EXPERIMENTAL INVESTIGATION OF AN INTEGRATED STRUT-ROCKET / SCRAMJET OPERATING AT MACH 4.0 AND 6.5 CONDITIONS

K. W. Nelson and C. W. Hawk
The University of Alabama in Huntsville
Huntsville, AL

ABSTRACT

A series of tests were conducted to investigate RBCC performance at ramjet and scramjet conditions. The hardware consisted of a linear strut-rocket and a dual-mode scramjet combustor. The hardware was tested at NASA Langley Research Center in the Direct Connect Supersonic Combustion Test Facility at Mach 4.0 and 6.5 simulated flight conditions. The objective of the research was to test the hardware with and without rocket operation at these conditions. The data could then be analyzed to determine whether or not operating in this mode has any performance advantages over operating as a pure airbreather. Though the rocket chamber pressure capability was not as high as was desired, the data suggests that the additional thrust of the rocket makes this mode a viable option assuming that the vehicle (inlet) drag and weight penalties are not too severe. Other findings suggest that a different type of fuel injection than that which was tested be used to increase performance. The rocket also behaved as a pilot / flame-holder at the higher Mach number condition, where the burning of the injected fuel was otherwise difficult to maintain, unless at high flow rates. The thrust measurement system installed for this program gave good results compared with the thrust calculated from pressure integration.

INTRODUCTION

A rocket-based combined-cycle (RBCC) engine is unique in that it combines rocket and airbreathing components into a single propulsion unit. There are many variants of the RBCC, but perhaps the simplest are the ejector ramjet and ejector scramjet. Modern ejector scramjet RBCCs are basically modular, or 2-D, scramjet ducts with several rocket ejectors mounted in the bases of fuel injector struts or in steps along the side-walls of the duct. A number of these designs are currently being tested under the NASA - Marshall Space Flight Center ARTT program.

The modes of operation of the RBCC vary as the vehicle accelerates through the atmosphere, and into space in the case of a launch vehicle. In general, the ejector scramjet's modes are from air-augmented rocket through Mach 3, ramjet and scramjet through Mach 8 to 15, followed by a conventional rocket mode with a very large area ratio to orbit. It is the ramjet and scramjet modes that were the focus of this study.

BACKGROUND

For an accelerator class vehicle, which includes launch vehicles, it can be shown that both specific impulse (I_{sp}) and thrust to weight ratio (T/W) are of great importance to the overall performance of the vehicle. Starting with a simple free body diagram of a vehicle with a horizontal trajectory and assuming small off-axis angles, summing the forces in the direction of travel yields $F = T - D$. Here, T is the thrust and D is the drag.

For an accelerator, the change in velocity per unit change in mass is critical. dV/dm can be derived as follows:

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$$\frac{dV}{dm} = \frac{dV/dt}{dm/dt} = \frac{a}{\dot{m}}$$

$$\frac{dV}{dm} = \frac{F}{\dot{m}m} = \frac{T - D}{\dot{m}m}$$

$$\frac{dV}{dm} = \frac{I_{sp}g}{m}\frac{T - D}{T}$$

$$\frac{dV}{dm} = \frac{I_{sp}g}{m}\frac{T/W - \dfrac{L/W}{L/D}}{T/W}, \qquad (1)$$

where L is lift and W is weight. Assuming T/W, L/W, L/D and I_{sp} are averaged values over the velocity range of interest, integrating (1) yields

$$\Delta V = -I_{sp}g\frac{T/W - \dfrac{L/W}{L/D}}{T/W}\ln(1-\zeta), \qquad (2)$$

where ζ is the propellant mass fraction.

By choosing a ΔV and ζ, a curve can be generated from equation (2) as shown in Figure 1. Based on the operational range of a ramjet / scramjet (typically 3 < M < 8), the curve shown corresponds to a ΔV of 5000 ft/sec, a L/D of 2.5, and a L/W of 1.0. A ζ was chosen such that the curve passes through the T/W and I_{sp} values of H_2/O_2 rockets. Note that typical ramjet / scramjet powered vehicles lie on the vertical portion of this curve where $T \approx D$.

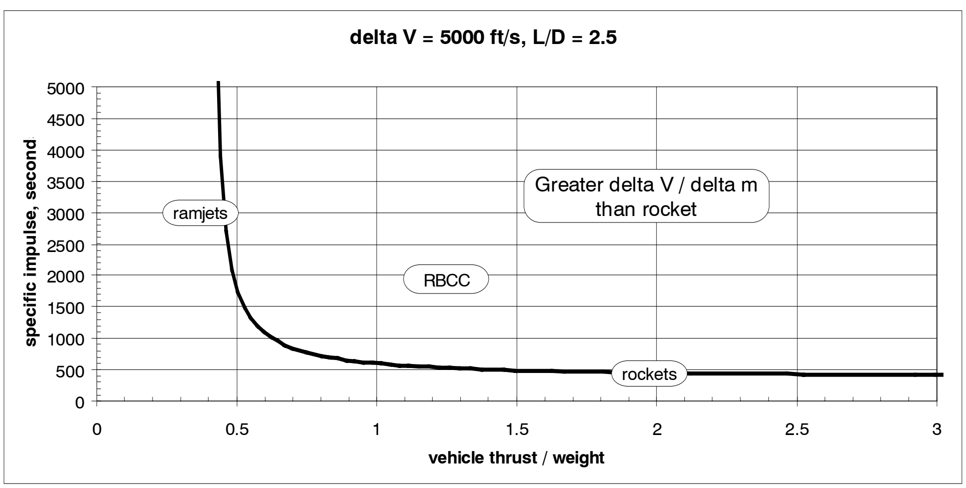

Figure 1. - Plot of ΔV equation for horizontal accelerator.

From this simple derivation, vehicle / propulsion systems located above and to the right of the curve would have better overall performance (lower ζ) than the systems that are currently available. In the case of a RBCC, this relation suggests that operating the primary rocket ejectors

in the ramjet / scramjet mode will indeed reduce I_{sp}, but overall performance could be significantly greater due to the additional T/W.

