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APPLICATION OF HIGH DENSITY NITRIC ACID OXIDIZER AND UDMH WITH SILICONE FLUID ADDITIVE FUEL TO THE AGENA ROCKET ENGINE

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APPLICATION OF HIGH DENSITY NITRIC ACID OXIDIZER AND UDMH WITH SILICONE FLUID ADDITIVE FUEL TO THE AGENA ROCKET ENGINE

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ABSTRACT

Results of an Agena engine fire test program utilizing higher performance propellants substituted for the standard IRFNA/UDMH combination are presented. A six lb-sec/lb gain in specific impulse and a three percent improvement in bulk density was obtained with the current engine utilizing the higher performance propellants.

Use of 1% silicone fluid fuel additive, which produced a *self replenishing silica film on the combustion gas side wall* caused a 33% heat flux reduction.

Previous experiments with silicone fluid propellant additive were conducted under Project Hermes at General Electric Company, Malta, New York and under Bell Aerospace Company independent research program in 1961. The work reported herein is the first application of HDA/UDMH plus silicone to gas generators and a complete rocket engine.

The program, culminating in flight worthiness demonstration, accumulated over 2400 seconds of operation involving 11 engine, 11 injector and 4 gas generator level tests. *Propellant physical and chemical properties were determined.*

INTRODUCTION

Performance improvement studies showed that the payload of the Agena Ascent Stage could be significantly increased without changing either the engine or the tankage. This was accomplished by propellant modifications which provide higher bulk density and higher specific impulse at the same volumetric mixture ratio as the standard engine.

This paper documents the engine test program conducted by Bell Aerospace Company under Lockheed Missiles and Space Company Subcontract to U.S. Air Force Contract FO4701-68-C-0235, and was performed during the period 1 February to 1 August 1970.

The objectives of this program were:

1. Investigate the feasibility of substitution of high density acid (HDA) in the current engine.
2. Evaluate thrust chamber heat transfer characteristics and gas generator operation.
3. Demonstrate satisfactory engine operation, performance gain and durability with HDA/UDMH propellants.
4. Establish propellant physical and chemical properties.

ENGINE DESCRIPTION

The Agena Rocket Engine shown in Figure 1 consists of turbine pump and thrust chamber assemblies installed in a gimballed mount.

