8. ADVANCED ENGINE SYSTEMS

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The evolution in propulsion systems as we now see it is shown in figure 1. The M-1 engine will be available in about the 1967 to 1968 time period and will produce 1.5-million pounds of thrust. I might say that the M-1 is adaptable, or can be slightly modified, to use a forced-deflection-type nozzle. Considerably higher performance will result (approximately 435 seconds of specific impulse). The advanced engine, which is the main concern of these discussions, is shown in figure 1. Note that the specific impulse has increased from 430 to 450 seconds at altitude. In the 1978 to 1979 time period, ducted rockets are shown adapted with the engine itself. A winged vehicle also using the advanced ducted rocket concept is shown in the 1980 time period.

Figure 2 shows a configuration of the advanced engine. The sea-level specific impulse is 383 seconds, whereas the vacuum specific impulse is 450 seconds. This engine will operate at a 2,500-psia thrust chamber pressure. It uses a staged-combustion cycle which I will discuss subsequently. It also incorporates a forced-deflection nozzle for altitude compensation.

We are operating at the optimum mixture ratio of 6.0. This is not necessarily the optimum mixture ratio of the engine, but it is the optimum mixture ratio of the stage itself. One of the salient features of the engine is the single integrated pump, around which is clustered primary and secondary combustors. There are 12 primary combustors (gas generators) and secondary combustors (main thrust chambers) clustered around a single centrally located pumping system which feeds into a single nozzle skirt. The thrust vector control is obtained by secondary injection. The figure also shows the forced-deflection nozzle.

For the purpose of this study we have chosen 24-million pounds of thrust, which about fits the Nova application. In addition to that, we have selected a tank diameter of 80 feet. It is quite important that we have a tank diameter in order to size the nozzle.

Figure 3 shows staged-combustion and gas generator engine cycles. The gas generator drives the turbine and the turbine gas is discharged overboard. The turbine drive fluid is not completely combusted before being exhausted from the engine; thus, potential thrust is lost. You can gain back some of this energy, but a good part of it is lost.

In the staged combustion cycle shown on the left the fuel pump and the oxidizer pump are placed as indicated. Part of the fuel from the pump goes through the combustion chamber to cool it. A major part of it flows into the primary combustor. Oxidizer is injected with the full flow to give a gas temperature in the primary combustor from 1,000° to 1,600° F. All of this gas drives the pump; the gas is discharged into the combustion chamber and burned at an optimum mixture ratio. Therefore, the energy that is normally lost from the gas generator cycle is gained in the staged-combustion cycle.

Figure 4 is an artist's sketch of the advanced engine. We see that the pumps are located with one pump on top and one pump on the bottom. One pump feeds one propellant and the other pump feeds the other propellant into the primary combustors. The gas is then discharged through the turbine to drive the pumps. The gas is then collected in a common manifold and discharged into the secondary combustors.

The figure shows the secondary combustors and the forced-deflection nozzle extension. The complete system is not shown. At the top is shown the tank bottom itself, with the upper pump located in the tank.

A modular engine can be used with a plug or forced-deflection system. Such a configuration is shown in figure 5. About a 2-million-pound thrust module fits the forced-deflection size in which we would cluster 12 to 14 modules in a forced-deflection nozzle.

If the plug nozzle is used, since it has a larger diameter, the thrust rating would be somewhat lower, but you would need a few more engines. The configuration shown uses 16 engines. The oxidizer pump uses a separately driven inducer, either hydraulically or mechanically driven. The nozzle configuration can be seen on the right.

The installation is shown in figure 6. The specific impulse we have already discussed. In regards to the weight, the 312,000 pounds includes the thrust structure; it is determined on the basis of wet weight, including the frames, lines, and the propellants in the lines. So, it is the complete weight as attached to the missile tank itself. PFRT is 1970; qualification is 1973. Note that the area ratio is determined by the tank diameter. This one has an area ratio of 120.

The modular concept is shown in figure 7. We have in this configuration 12 clustered modules. The area ratio in this case is 100, and it is based upon the fact that the throat area is larger on this configuration than it was on the previous one because spare modules are used to obtain high reliability. The wet weight is 423,000 pounds. Note that the specific impulses are somewhat lower on this configuration, with a vacuum I_{sp} of 448 seconds. This is because there is a lower nozzle area ratio. Again, development time period is about the same.

The plug nozzle configuration for the modular engine system is shown in figure 8.

Figure 9 shows five 6-million-pound-thrust engines clustered, one engine is a spare for engine-out. On the module-clustered engines (either the plug or the forced-deflection nozzle), in order to maintain the same reliability as with a single pump configuration, you have to have engine-out capability.