OBJECTIVE

The main objective of this research was to test an integrated strut-rocket / scramjet and collect data both with and without rocket ejector operation. In so doing, the information gathered should provide a better understanding of the dynamics of the flow in this combined mode as well as determine whether there are any advantages or disadvantages to operating the rocket while in an airbreathing mode. Other objectives of this study were to develop a means to measure the thrust produced during the runs directly and to generate an early RBCC database.

APPARATUS

The experimental apparatus consisted of a linear, strut-rocket and a dual-mode scramjet combustor. The hardware was tested in the Direct Connect Supersonic Combustion Test Facility at NASA Langley Research Center. The facility's hydrogen / oxygen / air vitiated heater is capable of simulating flight total enthalpies up to Mach 7.5. For this study, a Mach 2.5 centerbody, facility nozzle was used to mate the heater to the scramjet combustor. The centerbody nozzle served as the leading edge of the strut that housed the rocket ejector.

The gaseous hydrogen / oxygen linear, strut rocket was manufactured and previously tested by Aerojet Propulsion Company.[1] The assembly contains three individual, rectangular (2-D) rockets separated by structural stiffeners. The rocket assembly is both water cooled and hydrogen film cooled. A 20% silane solution (SiH_4) was used to "backlight" the rocket.

The scramjet duct consisted of three sections: a constant area combustor, a divergent section, and an expansion section, or exit nozzle. The initial length of the constant area combustor housed the rocket strut assembly, where the rocket was fastened vertically between two nickel plates. At the base of the plates are eight 1/8 in. (3.2 mm) diameter sonic fuel injectors that injected gaseous hydrogen tangential to the vitiated air flow. For more detailed information on the apparatus and hardware, refer to Nelson, et al.[3]

INSTRUMENTATION

Each of the gaseous supply lines to the facility heater, fuel injectors, and strut rocket were equipped with a metering orifice and instrumented to calculate the mass flow rates. Heater total pressure, total temperature, and rocket chamber pressure were also measured. Over 200 static pressure taps were located on the scramjet duct walls. In addition, a direct thrust measurement system, which is described in detail below, was designed for this experiment.

THRUST MEASUREMENT

Since the flowfield of an ejector type RBCC is very difficult to determine analytically, and interpreting the pressure integral from static wall pressure taps can be misleading, a direct thrust measurement system was developed for this experiment. Two 1 in. thick stainless steel plates with a slip-joint, o-ring seal were located between the facility nozzle section and the scramjet combustor section as shown in Figure 2. A tension/ compression, strain-gage load cell rated to 3000 lb. was located between the plates. The plates provided a metric break so the net forces downstream of the facility nozzle are independent of the facility. Since the rocket and injector plates are fixed to the scramjet combustor section, their contribution was also measured. Two adjustable carriages were designed to suspend the scramjet hardware from large "H" beams with linear bearings. The pillow-block bearings allowed the hardware to move axially. This arrangement also provided a means to align the individual pieces and simplified hardware assembly and disassembly.

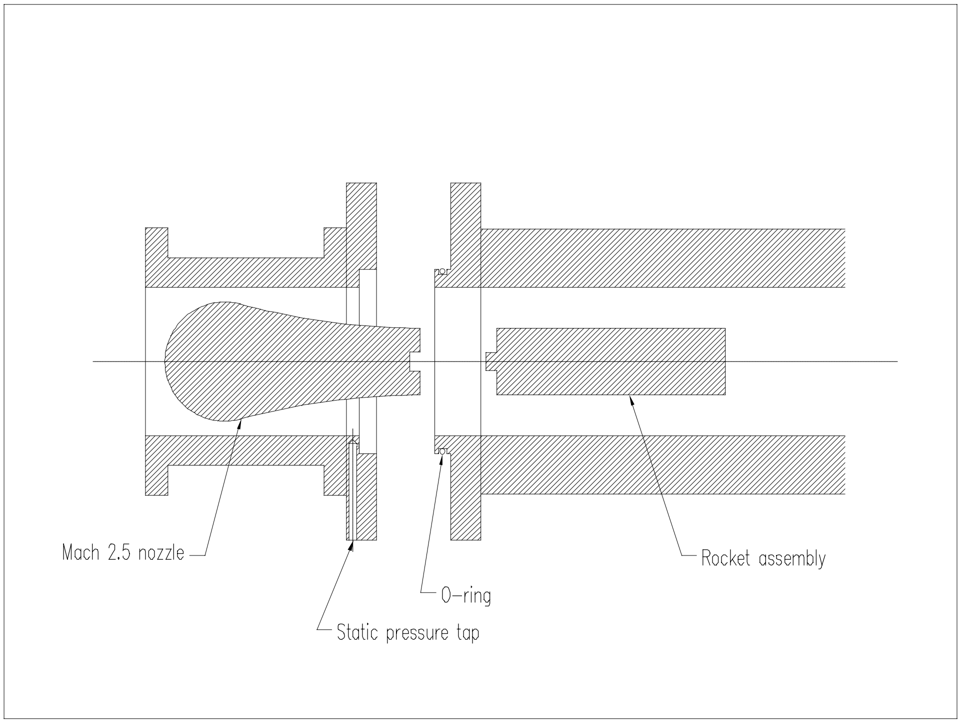

Figure 2. - Illustration of metric break for thrust measurement.

EXPERIMENTAL METHOD

FACILITY OPERATING CONDITIONS

The facility operating conditions chosen correspond to total temperatures at flight Mach numbers of 4.0 and 6.5. The total temperature values were 1600 °R and 3400 °R (890 K and 1900 K), respectively. The overall heater mass flow rates were such that an oxygen mole fraction of 20.95% was maintained. Since the effect of over and under-expansion of the rocket was of interest and the injected hydrogen mass flow rates were limited, a nominal heater exit pressure of 0.5 atm (50 kPa) was chosen. Table 1 lists the nominal heater conditions at the chosen total temperatures.

M_∞	T_T °R	P_T psia	P_2 atm	mdt lbm/s	mdt_{O2} lbm/s
4.0	1600	130	0.5	9.4	2.2
6.5	3400	130	0.6	6.0	1.5

Table 1. - Heater operating conditions.