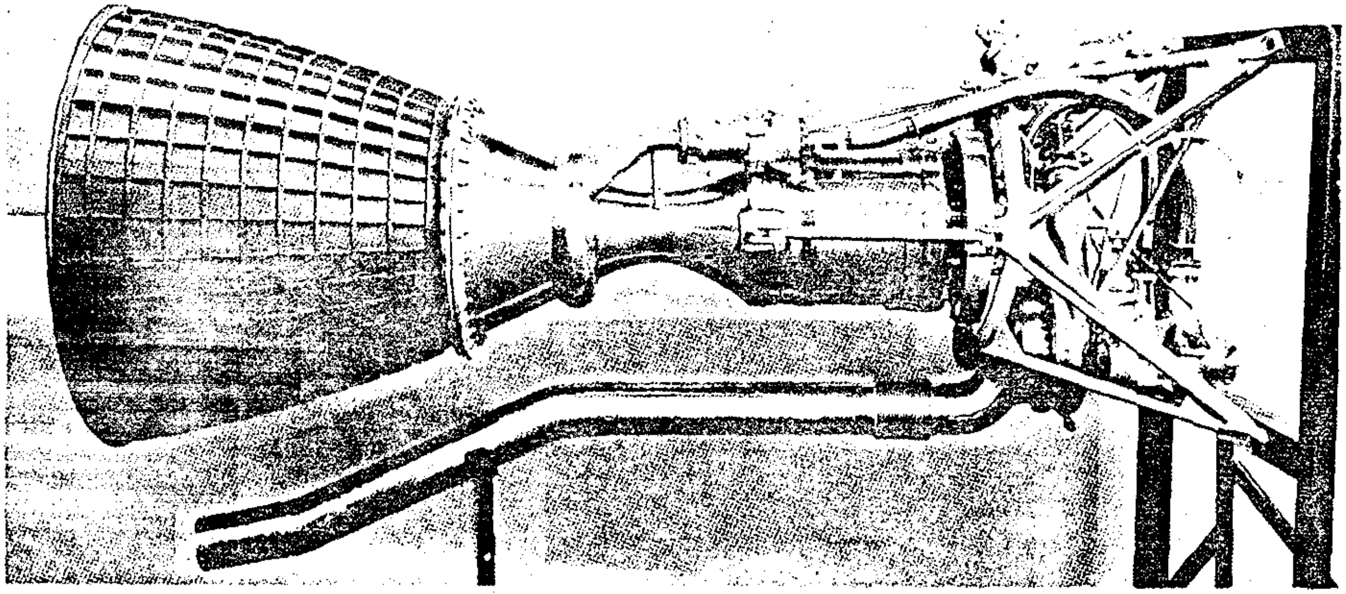

Figure 1. Agena Rocket Engine

**Project Engineer, Rockets and Propulsion Department*

The turbopump configuration features solid propellant starters, fuel rich gas generator driven single stage impulse turbine, and tandem geared propellant pumps.

The thrust chamber design features drilled aluminum wall fabrication with a radiation cooled nozzle extension.

Gas generator and thrust chamber valves are propellant pressure actuated.

This engine has been used in more than 300 flights with an observed reliability of greater than [illegible].

The engine is started by electrically energizing the solid propellant pyrotechnic squibs as shown in the engine schematic, Figure 2. Combustion gas from the solid propellant grains is directed through the gas generator injector to the rotor via the turbine nozzles. The turbine rotor drives the fuel and oxidizer propellant pumps through a gear train.

Gas generator operation is sustained by the pumped propellants. Fuel and oxidizer flow is controlled by cavitating venturi and a bipropellant valve actuated when the fuel pump discharge pressure reaches 250 psi. The turbine continues to accelerate, controlled by cavitating venturi, until power balance is attained at the design operating speed.

Thrust chamber propellant flow is pressure sequenced to provide oxidizer lead at start. As the oxidizer pump discharge pressure increases to 130 psi, it opens the thrust chamber oxidizer valve spring loaded poppet, allowing flow through the thrust chamber coolant passages and injector. When the pressure in the oxidizer injector manifold reaches 33 psi a pressure switch is actuated which in turn signals actuation of the thrust chamber fuel valve. The start transient, from command to 90% thrust, is completed in 1.1 seconds.

Shutdown of the engine is accomplished by removal of electrical power which causes the thrust chamber fuel valve and gas generator bipropellant valve to close. The thrust chamber oxidizer valve closes by spring action as the pump discharge pressure decays.

PROPELLANT SUBSTITUTION CONSIDERATIONS

Engine performance characteristics are presented in Table I for both the standard IRFNA/UDMH propellants and with HDA/UDMH plus 1% Si substituted at constant volumetric flowrate conditions. It can be seen that the higher density and higher performance associated with the use of HDA are reflected in increased propellant flowrate, mixture ratio, combustion pressure and thrust conditions. Additionally, the oxidizer pump inlet pressure requirement is increased due to the higher vapor pressure of the HDA.

Propellant characterization effort was conducted in support of the engine evaluation program.

TABLE I. ENGINE PERFORMANCE CHARACTERISTICS

	IRFNA UDMH	HDA UDMH + 1% Si
Engine		
Thrust, lbs, vacuum	16,000	16,925
Propellant flowrate, lb/sec	55	57
Mixture ratio	2.57	2.70
Fuel inlet pressure, psia (min.)	13.5	13.5
Oxidizer inlet pressure, psia (min.)	10.5	17.3
Thrust Chamber		
Thrust, lbs, vacuum	15,800	16,715
Propellant flowrate, lb/sec	53.4	55.4
Mixture ratio	2.80	2.94
Chamber pressure, psia	500	525
Turbopump		
Gas generator, flowrate, lb/sec	1.60	1.61
Gas generator mixture ratio	0.15	0.157
Turbine manifold pressure, psia	475	480
Turbine inlet gas temperature °F	1,450	1,450

