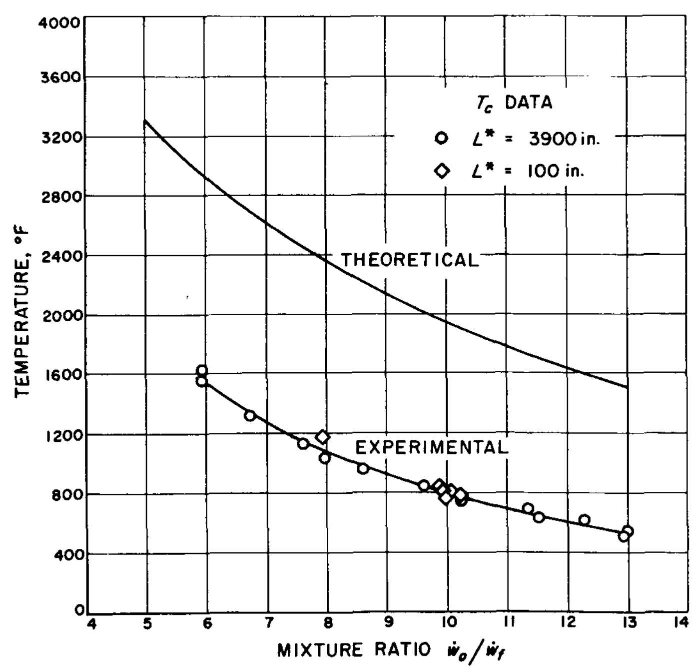

Fig. 5. Variation of gas temperature with mixture ratio for the N_2O_4–N_2H_4 bipropellant system in the high-mixture-ratio region

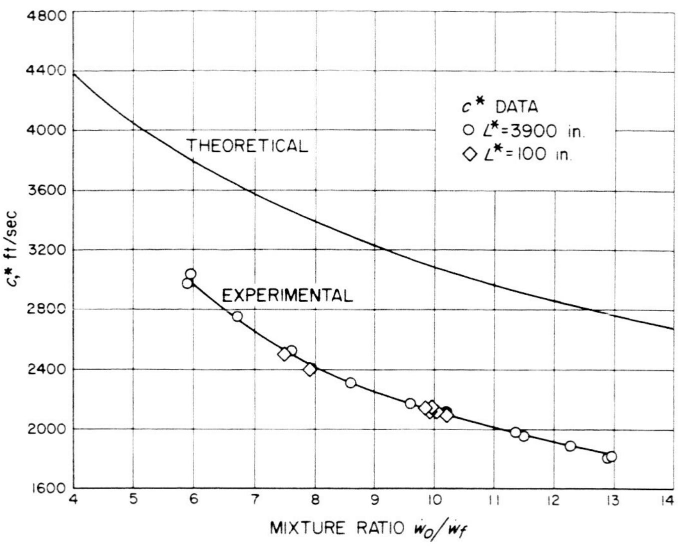

Fig. 6. Variation of characteristic exhaust velocity c* with mixture ratio for the N_2O_4–N_2H_4 bipropellant system in the high-mixture-ratio region

generation-system program mentioned in A-1. The injector, shown in Fig. 7, consisted of a 90-deg conical head having a single oxidizer-spray nozzle mounted axially at the apex of the cone and four fuel-spray nozzles mounted on the cone wall. Conventional oil-burner atomizing nozzles were used.* The combustion chamber was fabricated from stainless steel with an inside diameter of 3.5 in. and a wall thickness of 0.5 in. The nozzle-throat area was sized to give a chamber L^* of 250 in. The results of previous work done at the Jet Propulsion Laboratory had shown this L^* value to be a lower limit at which stable thermal decomposition of monopropellant hydrazine could be sustained reliably (Refs. 15 and 16). The experimental rocket-motor test setup is shown in Fig. 8.