ROCKET OPERATING CONDITIONS

The key parameters for the rocket operation were the oxidizer to fuel ratio (O/F), the chamber pressure (P_T), and the film-cooling fraction. The injected O/F ratio for this experiment was varied from 0 to 8, O/F=7.9 being stoichiometric. The amount of rocket film-cooling is characterized as a percentage of the total hydrogen flow into the rocket. A nominal value of 40%, also used during the Aerojet tests, was chosen for this experiment. The rocket operating pressure was limited by two factors, both related to film-cooling. The first limitation was the hydrogen supply pressure. Due to the pressure losses through the small film-cooling circuits, film-cooling flow rates greater than 0.14 lbm/s (0.068 kg/s) were not attainable. The second limitation was heat flux. The maximum allowable heat flux chosen for this study was 15 Btu/in^2-s (27 kJ/cm^2-s). This value was in agreement with the previous Aerojet tests.

Some theoretical rocket conditions based on a chamber pressure of 300 psia (2.07 MPa) are presented in Table 2. Also, Figure 3 shows the operating range for the rocket with a film-cooling fraction of 40%. Unfortunately, even after the installation of a new high pressure hydrogen and oxygen system, the desired chamber pressures of 2000 psia (14 MPa) achieved by Aerojet were unobtainable.

O/F	P_T psia	T_T °R	mdt lbm/s	P_2 atm	M_2	mdt_{H_2} lbm/s	mdt_{flm} lbm/s
2.00	300	3620	0.328	0.149	3.77	0.082	0.073
4.00	300	5420	0.350	0.197	3.43	0.035	0.047
6.00	300	6110	0.382	0.251	3.17	0.013	0.036

Table 2. - Rocket operating conditions at P_T=300 psia.

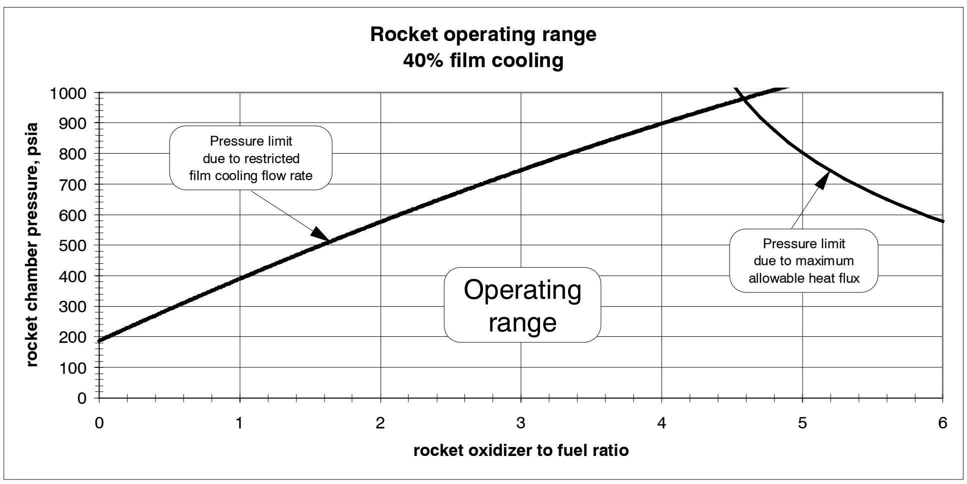

Figure 3. - Rocket operating range with 40% film cooling.

INJECTOR OPERATING CONDITIONS

As previously mentioned, the strut was equipped with eight tangential fuel injectors. The maximum hydrogen flow rate through these injectors was 0.17 lbm/s. This flow rate corresponds to fuel equivalence ratios of 0.6 at a heater total temperature of 1600 °R and 0.9 at 3400 °R. In order to reach fuel equivalence ratios on the order of 1.0, hydrogen only was injected through the rocket (O/F=0) for some of the runs.

RESULTS

Around 80 "hot" runs were made with the vitiated heater operating and the RBCC hardware installed in the facility. Many of these runs were to check out the various systems and to establish a reliable ignition sequence for lighting and sustaining rocket combustion while the facility was running. The results that follow were obtained only from the data for runs that were designated as "good" based on mole fraction of oxygen from the heater and relative closeness to the desired heater total temperatures and pressures.

To minimize the thermo-cycling of the hardware, the time at which the rocket and injector propellants were being fed overlapped. Therefore, each run actually satisfied three test conditions. First, while the heater was operating, the rocket propellant valves were opened to the desired settings. Then, with the rocket still firing, the hydrogen valve for the fuel injectors was opened. Next, the rocket propellants were shut off, leaving only the fuel injectors. And finally, the fuel injectors were turned off and only the heater was operating. The thrust data from this latter portion of the run was averaged with the other heater only thrust data and subsequently used as the "tare" thrust. It should be noted that the method for calculating thrust was simply to subtract the "tare" thrust from the raw, measured thrust. No attempt was made to calculate inlet drag and arrive at a net propulsive thrust. Thus, these thrust values cannot be compared to conventional rocket thrust data without some type of correction.

In addition, note that for public distribution of this paper, the actual thrust values that were calculated could not be presented here. Instead, a reference thrust, which is the calculated value divided by an arbitrary constant, is shown in the tables and charts to follow. Also, the reference thrust is used in the calculation of specific impulse rather than the calculated thrust, to produce a reference specific impulse. For the actual calculated thrust and specific impulse values, refer to Nelson, et al.[4]

RAMJET SIMULATION

Table 3 below lists the run conditions and measured thrust and I_{sp} for the ramjet simulation at a heater total temperature of 1600 °R. The table is divided into three categories, rocket only, rocket + injector, and injector only. Note that for runs 99 through 106, the rocket was only supplied with hydrogen, no oxygen (O/F=0). All the ramjet simulation plots that follow were generated from this table of data. Phi, or ϕ, is the fuel equivalence ratio. The total fuel equivalence ratio is calculated based on the sum of the theoretical unburned hydrogen from the rocket, the film cooling hydrogen, and the injected hydrogen.