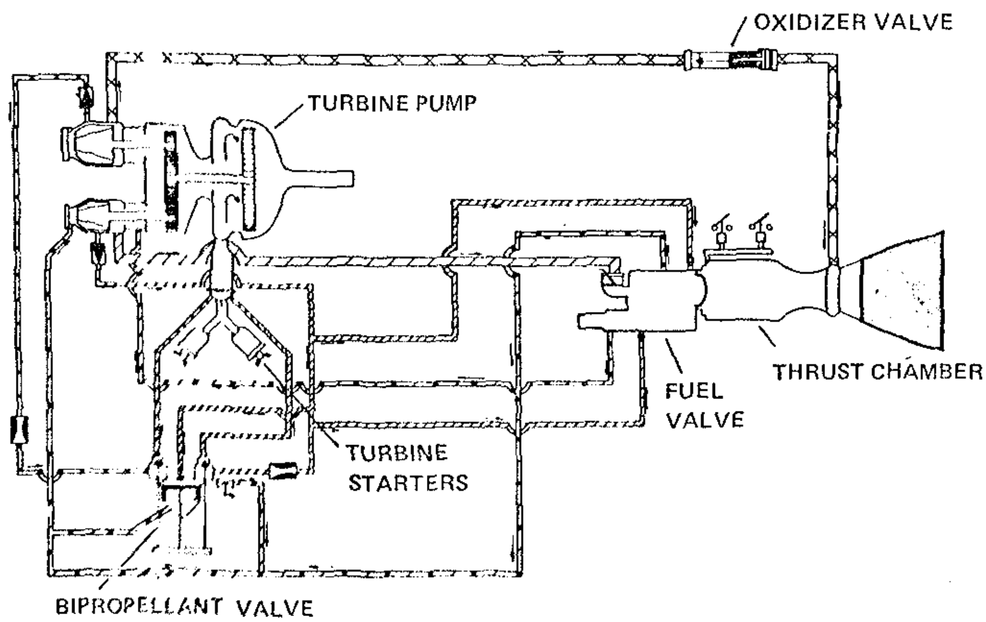

Figure 2. Engine Schematic

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(71-736) APPLICATION OF HIGH DENSITY NITRIC ACID OXIDIZER AND UDMH WITH SILICONE FLUID ADDITIVE FUEL TO THE AGENA ROCKET ENGINE

Page 4:Figure 5 shown below was omitted.

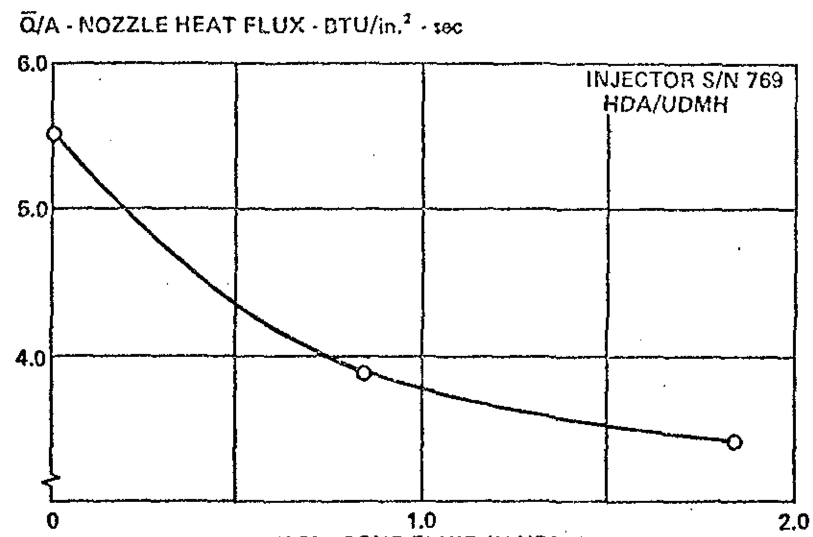


Figure 5. Effect of Silicone Additive Concentration on Nozzle Heat Flux Rate

Propellant physical and chemical properties are presented in Table II. Density, vapor pressure and formulation data for oxidizers and fuels are shown.

TABLE II. PROPELLANT CHARACTERISTICS

	PHYSICAL PROPERTIES			
	IRFNA	HDA	UDMH	UDMH + 1% Si
Specific Gravity 60/60	1.57	1.646	0.795	0.797
Vapor Pressure psia at 60°F	2.1	8.4	2.0	2.1

COMPOSITION

	WEIGHT %				
	OXIDIZERS		FUELS		
	HDA	IRFNA	UDMH		UDMH + Si
HNO_3	Bal.	Bal.	UDMH	98 min.	97 min.
NO_2	44-46	13-15	H_2O	0.3 max.	0.3 max.
H_2O	0.5 max.	1.5-2.5	*$(SiC, H_4O)_n$	-	0.9-1.1
HF	0.6-0.8	0.6-0.8			

*General Electric Company SF96 Silicone Fluid

Theoretical specific impulse comparisons are presented in Figure 3. It can be seen that HDA provides higher performance with UDMH than the standard IRFNA oxidizer. A potential gain of 6 lb-sec/lb is shown at constant volumetric flowrate mixture ratio conditions. Thus rated thrust chamber mixture ratio of 2.80 with IRFNA/UDMH corresponds to 2.94 with HDA/UDMH by substitution of HDA for IRFNA oxidizer without hardware change.