Therefore, you carry more engines than you actually have to have. We are talking about PFRT in 1970, quality testing in 1972. Note that the vacuum specific impulse is 448 seconds on this configuration.

There is much work being done in the area of higher performing engines, at least in the study phases, of ducted rockets, concepts of which are shown in figure 10. We think that if we are going to have high performance engines, single-stage-to-orbit, rendezvous, winged vehicles, and so forth, we should start advance technology work today, so that these engines will be available in the 1978 to 1980 or 1982 time period.

On the left is a fixed geometry duct, which is the simplest configuration. The rocket engines are clustered around the tank in this configuration. There is very little known as to actual performance of this engine in operation.

On the right is a variable geometry duct, with the rocket engines located around the tank. The intakes are different in this configuration from those in the fixed-geometry configuration. The variable-geometry configuration has a supersonic combustor and the fixed-geometry configuration has a subsonic combustor. When about Mach 8 is reached, the rocket engines are tilted over. The system now is a pure rocket with a forced-deflection-type nozzle. In this configuration we have a very large area ratio, in the neighborhood of 750.

As a further advance in this concept, and especially for winged vehicles, an improvement is obtained by the use of an air turborocket, figure 11. This engine has a precooler which is cooled by hydrogen. It cools the air down to near its saturation point. It is then compressed with a compressor; a higher compression ratio is obtainable with this system.

Note that an effective I_{SP} of approximately 770 seconds is achievable with this configuration. When about Mach 5 is reached, supersonic combustors on the wings are put into operation, which increases the pressure to get additional performance. A performance of approximately 850 seconds can be obtained. This is really an advanced type of engine. Again, I stress, we should be doing advanced technology work now.

Various nozzles, the conventional Delaval (C-D), plug, and forced-deflection nozzles, are compared in figure 12. The plot shows actual test data that have been run by Aerojet. The dash-dot curve shows the standard Delaval nozzle. The shaded area is for the forced-deflection nozzle. The heavy solid line is the 50-percent isentropic plug nozzle, and the dashed line is the zero-length plug nozzle.

All three configurations were at one area ratio. In actuality, you can't have the same area ratio because the diameters of the plug and forced-deflection nozzles are measured from different points. Thus, within a given envelope, forced deflection, because of its larger diameter, gives a higher area ratio and therefore higher performance.

Some comparisons with the Delaval engine are shown in figure 13 for a first-stage application. Vacuum specific impulse is given as a function of chamber pressure. Note that for the Delaval engine we have optimized the nozzle for its trajectory not just for altitude.

At a chamber pressure of 1,000 psia, which is about the conventional design today, we have a nozzle area ratio of 20. Vacuum specific impulse would be about 413 seconds. The solid line is for the staged-combustion cycle, and the dot-dash line is for the standard gas generator cycle.

It can be compared with the forced-deflection system. Our design point here with an area ratio of 120 would just fit the 120-foot diameter. As far as engine weight is concerned, a 20-percent weight reduction for the single-pump forced-deflection system over the conventional system is possible. Payload can be increased by 80 percent because of a lower weight and higher specific impulse.

Several hydrogen pump concepts are shown in figure 14. In the first concept a configuration with 20 stages is shown. The first eight stages of this configuration are identical to the M-1 system. The second is the M-1 configuration (an eight-stage axial-flow pump), and we have added to it a centrifugal stage to get the required pressure. We need 4,200 pounds of pressure for operation. The third concept is a two-stage centrifugal pump.

For engine installation, as engines get bigger and bigger, selection of the vector control system becomes quite important. Two advanced engine installations are shown in figure 15. In the rigid configuration we use secondary gas injection. We use heated hydrogen to give an effective gimbal angle of 1.7°, and then lox and hydrogen is used to go up to 4° or 5° effective gimbal angle. The gimbaled engine configuration is longer. We have bearing assemblies and the suction lines are longer.

In our analyses we used 6 million pounds of thrust. We save about 12,000 pounds in weight by using the gas injection configuration as compared with the gimbaled configuration. In addition, we have a simpler system and a shorter installation.