Considerable effort was expended to obtain reliable combustion gas temperature data in the range of 2000 to 3000°F; to this end, a water-cooled, shielded, aspirating-thermocouple probe was designed and fabricated. The design objectives for the thermocouple probe were (1) to surround the measuring junction with a radiation shield, and (2) to decrease the flow velocity past the junction. The probe consisted of an 0.062-in.-diameter, platinum-sheathed thermocouple tube, with 0.010-in.-diameter platinum and 90% platinum–10% rhodium wires embedded in swaged magnesium-oxide insulation. The thermocouple tube was concentrically covered by an 0.125-in.-diameter × 0.010-in. wall-thickness tantalum aspirating tube. This tube also acted as a radiation shield.

*Manufactured by the Delavan Mfg. Co., West Des Moines, Iowa.

The two tubes were held concentric by a sleeve spacer, which was silver-soldered in place. To prevent the silver solder from melting at the high temperatures, a water-cooling jacket fabricated from ⅜-in. tubing was soldered over the probe end. Although tantalum has very poor oxidation resistance, satisfactory operation was obtained in the highly reducing atmosphere of the low-mixture-ratio tests. A sketch of the thermocouple probe is shown in Fig. 9.

Two series of tests were conducted over the mixture-ratio range of 0.0 to 0.55. Typically, the tests were performed at a nominal chamber pressure of 300 ±15 psia, and were of 6-sec duration. The first series of tests was made without thermocouples, the second with the thermocouple probes described in the preceding paragraph. Five such probes were mounted at 72-deg increments in the aft portion of the combustion chamber immediately upstream of the converging section of the nozzle. Each point in the reduced data represents an arithmetic mean of the five thermocouple measurements made at various depths in a cross-sectional plane of the test motor. Two probes were mounted approximately on the motor centerline, two at a depth of 0.9 in. (one-quarter of the motor diameter), and one at a depth of ½ in. The greatest single