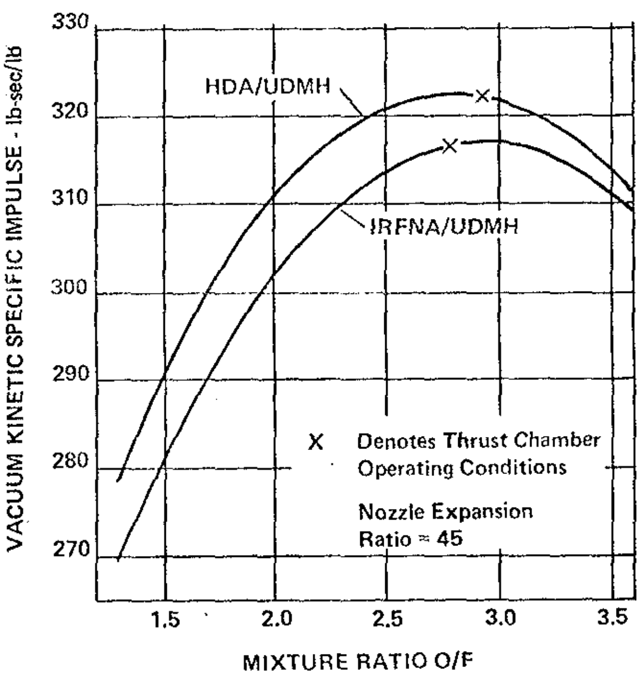


Figure 3. Comparison of Theoretical Specific Impulse Performance

Theoretical gas generator combustion gas properties for the propellant combinations utilized are presented in Table III. It is shown that the combinations IRFNA/UDMH and HDA/UDMH are nearly identical while HDA/UDMH + 15% JP4 yields lower temperature and energy content combustion gas. The latter propellant combination was considered during early feasibility effort.

TABLE III. THEORETICAL GAS PROPERTIES FOR GAS GENERATOR, O/F MIXTURE RATIO = 0.15

	IRFNA/ UDMH	HDA/ UDMH	HDA/UDMH + 15% JP4
Combustion Temperature °R	2102	2128	2060
Gas Molecular Weight lb/Mol	14.0	13.9	14.0
Characteristic Velocity - ft/sec	4100	4131	3946

Calculations were conducted to assess the margin of the regeneratively cooled thrust chamber. A comparison of thrust chamber thermal margin presented in Figure 4 shows that little margin exists with HDA/UDMH. This is due to increased combustion side heating and a reduction in coolant capability of the HDA. The margin with HDA/UDMH + 1% Si is greatly improved due to the thermal effectiveness of the silica film deposited on the combustion gas side. Thermal transport properties of HDA are not well defined and ultimate heat flux value shown was extrapolated from IRFNA heated tube data, Reference 1.

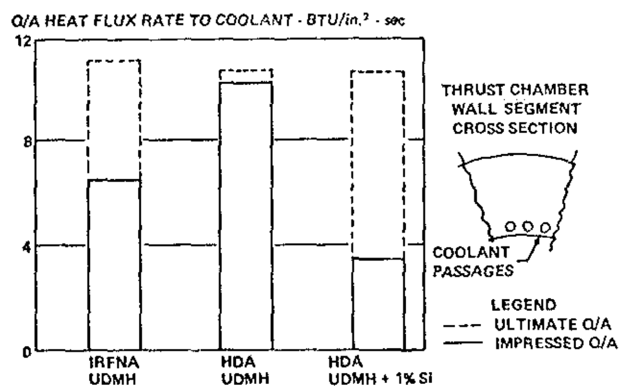


Figure 4. Comparison of Thrust Chamber Thermal Margin

TEST PROGRAM

Fire tests conducted during the HDA program may be grouped, chronologically, into three phases (1) feasibility, (2) definition and (3) flight worthiness.

Feasibility tests were conducted initially, utilizing HDA/UDMH + 15% JP4 and HDA/UDMH propellants, at the engine assembly level.