Rocket only 1600 °R

run #	mdt_inj	mdt_rkt	Pt_rkt	O/F_rkt	PHI_inj	PHI_tot	F ref	Isp ref
71	0	0.350	299	2.79	0	0.373	0.856	2.44
72	0	0.412	339	3.83	0	0.314	0.942	2.28
73	0	0.597	500	4.13	0	0.415	1.288	2.16
76	0	0.429	337	5.26	0	0.225	0.716	1.67
77	0	0.457	372	2.05	0	0.597	1.166	2.55
78	0	0.451	371	2.17	0	0.571	1.112	2.47
79	0	0.437	367	2.08	0	0.581	1.122	2.57
80	0	0.154	127	4.95	0	0.087	0.352	2.29
81	0	0.649	535	3.80	0	0.482	1.400	2.16
82	0	0.162	128	0.98	0	0.333	0.554	3.41
83	0	0.141	121	1.53	0	0.229	0.512	3.62
84	0	0.238	199	4.18	0	0.161	0.616	2.59
99	0	0.158	80	0	0	0.579	0.922	5.84
101	0	0.160	81	0	0	0.575	0.972	6.08
106	0	0.149	77	0	0	0.552	0.852	5.71

Rocket + injector 1600 °R

run #	mdt_inj	mdt_rkt	Pt_rkt	O/F_rkt	PHI_inj	PHI_tot	F ref	Isp ref
71	0.070	0.366	300	2.94	0.252	0.617	0.984	2.26
76	0.098	0.421	338	5.12	0.360	0.587	1.036	2.00
78	0.065	0.443	362	2.11	0.231	0.808	1.248	2.46
79	0.138	0.436	362	2.07	0.479	1.040	1.426	2.48
80	0.121	0.150	126	4.82	0.420	0.504	0.698	2.57
81	0.162	0.655	542	3.84	0.570	1.055	1.760	2.15
82	0.097	0.161	126	0.97	0.334	0.655	0.990	3.84
83	0.094	0.138	119	1.67	0.325	0.532	0.792	3.42
84	0.130	0.213	161	8.03	0.446	0.495	0.856	2.50
99	0.148	0.159	80	0	0.543	1.127	1.480	4.84
101	0.074	0.160	81	0	0.264	0.834	1.356	5.80
106	0.075	0.149	77	0	0.279	0.836	1.170	5.23

Injector only 1600 °R

run #	mdt_inj	mdt_rkt	Pt_rkt	O/F_rkt	PHI_inj	PHI_tot	F ref	Isp ref
71	0.071	0	0	0	0.257	0.257	0.504	7.04
76	0.098	0	0	0	0.359	0.359	0.600	6.13
78	0.065	0	0	0	0.232	0.232	0.522	7.99
79	0.138	0	0	0	0.482	0.482	0.932	6.75
80	0.116	0	0	0	0.396	0.396	0.764	6.59
81	0.160	0	0	0	0.561	0.561	1.052	6.57
82	0.095	0	0	0	0.326	0.326	0.666	6.97
83	0.095	0	0	0	0.323	0.323	0.656	6.93
84	0.129	0	0	0	0.453	0.453	0.982	7.65
99	0.142	0	0	0	0.518	0.518	0.926	6.55
101	0.076	0	0	0	0.267	0.267	0.560	7.41
106	0.075	0	0	0	0.280	0.280	0.494	6.54

Table 3. - Data for ramjet simulation (1600 °R).

Looking first at specific impulse, Figure 4 is a plot of specific impulse versus rocket O/F for the rocket only and rocket + injector tests. The data points appear to follow a trend where lower rocket O/F values provide higher I_{sp} independent of rocket chamber pressure. This is somewhat expected since less oxygen is being used at the lower O/F ratios. What is not expected is that the I_{sp} is not effected by the additional fuel through the fuel injectors. One might initially predict that the I_{sp} would increase with the added fuel, but this is indeed not the case here. This is the first indication that when the fuel is injected while the rocket is in operation, the fuel does not burn efficiently.

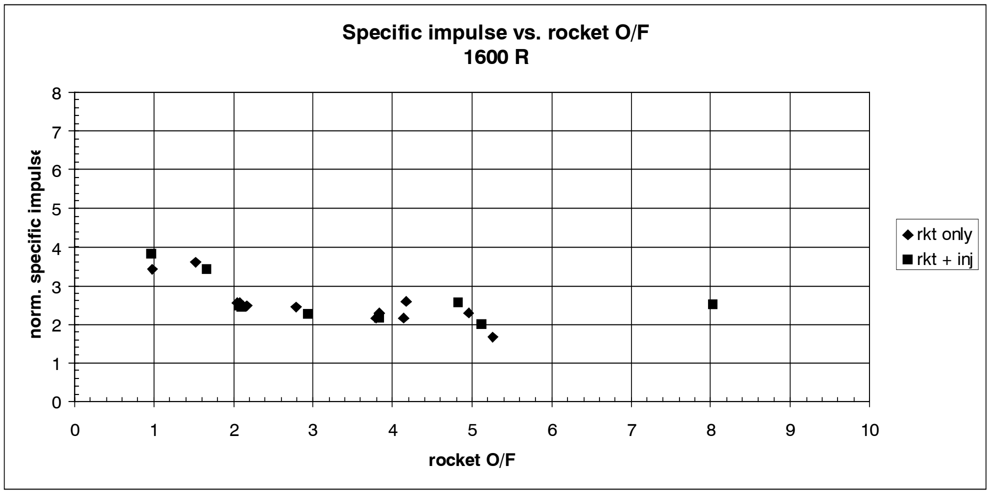

Figure 4. - I_{sp} versus O/F (1600 °R).

Similarly, Figure 5 is a plot of specific impulse versus fuel equivalence ratio for the H_2 only rocket tests and the injector only tests. From this plot one can see that the I_{sp} with the H_2 fed rocket is less than that of the H_2 injectors. Considering that the hydrogen exiting the rocket is around Mach 3.5, it is much colder than the hydrogen exiting the injectors. Thus, the "supersonic injector" rockets are less efficient than the sonic injectors and produce less thrust per unit mass flow.