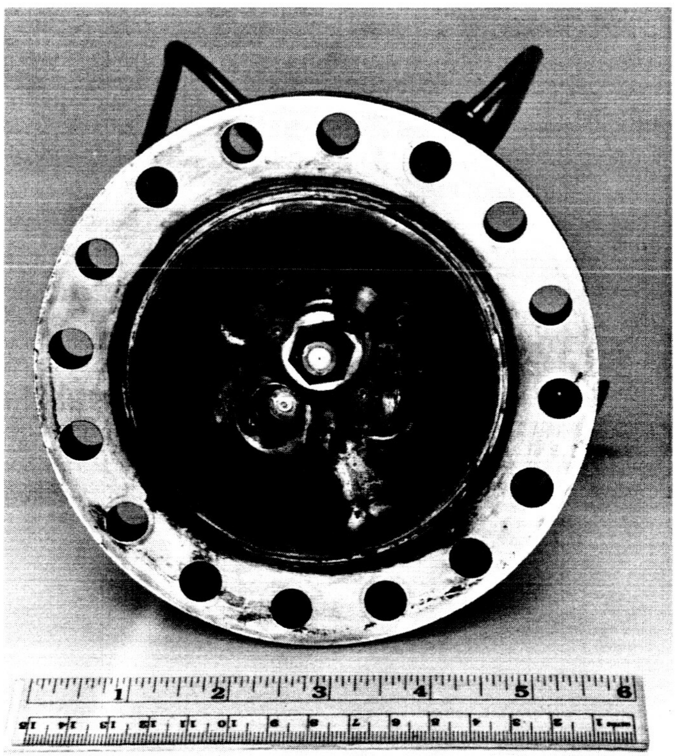

Fig. 7. Injector used in low-mixture-ratio survey test program

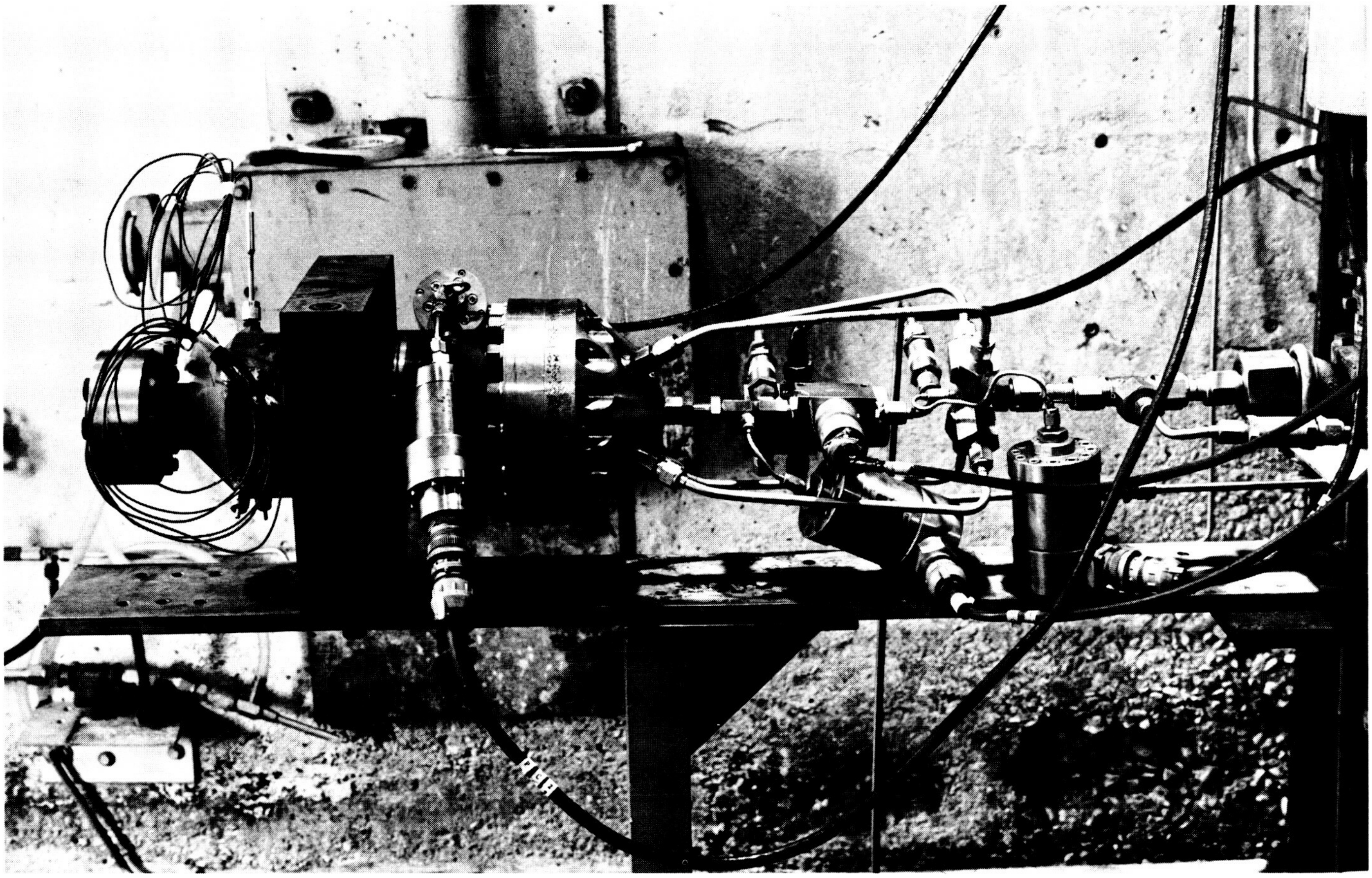

Fig. 8. Experimental test setup of low-mixture-ratio survey combustor

source of error introduced in obtaining a good average temperature was found to be the nonuniform mixing of the combustion gases. The injector design caused hot spots or hot streaks through the chamber. This streaking became more noticeable as the mixture ratio was increased (more flow was introduced through the single oxidizer spray jet). Since the thermocouples are point-source probes, the temperature measured is particularly dependent on the thermocouple location in relation to the injector hot streaks. Thus, it is assumed that, by taking an arithmetic mean of the five probes, a reasonably accurate value of the average gas temperature was obtained.