Definition test included gas generator, thrust chamber injector and engine assembly level fire tests. These tests encompassed a wide range of operating conditions and provided the data base for defining the subsequent flight worthiness program.

Flight worthiness tests were conducted on a refurbished engine assembly which completed the HDA ground test effort.

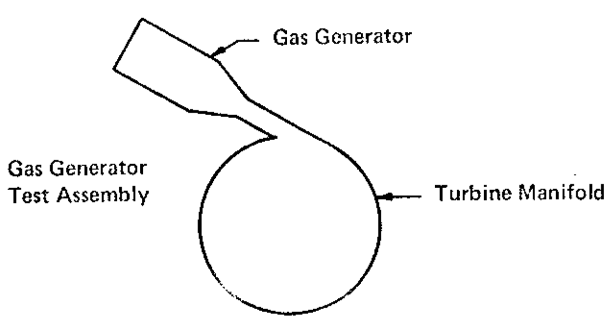
Discussion of gas generator thrust chamber injector and engine test program is included in the following sections.

GAS GENERATOR TEST

Four 30-second duration tests were conducted to evaluate combustion operating with HDA/UDMH + 15% JP4 and HDA/UDMH + 1.8% Si. A gas generator test summary is presented in Table IV, on which is shown the gas generator-manifold test assembly.

TABLE IV. GAS GENERATOR TEST SUMMARY S/N 675

Test No. (Z1)	Oxidizer/Fuel	Wt lb/sec	O/F	P_m psia	C* ft/sec	Gas Temp °F
166	HDA/UDMH + 15% JP4	1.57	0.257	431	3367	1508
167A	HDA/UDMH + 1.8% Si	1.53	0.150	420	3370	1457
167B	HDA/UDMH + 1.8% Si	1.62	0.155	450	3413	1471
167C	HDA/UDMH + 1.8% Si	1.53	0.176	433	3482	1507
Typical	IRFNA/UDMH	1.55	0.150	465	3363	1450



The test assembly was removed from the feasibility test engine S/N TO-1.

Operating conditions for the initial test, with HDA/UDMH + JP4, were planned to provide the maximum mixture ratio (0.26 O/F) available in the existing turbopump, based on pump discharge pressure capability. Previous engine tests with these propellants showed that the energy content of the gas generator was too low at 0.15 mixture ratio, O/F, to sustain turbine speed. Test results indicated that mixture ratio of about 0.25 would provide satisfactory combustion gas energy content and temperature.

A test series was then conducted with HDA/UDMH + 1.8% Si. Test conditions were planned to include variations in mixture ratio and total propellant flow rate. Test results showed that combustion gas properties (energy content and temperature) were slightly higher than for typical IRFNA/UDMH conditions and operation was within allowable material margins.

THRUST CHAMBER INJECTOR TEST

A summary of injector test results are shown in Table V which includes a sketch of the test assembly. Eleven test firings, each of 5 seconds duration were conducted. Nine of the tests were performed with injector S/N 769 which was removed from the feasibility test engine S/N TO-1. Two acceptance tests were made with injector S/N 811 which was assembled in the definition test engine thrust chamber. The principal objective of these tests was to establish injector heat transfer characteristics over a range of operating conditions by determining the average heat flux rate to the water cooled nozzle.

An initial baseline firing was performed with IRFNA/UDMH which yielded nozzle average heat flux, Q/A, of 5.15 BTU/in^2-sec.

Then a series of HDA/UDMH test firings were conducted over a range of mixture ratio conditions. Results from these tests showed that nozzle Q/A increased about 7% with HDA oxidizer operating at mixture ratio, O/F, corresponding to constant volumetric flowrate conditions, as indicated comparing test No. 123 and 124. Further, the Q/A appears to be rather insensitive to mixture ratio, comparing test Nos. 124, 125, 126 and 127.

Operation with HDA/UDMH + 15% JP4 showed a reduction in Q/A to a level about 10% lower than that obtained with HDA/UDMH.

The most significant test results were obtained from test Nos. 131 and 133 with HDA/UDMH + Si. The effect of silicone fluid additive concentration on nozzle heat flux is presented in Figure 5, which shows the insulating characteristic of the silica film formed on the combustion gas side surfaces. Post-test inspection of the injector test assembly showed no evidence of silica deposits. It was postulated that the oxidizer post flow at shutdown removed the silica deposit.