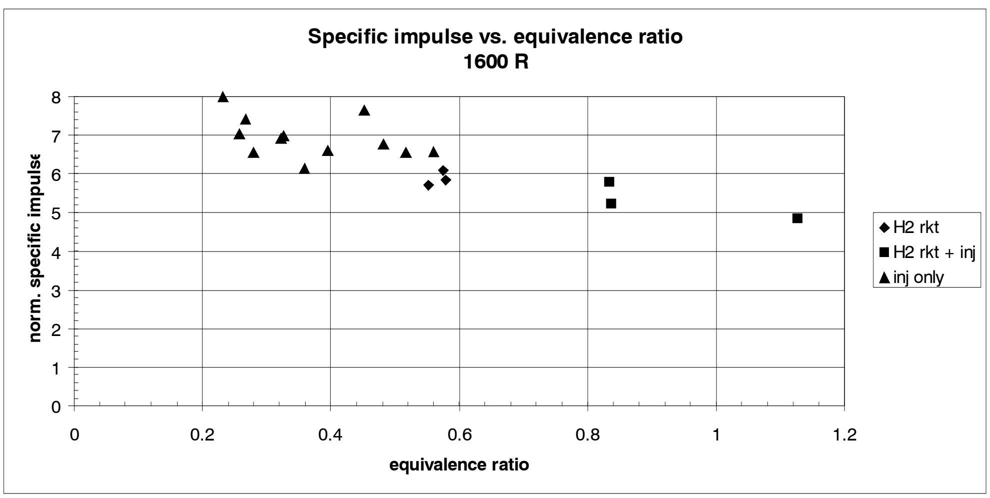

Figure 5. - I_{sp} versus ϕ (1600 °R).

All the tests can be compared on a single plot of thrust versus propellant mass flow rate as in Figure 6. The slope of a line passing through the origin and the data point of interest is equivalent to the specific impulse. The hydrogen only data, solid symbols, lies along a fairly straight line with a larger slope (greater I_{sp}). The scatter in the H_2/O_2 rocket data is mostly due to the various rocket O/F values tested. Typically, lower rocket O/F yields higher I_{sp} as illustrated in Figure 4.

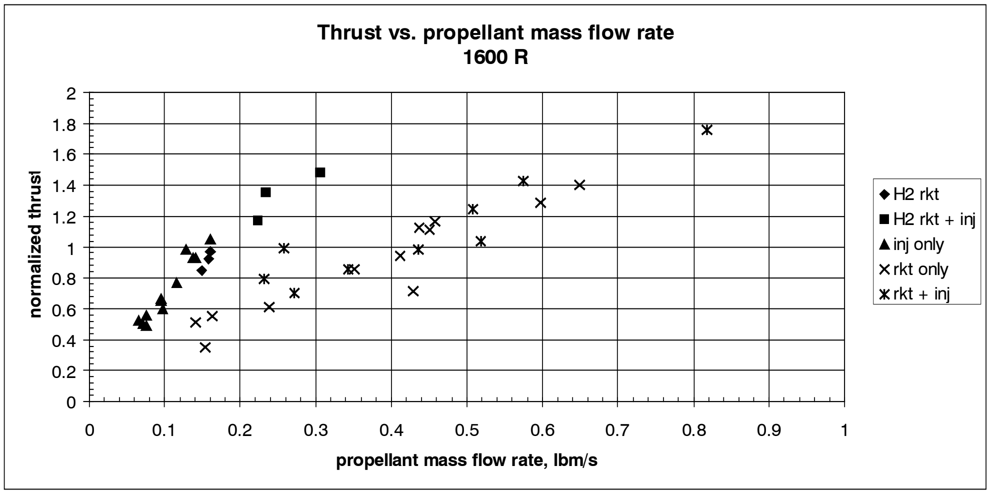

Figure 6. - Thrust versus mass flow rate (1600 °R).

Previously, Figure 4 suggested that the rocket actually hinders the hydrogen from the fuel injectors from burning. In fact, a close examination of runs 80 and 84 (Table 4) reveals that the thrust produced by the injectors alone in these runs, is actually greater than the thrust produced with rocket + injectors, at the same injector mass flow rate. The pressure profiles for runs 80 and 84, which could not be included in this paper for distribution reasons, support this conclusion. Apparently, the fuel from the injectors is being entrained into the rocket exhaust and "carried" downstream before it is able to mix with the air and burn. Under these circumstances, injecting the fuel from the side-walls would most likely have been a more efficient means of injection.

Run # 80

	mdt_inj	*mdt_rkt*	*Pt_rkt*	*O/F_rkt*	*PHI_inj*	*PHI_tot*	*F ref*	*Isp ref*
rocket	0	0.154	127	4.95	0	0.087	0.352	2.29
rkt+inj	0.121	0.150	126	4.82	0.420	0.504	0.698	2.57
injector	0.116	0	0	0	0.396	0.396	0.764	6.59

Run # 84

	mdt_inj	*mdt_rkt*	*Pt_rkt*	*O/F_rkt*	*PHI_inj*	*PHI_tot*	*F ref*	*Isp ref*
rocket	0	0.238	199	4.18	0	0.161	0.616	2.59
rkt+inj	0.130	0.213	161	8.03	0.446	0.495	0.856	2.50
injector	0.129	0	0	0	0.453	0.453	0.982	7.65

Table 4. - Data for runs 80 and 84.

Figure 7 is similar to Figure 1 except the vehicle drag and weight is not known so measured thrust is used instead of net thrust and T/W. Since the same hardware is being compared for all these tests, measured thrust is sufficient. As before, data points toward the upper right hand corner of the graph are the most desirable. Perhaps a better method of comparing the various tests is to choose those with the same total fuel equivalence ratio. Figure 8 is the same as Figure 7 but only includes tests points with total fuel equivalence ratios of around 0.53.

Clearly, for the hydrogen only tests (airbreather), the higher fuel equivalence ratios are the most desirable. These conditions produce greater thrust and the specific impulse is held pretty well constant to a point. In the case of the integrated H_2/O_2 rocket / ramjet, the reference specific impulse was around 2.6 for the range of rocket chamber pressures tested. Assuming this trend would continue for rocket operation at higher chamber pressures, the integrated rocket / ramjet may actually have better performance due to the enormous amount of thrust capability even though the I_{sp} is considerably less than the H_2 only tests. Note, however, that this assumption depends heavily on the vehicle (inlet) drag and weight. In addition, the specific impulse of the integrated system could be improved by injecting the additional fuel from the side-walls instead of the strut.