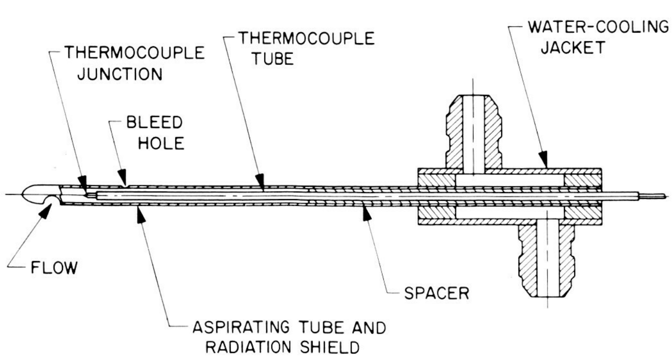

Fig. 9. Water-cooled, shielded, aspirating thermocouple probe

Chamber pressure was recorded from an instrumentation port located on the chamber side wall. Propellant-flow measurements were accomplished through the use of orifice plates and differential-pressure transducers.

2. Results

A total of 60 tests were made with the fuel-rich gas generator. As indicated, two series of tests were conducted; the first was accomplished with a minimum of instrumentation to determine the suitability of the test equipment and the second with the already described thermocouple probes. Data from all of the tests are reported in Table 3. As a result of previous investigations at JPL concerning the monopropellant characteristics of hydrazine, it was recognized that the presence of ammonia in the fuel-rich exhaust products would, at least in the very low-mixture-ratio range, lead to non-equilibrium conditions. As was done in the case of the oxidizer-rich regime, thermochemical performance calculations were carried out in order to ascertain where such calculations might be valid for prediction of combustor operation and where their use would lead to error. For ready comparison, a plot of the combustion temperature as theoretically predicted and the actual data obtained are shown in Fig. 10.

Table 3. Low-mixture-ratio bipropellant N_2O_4–N_2H_4 test data, $L^* = 250$ in.

Test No.		$\dot{w}_f$ lb/sec	$\dot{w}_o/\dot{w}_f$	p_c psia	c^* ft/sec	T_1 °F	T_2 °F	T_3 °F	T_4 °F	T_5 °F	T_{mean} °F	Test duration sec	Remarks
410	A	0.55	0.040	301	4513							6	
	B	0.56	0.043	304	4478							6	
	C	0.56	0.050	305	4485							6	
	D	0.56	0.053	308	4497							6	
	E	0.57	0.052	308	4479							6	
411	A	0.58	0	302	4307							6	
	B	0.58	0	304	4334							6	
	C	0.58	0	304	4325							6	
	D	0.57	0.014	301	4378							7	
	E	0.57	0.014	302	4376							6	
	F	0.57	0.023	301	4409							6	
	G	0.57	0.035	304	4452							6	
	H	0.56	0.050	301	4426							6	
	I	0.56	0.051	302	4475							6	
	K	0.56	0.052	303	4473							6	
412	B	0.51	0.230	295	4800							7	
	C	0.51	0.246	299	4871							7	
	D	0.51	0.252	300	4879							7	
	E	0.51	0.253	301	4890							7	
413	A	0.50	0.239	298	4887							7	
	B	0.51	0.243	301	4898							7	
	C	0.49	0.300	297	4986							7	
	D	0.49	0.305	299	5034							7	
	E	0.50	0.301	298	4958							7	
414	A	0.47	0.293	287	5054							7	
	B	0.48	0.322	297	5047							7	
	C	0.49	0.354	306	5114							7	
	D	0.49	0.395	312	5145							7	
	E	0.48	0.444	315	5241							7	
416	B	0.47	0.322	284	5037							6	
	C	0.47	0.373	290	5124							6	
	D	0.49	0.535	310	5308							6	
	E	0.48	0.552	310	5328							6	
422	A	0.56	0.034	301	4455	2088	2051	2082	2027	1899	2029	16	Both bipropellant and monopropellant data obtained; 6-sec bipropellant operation, 10-sec monopropellant operation
	A	0.55	0	289	4350	2030	1960	2015	1846	1874	1945	16	
	B	0.57	0.033	304	4437	2054	2018	2051	1988	1852	1993	16	
	B	0.56	0	293	4332	2027	1908	2015	1821	1720	1898	16	
	C	0.58	0.030	306	4397	2033	2000	2030	1975	1793	1966	16	
	C	0.57	0	297	4318	2021	1920	2009	1815	1702	1893	16	
423	A	0.54	0.041	293	4488	2012	2012	2024	2030	2277	2071	5	Bipropellant data only
	B	0.55	0.039	295	4450	1988	1981	2000	2021	2148	2026	5	
	C	0.55	0.044	299	4459	2000	1991	1994	2021	2157	2033	5	
424	A	0.52	0.085	286	4577	2139	2166	2139	2091	2024	2112	5	
	B	0.53	0.098	292	4555	2148	2154	2142	2139	2091	2135	5	
	C	0.53	0.109	292	4596	2151	2175	2148	2118	2054	2129	5	
	D	0.51	0.145	287	4649	2244	2280	2205	2193	2112	2207	5	
	E	0.51	0.154	290	4683	2271	2349	2250	2193	2112	2235	5	
	F	0.52	0.154	291	4677	2277	2360	2250	2259	2109	2251	5	
425	A	0.51	0.214	301	4860	—	2399	2417	2357	—	2391	5	
	B	0.52	0.212	302	4830	—	2372	2432	2393	2148	2336	5	
426	A	0.53	0.209	312	4805	2435	2372	2444	3250	2241	2548	5	
	B	0.52	0.234	307	4846	2489	2328	2477	3140	2190	2525	5	
	C	0.49	0.275	295	4915	2658	2337	2673	3110	2202	2596	5	
	D	0.50	0.267	297	4903	2592	2295	2610	3240	2235	2595	5	
427	A	0.52	0.316	315	5019							5	Thermocouple data not valid; tantalum shield eroded
	B	0.50	0.284	303	5000							5	
	C	0.51	0.301	308	4990							5	
	D	0.51	0.303	308	4975							5	
	E	0.49	0.365	305	5087							5	
	F	0.50	0.362	306	5068							5	