The silica film was observed on the thrust chamber combustion side surfaces during a later engine fire test.

ENGINE TEST

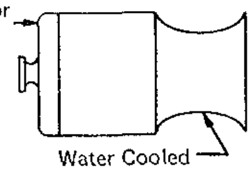
Eleven sea level engine fire tests were conducted with HDA oxidizer in performing feasibility, definition and flight worthiness phases, utilizing three engine assemblies as presented in Table VI. All of the tests were performed in a facility equipped with a parallel tank arrangement to provide in-run propellant sequencing and extended duration (640-second) test capabilities. The thrust chamber nozzle extension was not included in the test engine assembly.

The feasibility test series conducted with Engine TO-1 was performed prior to the component level testing. The initial test was planned to include an in-run propellant switch to HDA/UDMH + 15% JP4. However, a ground safety malfunction shutdown occurred during the propellant sequencing. Operation during sequencing was performed such that the fuel switch was accomplished 1 to 2 seconds prior to the oxidizer switch. Shutdown occurred during IRFNA/UDMH + 15% JP4 operation due to low gas generator performance. The low performance indicated that higher mixture ratio conditions would be required to obtain satisfactory operation.

TABLE V. INJECTOR TEST SUMMARY

Inj. S/N	Test No.	Oxidizer/Fuel	Wt lb/sec	O/F	P_c psia	C* ft/sec	$\dot{Q}$/A Nozzle BTU/in.^{2} - sec	
769	123	IRFNA/UDMH	53.2	2.83	506	5175	5.20	(5.15) (1)
769	124	HDA/UDMH	53.5	2.96	512	5206	5.60	(5.50)
769	125	HDA/UDMH	54.3	2.77	523	5248	5.64	(5.44)
769	126	HDA/UDMH	56.6	2.09	553	5318	6.12	(5.65)
769	127	HDA/UDMH	47.42	2.81	460	5284	5.03	(5.38)
769	128	HDA/UDMH + 15% JP4	53.5	2.98	511	5199	5.02	(4.97)
811	129	HDA/UDMH	53.3	2.88	515	5262	5.61	(5.48)
811	130	HDA/UDMH	53.6	2.99	512	5200	5.59	(5.49)
769	131	HDA/UDMH + 1.8% Si	53.8	2.98	515	5214	3.50	(3.42)
769	132	FNA(27% NO_2)/ UDMH	54.4	2.89	517	5173	5.34	(5.20)
769	133	HDA/UDMH + 0.8% Si	53.8	2.92	517	5234	3.96	(3.86)

(1) Values in parentheses corrected to 500 P_C.



Injector Test Assembly

TABLE VI. ENGINE TEST SUMMARY

Engine S/N	Test H3	Planned Duration sec	Oxidizer/Fuel	Prop Temp °F	Remarks
TO-1	570	30 30	IRFNA/UDMH HDA/UDMH + 15% JP4	40	Shutdown during sequence at 34 seconds due to low gas generator performance.
TO-1	571	30 30 30	IRFNA/UDMH HDA/UDMH IRFNA/UDMH	10 10 10	
TO-1	572	240	HDA/UDMH	50	Throat burnout at ~120 seconds, run duration completed.
376X	573	30 30 30	IRFNA/UDMH HDA/UDMH + Si IRFNA/UDMH	60 60 60	Shutdown due to test cell component failure at 68 seconds.
376X	574	240 30	HDA/UDMH + Si IRFNA/UDMH	60 60	
376X	575	120 60 160	IRFNA/UDMH HDA/UDMH + Si HDA/UDMH + Si	40 40 40	Pump inlet pressure decay test.
376X	576	60 60	HDA/UDMH + Si HDA/UDMH + Si	30 90	
376X	577	640	HDA/UDMH + Si	60	
661	578	240	IRFNA/UDMH	60	Engine checkout
661	579	240 40	HDA/UDMH + Si IRFNA/UDMH	60 60	
661	580	240 40	HDA/UDMH + Si IRFNA/UDMH	32 32	Shutdown during sequence at 240 seconds.
661	581	240 120	HDA/UDMH + Si IRFNA/UDMH	90 90	

The next test firing included an in-run switch to HDA/UDMH with propellants conditioned to 10°F to provide increased thermal margin, and engine operation was satisfactory.