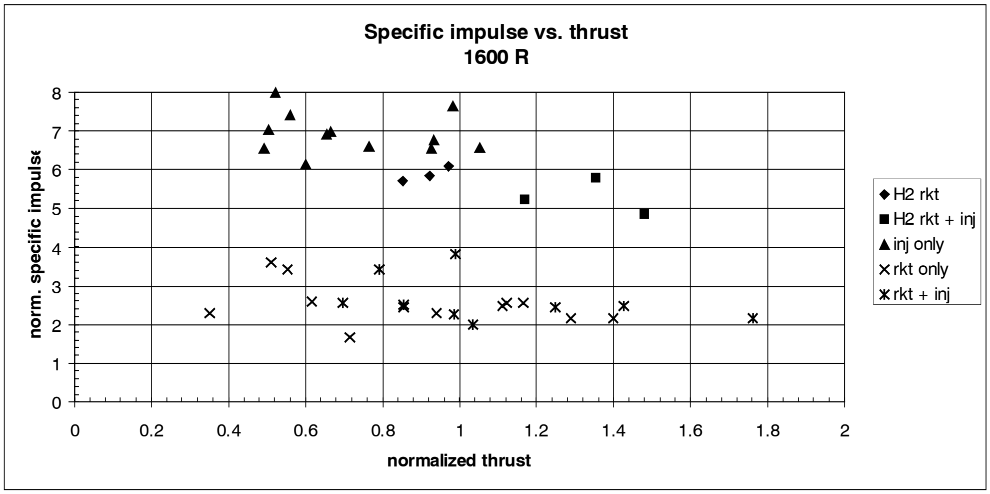

Figure 7. - I_{sp} versus thrust (1600 °R).

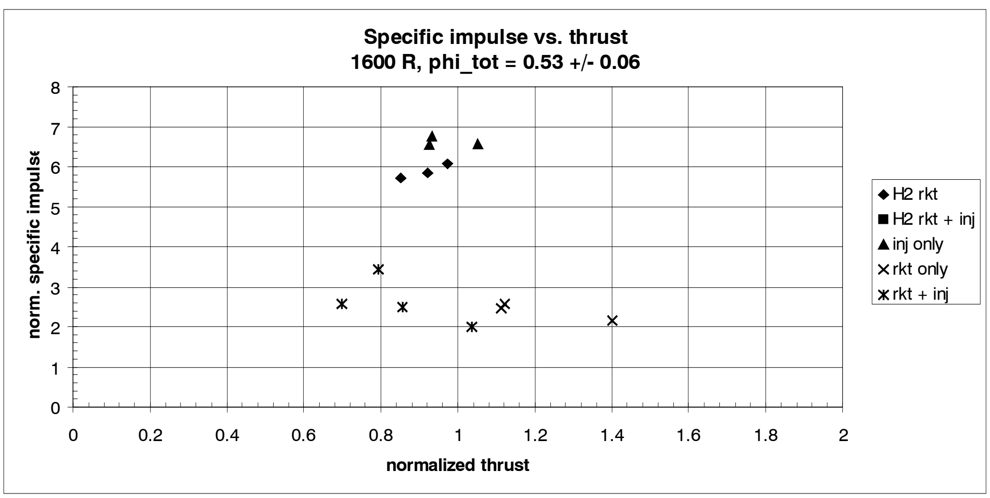

Figure 8. - I_{sp} versus thrust for $\phi \approx 0.53$ (1600 °R).

SCRAMJET SIMULATION

Table 5 lists the run conditions and measured thrust and I_{sp} for the scramjet simulation at a heater total temperature of 3400 °R. Again, the table is divided into three categories, rocket only, rocket + injector, and injector only. At this heater temperature it was very difficult to establish and maintain combustion of the hydrogen from the fuel injectors unless the rocket was in operation. Since it was desirable to reduce the run time on the hardware anyway, the injector only tests were discontinued after run 90 but resumed during the H_2 rocket tests. It was found that the injected hydrogen only burned if the fuel equivalence ratio was greater than 0.6. Though unexpected, this phenomena can be accounted for based on previous studies.[2] Tests where the fuel did not burn are not included in this table. Also, none of the hydrogen only rocket tests burned and were therefore also excluded from the table.

Rocket only 3400 °R

	mdt_inj	*mdt_rkt*	*Pt_rkt*	*O/F_rkt*	*PHI_inj*	*PHI_tot*	*F ref*	*Isp ref*
86	0	0.419	339	3.71	0	0.487	0.520	1.24
87	0	0.419	335	3.97	0	0.432	0.532	1.27
88	0	0.414	337	3.85	0	0.451	0.560	1.35
89	0	0.402	322	1.92	0	0.814	0.706	1.76
90	0	0.424	334	4.94	0	0.350	0.504	1.19
91	0	0.371	313	4.71	0	0.331	0.560	1.51
92	0	0.112	99	3.93	0	0.123	0.358	3.21
93	0	0.630	501	4.73	0	0.559	0.958	1.52
94	0	0.688	552	3.66	0	0.813	0.912	1.33
95	0	0.725	544	6.72	0	0.370	0.900	1.24
96	0	0.194	150	1.56	0	0.458	0.294	1.51
97	0	0.284	209	7.90	0	0.108	0.254	0.89
98	0	0.163	129	3.94	0	0.174	0.244	1.49

Rocket + injector 3400 °R

	mdt_inj	*mdt_rkt*	*Pt_rkt*	*O/F_rkt*	*PHI_inj*	*PHI_tot*	*F ref*	*Isp ref*
86	0.096	0.414	339	3.65	0.517	1.011	0.846	1.66
87	0.163	0.414	336	3.91	0.831	1.268	0.994	1.72
88	0.050	0.405	338	3.69	0.257	0.716	0.792	1.74
89	0.019	0.401	321	1.88	0.100	0.924	0.960	2.28
90	0.123	0.438	338	5.12	0.652	0.991	0.880	1.57
91	0.123	0.382	318	4.90	0.655	0.972	1.074	2.12
92	0.117	0.111	97	3.90	0.608	0.730	0.764	3.35
93	0.037	0.638	512	4.89	0.204	0.748	1.106	1.64
94	0.039	0.688	562	3.64	0.211	1.028	1.124	1.55
95	0.081	0.716	553	6.65	0.429	0.797	1.184	1.49
96	0.126	0.193	149	1.55	0.668	1.132	0.592	1.85
97	0.170	0.286	209	7.82	0.888	0.998	0.720	1.58
98	0.174	0.158	124	4.17	0.906	1.063	0.730	2.19

Injector only 3400 °R

	mdt_inj	*mdt_rkt*	*Pt_rkt*	*O/F_rkt*	*PHI_inj*	*PHI_tot*	*F ref*	*Isp ref*
87	0.164	0	0	0	0.833	0.833	0.652	3.98
90	0.129	0	0	0	0.690	0.690	0.560	4.35
111	0.160	0	0	0	0.826	0.826	0.618	3.85

Table 5. - Data for scramjet simulation (3400 °R).