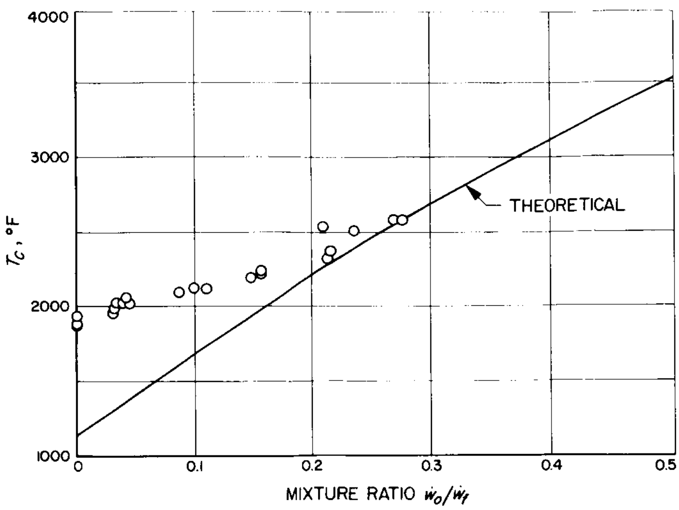

Fig. 10. Variation of gas temperature with mixture ratio for the N_2O_4–N_2H_4 bipropellant system in the low-mixture-ratio region