A 240-second duration test was then performed with HDA/UDMH at propellant temperatures of 50°F. Thrust chamber coolant passage burnout occurred at about 120 seconds although the planned duration of 240 seconds was completed. This result indicated marginal thrust chamber operation with HDA/UDMH and emphasis was directed toward reducing the heat flux with silicone fluid fuel additive.

The definition engine test series was conducted with engine 376X and was performed subsequent to gas generator and thrust chamber injector level tests. All of the testing utilized HDA/UDMH + 1% Si propellants.

Test H3-573 was conducted to establish a performance base line and checkout the refurbished engine assembly. Tests 573, 574 and 575 were utilized to establish the characteristic velocity c* performance gain from HDA/UDMH + Si with in-run propellant sequencing to minimize measurement dispersion effects. The average change in c* for these tests indicated a gain of 80 ft/sec (1.6%) as shown in Figure 6. This c* performance gain was higher than the 45 ft/sec predicted from theoretical comparisons, which indicates somewhat higher combustion efficiency with HDA/UDMH + Si. Specific impulse measurement was precluded, since the nozzle extension was not included in the sea level test engine. Based on post run observation of the silica deposits observed on the nozzle combustion gas side surfaces some reduction in nozzle throat area occurs during operation which results in slight (estimated to be in the order of 0.1 to 0.2%) error in c* derived from measured throat dimensions.

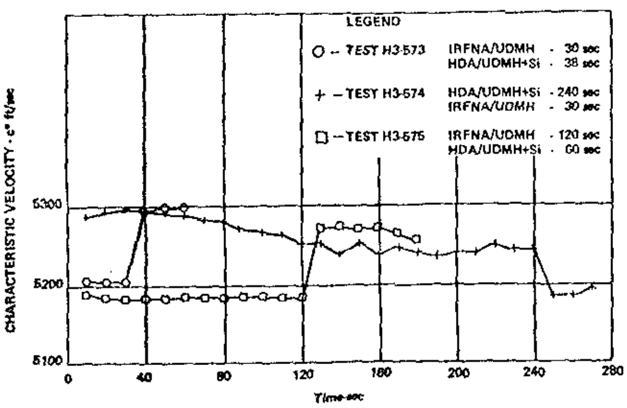


Figure 6. Characteristic Velocity Time History

The thermal effect of the silicone fuel additive during regenerative operation is presented in Figure 7 which shows the heat rejection time history for Test 571 operated on HDA/UDMH and Test 574 with HDA/UDMH + 1% Si. Thermocouples were installed at the inlet and at manifolds located at the chamber-to-nozzle joint and the injector-to-chamber joint, which allowed measurement of the coolant temperature rise from which the heat rejection to the nozzle and combustor portions was derived. Comparing the derived data it is shown that the silicone additive reduced the heat rejection rate by 25% in the nozzle portion and 35% in the

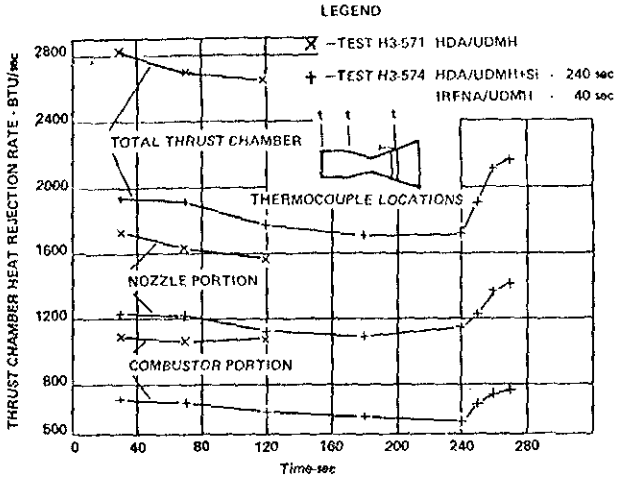


Figure 7. Heat Rejection Time History

combustor portion. This suggests that a thicker SiO_2 film forms in the combustor than in the nozzle region.

A study was performed to determine the thermal resistivity of the silica film deposited on the nozzle wall. It was judged that thickness/thermal conductivity, t/k, value of 0.00025 in-ft²-hr°F/BTU-in. correlated with the reduction in nozzle heat flux observed during thrust chamber injector fire tests. Assuming a typical thermal conductivity value for SiO_2 of 43 BTU-in/in-ft²-hr°F (Reference 2) implies a silica film thickness of about 0.001 inch.