The hydrogen's unwillingness to burn was not a complete loss. It supported a potential benefit conceived at the beginning of this study that the rocket could also act as a flame-holder and/or pilot in the airbreathing mode. Running in this combined mode could, therefore, reduce the need for complicated injection schemes.

There is some question with regards to the thrust measurement of run 92. In all of the other runs, the thrust is proportional to the rocket chamber pressure in the rocket only tests. At this time, no single rational reason for this anomaly has been pinpointed. One possibility is that offset is within the uncertainty of the calculation since the mass flow rates and rocket chamber pressure are much lower in this run than any of the others. For the time being, the data for run 92 has been included in the table, but is not included in the plots that follow.

In Figure 9, the rocket + injector tests performed slightly better than the rocket only tests. Recall that for the ramjet conditions the injector did not significantly improve rocket only performance. Also, the average I_{sp} is much less than that for the ramjet tests, as would be expected. This was also the case for the injector only tests as indicated in Figures 13 and 14.

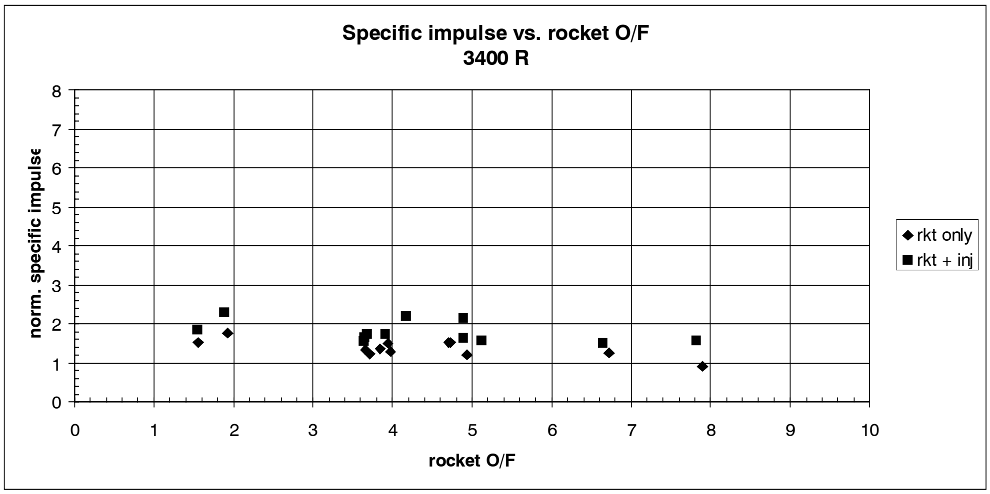

Figure 9. - I_{sp} versus O/F (3400 °R).

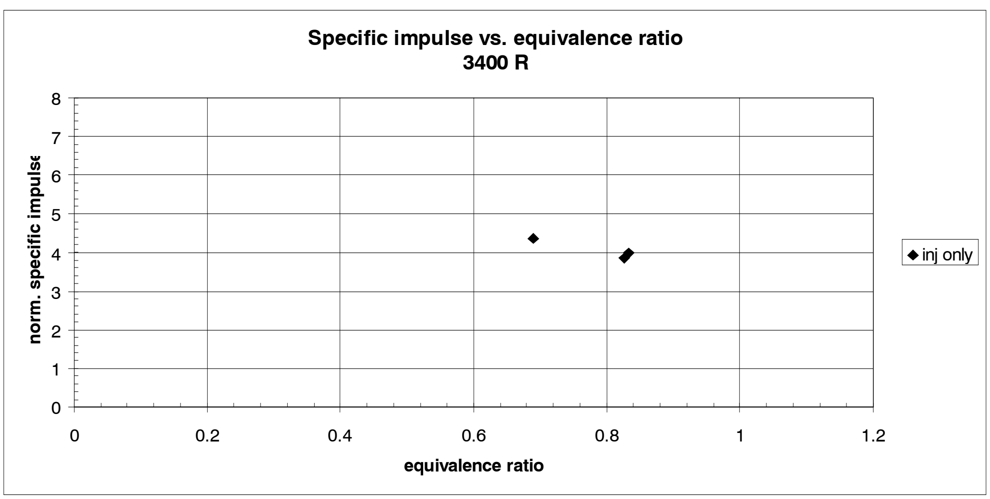

Figure 10. - I_{sp} versus ϕ (3400 °R).

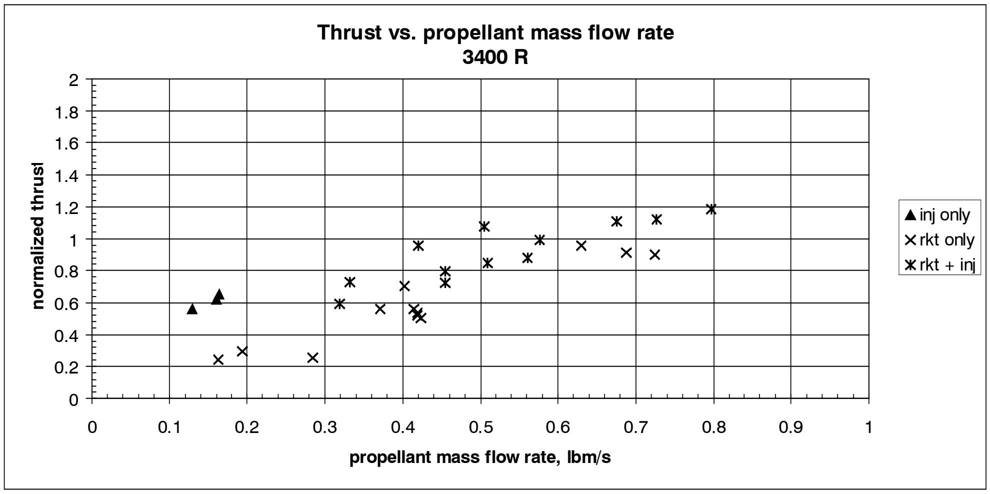

Figure 11. - Thrust versus mass flow rate (3400 °R).