It is interesting to note from the plot the divergence from chemical equilibrium found at the low mixture ratios. Indeed, it would appear that chemical equilibrium is not approached until a mixture of approximately 0.3 is reached. Thus, as expected, it is quite inaccurate to use equilibrium thermochemical data below this mixture ratio in combustion devices of reasonable residence time and chamber pressure. In order to better characterize this non-equilibrium region, a comparison of a series of theoretical performance curves for arbitrary fractions of ammonia dissociation have been plotted in Fig. 11, along with the experimental data obtained in this investigation. The thermochemical performance data plotted were obtained from Ref. 7 and were calculated by assuming that all of the N_2O_4 was reduced by the N_2H_4 to H_2O and N_2. This reaction requires two moles of N_2H_4 for every mole of N_2O_4. The remaining N_2H_4 was assumed to decompose in accordance with the following equation (for 0% NH_3 decomposition):

$$N_2H_4 \rightarrow NH_3 + \tfrac{1}{2}N_2 + \tfrac{1}{2}H_2$$

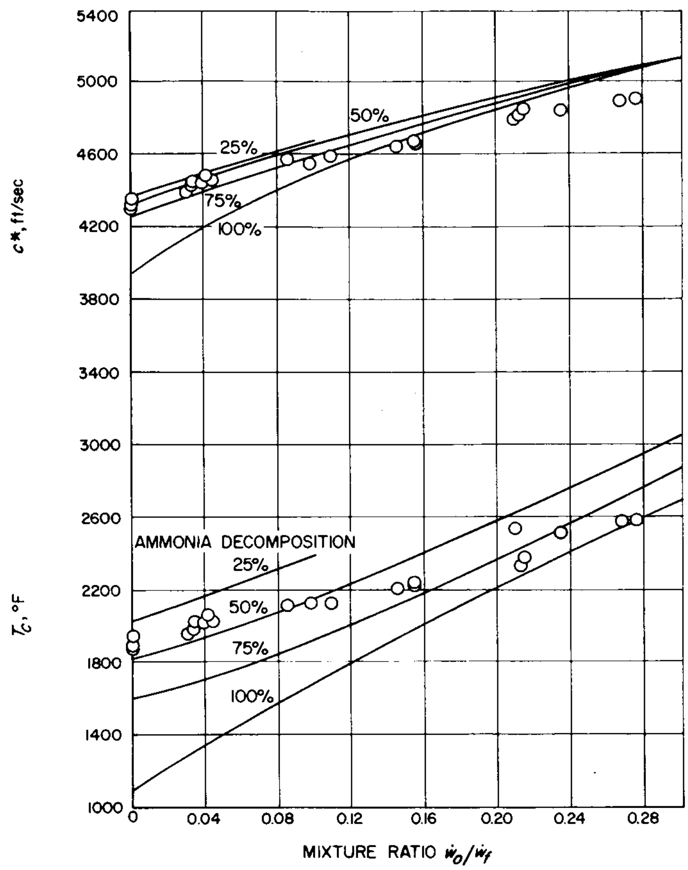

Fig. 11. Relationship of experimental data in the low-mixture-ratio region with calculated data having assumed percentages of ammonia decomposition

From this plot (Fig. 11), an estimate of the actual percentage of ammonia dissociated can be obtained. At a mixture ratio of zero, the experimental data check well with data previously obtained in thermal decomposition hydrazine gas generators (Ref. 11) in which values of 29% ammonia dissociation resulted. The experimentally obtained c^* for both series of tests is shown in Fig. 12.

III. CONCLUSIONS

The combustion of nitrogen tetroxide–hydrazine was found to be smooth and reliable over wide mixture-ratio ranges, in particular from 0 to 0.55 and from 5 to 13.0. The design and operation of suitable combustion devices in these off-mixture-ratio regimes was a straightforward task requiring only the use of conventional rocket-engine design techniques.

The off-mixture-ratio experimental performance obtained with this propellant system pointed up the significance of kinetic effects in low-temperature combustion systems as well as the necessity for using caution in applying thermochemical equilibrium performance calculations. In the oxidizer-rich regime, experimental combustion temperatures obtained were considerably lower than predicted because of the lack of exothermic dissociation on the part of the nitrogen oxides; in the fuel-rich region, the measured temperatures were higher than predicted because of the lack of endothermic dissociation of ammonia.

It would appear that over the mixture-ratio range investigated in the oxidizer-rich regime, the effect of varying L^* over wide limits is of little significance in altering the resultant nature of combustion products. For design purposes, the assumption of a stable gas mixture of combustion-gas temperature and characteristic exhaust velocity versus mixture ratio as determined in this study would seem to be logical. In the fuel-rich region, the use of the experimental data developed is suggested up to a mixture ratio of approximately 0.3; above this mixture ratio, assumption of a percentage of the thermochemical equilibrium value appears to be suitable.