Motion pictures of the fire tests clearly showed the self replenishing characteristic of the silica film on the nozzle surface including the throat region. Some areas of film appeared to spall away from the wall but were immediately covered with new deposit.

The effect of oxidizer suction pressure on pump performance was also determined during Test 575. Pump inlet pressure decay was initiated to 180 seconds into the test and continued for an additional 160 seconds. Data from this test showed that pump head rise was unaffected until inlet pressure decayed to 10 psia, at which time a 1% degradation in head occurred. These conditions correspond to a net positive suction head of 10.4 feet. The test verified that satisfactory operation results at net positive suction equivalent to that required for IRFNA.

Test 576 was conducted with an in-run propellant temperature change to establish the effect of temperature on performance. A c* increase of 15 ft/sec was indicated when the fuel and oxidizer temperature was switched from 32° to 87°F.

The final test of the definition phase, Test 577 was of 640 second duration durability firing. Following this test it was observed that a white solid deposit 20 to 30 mils thick formed on the thrust chamber wall near the injector face. The deposit was identified as silica (SiO_2) with an apparent density of approximately 50 lbs/ft³.

The engine was disassembled and inspected. Evidence of loose gray colored powder-like deposit was observed on the internal surfaces of the gas generator and turbine manifold. Turbine nozzles and rotor blade passage surfaces were normal.

The flight worthiness engine test series was conducted with Engine S/N 661 following the initial checkout test with IRFNA/UDMH, three firings were conducted operating with HDA/UDMH + 1% Si.

Satisfactory engine operation was demonstrated with propellants conditioned to 60, 32 and 90°F. A ground safety shutdown occurred during test 580 when propellant switching was sequenced. This was not associated with engine operational requirements and resulted in a shutdown with minimum oxidizer postflow which left some of the combustion side silica deposit intact. Photographs of the injector and combustor wall, and nozzle surface after water flush are shown in Figures 8 and 9.

Figure 8. Post-Test 580, View of Injector and Combustor Wall

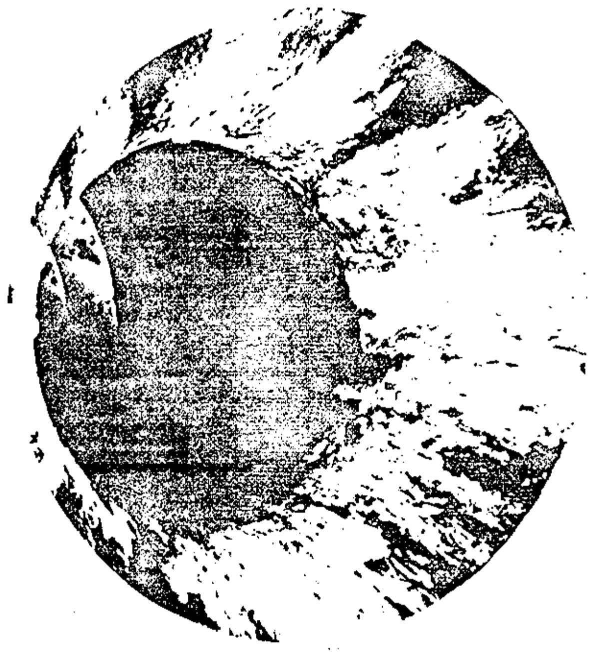


Figure 9. Post-Test 580, View of Nozzle Surface

FLIGHT TEST

The launch of an Agena vehicle utilizing HDA/UDMH + 1% Si, was successfully accomplished by Lockheed Missiles and Space Company. Post flight performance analysis based on vehicle trajectory measurements indicated a specific impulse gain of 6 lb-sec/lb.

CONCLUSIONS

Satisfactory operation of the current production engine was demonstrated utilizing HDA/UDMH + Si propellants at temperatures of 32, 60 and 90°F.

A characteristic velocity performance gain of 1.0 to 1.5% was obtained relative to IRFNA/UDMH propellants. Nozzle heat flux was reduced by 33% by formation of a combustion gas side silica film achieved with the addition of 1% silicone fluid in the UDMH.

Oxidizer pump minimum net positive suction head required for HDA operation was determined to be equivalent to that required for IRFNA.

Engine flight worthiness for single burn missions was demonstrated.

REFERENCES

1. Heat Transfer Characteristics of RFNA Bell Aerospace Company Report No. 117-982004, July 1956

2. Technical Documentary Report ML-TDR-64-5, April 1964