As before with the ramjet simulation, Figures 15 and 16 compare the specific impulse and measured thrust of the individual tests, where Figure 13 includes only those tests where the total fuel equivalence ratio was around 0.76. Once again, the increased thrust of the combined operation of the rocket and airbreather shows potential. As mentioned previously, however, this greatly depends on the drag and weight of the vehicle.

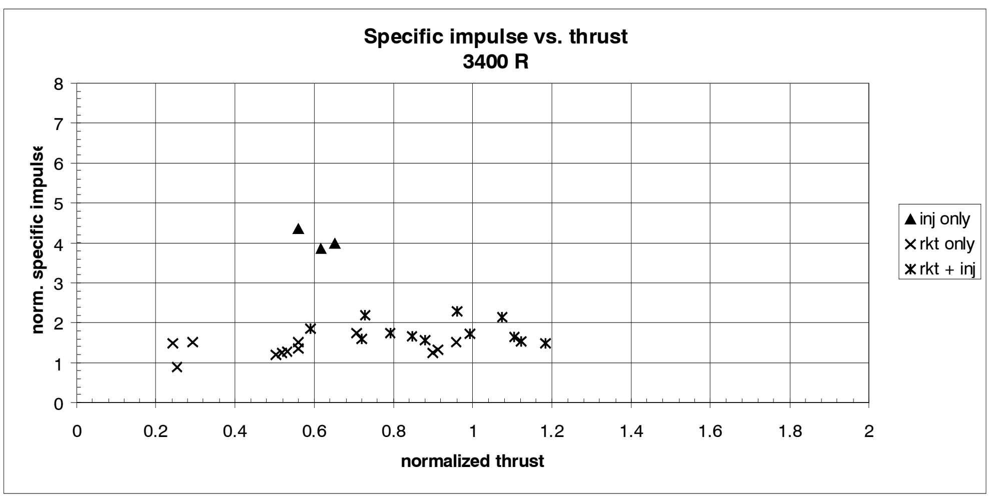

Figure 12. - I_{sp} versus thrust (3400 °R).

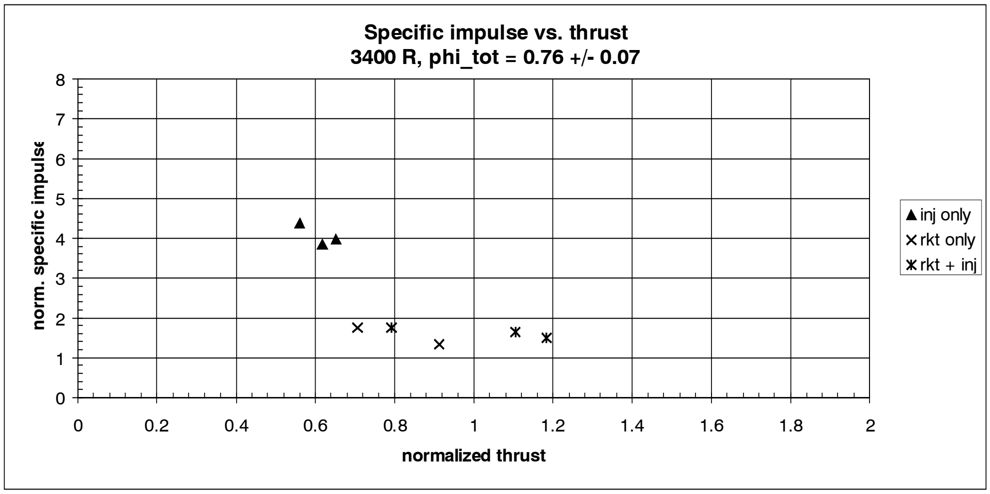

Figure 13. - I_{sp} versus thrust for $\phi \approx 0.76$ (3400 °R).

MEASURED VERSUS CALCULATED THRUST

The measured thrust from the load cell compared very well to the calculated thrust from pressure integration for the ramjet simulation. Here the difference in the two values was less than 15%, with the measured value always being the greater of the two. Given that the pressure integration method is generally less accurate because of the spacing of the pressure taps and the non-uniformity of the flow, more confidence was placed in the measured thrust data.

For the scramjet simulation, however, the error was up to 30% and there was no consistency with which value was greater. For the rocket only tests, the measured data was linearly proportional to the rocket chamber pressure as one would expect. Some contributors to the large difference in the two values could be the stronger influence of the shocks at the higher temperature condition and the possibility of heating effects on the load cell.

SUMMARY AND CONCLUSIONS

Based on the data collected during this study, rocket operation during the ramjet and scramjet modes is worth further investigation. The tremendous thrust advantage of the added rockets may very well overcome the specific impulse penalty in terms of overall performance. Both vehicle drag and weight, which were not included here, will strongly influence the overall performance of a vehicle operating in this mode.

During the ramjet simulation, the negligible additional thrust from the fuel injectors during rocket operation indicated that the injected hydrogen was not burning efficiently. Apparently, the fuel is being entrained into the rocket exhaust and "carried" downstream before it as able to mix with the air and burn. Under these circumstances, injecting the fuel from the side-walls would most likely have been a more efficient means of injection.

At the higher heater total temperature it was very difficult to establish and maintain combustion of the hydrogen from the fuel injectors unless the rocket was in operation. This information helped substantiate another application of the rocket as a pilot / flame-holder. The hot rocket exhaust could thus alleviate much of the complexity in efficient injector designs.

Finally, the measured thrust from the load cell compared very well to the calculated thrust from pressure integration for the ramjet simulation. Though the error was greater in the scramjet simulation, a great deal of confidence is placed in the measured values.

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