NOMENCLATURE

c^* characteristic velocity, ft/sec

L^* characteristic length, in.

$\dot{w}_o$ weight flow of oxidizer, lb/sec

$\dot{w}_f$ weight flow of fuel, lb/sec

T_c reaction temperature, °F

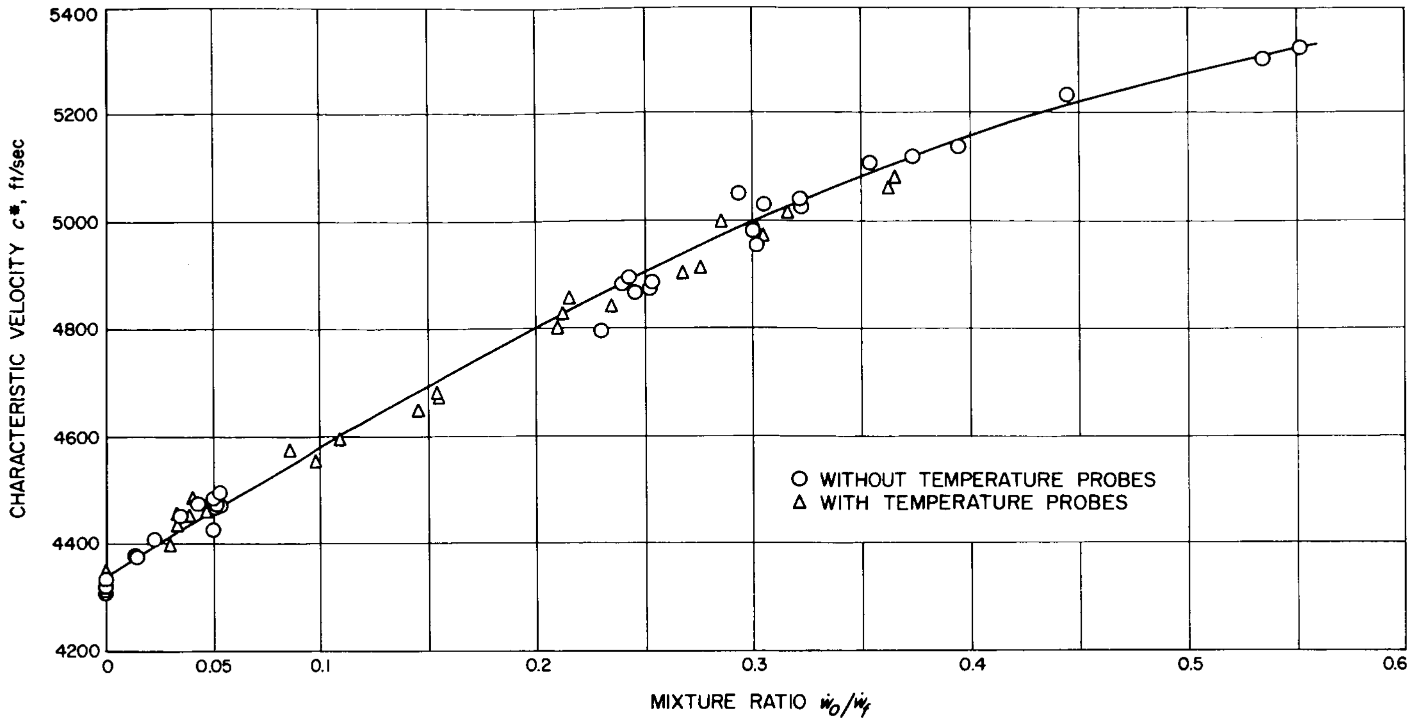

Fig. 12. Variation of characteristic exhaust velocity c* with mixture ratio for the N_2O_4–N_2H_4 bipropellant system in the low-mixture-ratio region

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