For multiple module propulsion systems considerations for engine-out are as follow:

(1) Extra engines are required
(2) Additional components for isolation
(3) Compensation for thrust vector change
(4) Detection of failures
(5) Location of secondary gas injection
(6) Trajectory correction for random thrust change
(7) Rendezvous schedule effect

Multiple engine installations are shown in figure 16, with and without redundancy. For the system without redundancy, there is a straight-through cooling system. For the system with redundancy the secondary combustor and the skirt can be seen. Note that we have to add additional manifolds, as well as two more valves in the system for each module in order to be able to cool the skirt. If we assume that an engine goes out, and we have to shut it down, the exhaust gases of the two adjacent engines are going to impinge and overheat the skirt; so, we have to cool the skirt.

As far as the advanced technology is concerned, we need to do a lot of work. (See fig. 17.) In order to get an advanced engine, whether it is the forced-deflection engine or a plug configuration, a single pump or a module, work should be started today and it should be started in earnest, a concentrated effort, in order to have this engine available in the 1970 to 1975 time period.

We need to concentrate our effort on the development of the staged combustion cycle. We have done some work on storable propellants but need to do a lot more on lox/lh_2 propellants. We are considering high chamber pressure (which applies to all of the pump-fed engine configurations). We need to do advanced technology work in areas of combustion and heat transfer.

We have selected a 2,500-pound chamber pressure rather than a higher pressure. The problems are difficult enough at 2,500 psia. We need to do work in areas such as cooling, pumping system, bearing and seals, axial thrust (as far as the pump is concerned), and thrust vector control. We need to know more about the characteristics of the staged-combustion engine system. We need to do work on split-flow impellers because in the configuration we have a high pressure flow to the primary combustor and a lower pressure to the main or secondary combustor. A split flow impeller results in higher cycle efficiency.

The next part of this discussion concerns our advanced technology as we see it and also our program plans for the development of a complete engine.

As we all know, in engine development, you get a contract to develop a full-sized engine and you go about it just about as fast as possible especially if the hardware is sizable. Any mistake that you make during the program is a very costly one, since you are conducting many program phases concurrently. You are developing it today; you are releasing hardware for PFRT configurations tomorrow; and if you have made a mistake it will be reflected in a great deal of hardware. This will be very costly.

Figure 18 depicts steps in advanced technology programs. This schedule is for a Nova application for which we need an engine in the latter part of the 1970's. Now is the time to start advanced technology work, to do this work on a scale size that is convenient to handle and economical to operate, and to work out all critical problems.

For the purpose of our study we picked a 170,000-pound-thrust engine for the breadboard engine. We have facilities available for this. It is a thrust size that we can use to work out many of the problems in staged-combustion cycle operation and heat transfer. So, we have a pump of 170,000 pounds thrust and primary and secondary combustors of 170,000 pounds thrust.

The breadboard engine nozzle is similar to the segment of the full-scale engine. It is a subscale module. With this system we can solve critical problems in the primary and secondary combustions such as heat transfer and staged-combustion cycle characteristics. This is the most economical way to solve these problems.

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The demonstration engine follows the breadboard engine. If a modular configuration is to be used - in other words, if we are using a plug or forced-deflection nozzle with individual modules - we cluster breadboard engines into a complete system. If we cluster 12 of these breadboard engines as we have developed here, the result is a 2-million-pound thrust engine. If we decide to use a single pump, then we would develop a pump for a 2-million-pound thrust engine, using the primary and secondary combustors that have been developed under the breadboard engine program. Time-wise mid-1966 is programed for the clustered module configuration, with the additional time (dashed lines) required to develop a 2-million-pound-thrust single-pump configuration.

In addition to the bearings and seals, heat transfer, performance evaluation, staged-combustion cycle investigations, and so forth, we want to do additional cold flow testing on the various nozzle configurations such as the short plug configurations. We need to do thrust vector control work. If we are going to use the modular concept, we have to investigate, in detail, the effects of engine-out on thrust vector control response and effectiveness. Again, air augmentation should be started now.

In our engine development concept (fig. 19), in developing high-chamber-pressure engines, it has been our experience as well as the results of our analysis that the use of a high-pressure pump for the development of the combustion chambers is the most economical way to proceed both in time and money. Also, the pump can be used to obtain scale-model data applicable to the larger engine pumps that follow.

As an example, under an Air Force contract, we were operating at the high chamber pressure, and it was a question as to whether we would build and install large, heavy, pressurized feed tanks for the development of the combustion chambers, or build pumps. Our analysis showed that the pressure-fed system would cost us about $4 million, and would have only short-duration capability.

It was decided to use high-pressure pumps. The pump development cost us approximately $600,000. So, there is quite a saving in money. We also had the flexibility of longer duration with the high-pressure pump whereas in a pressurized feed system we did not.

Figure 19 depicts the method by which we would develop a large engine. When facilities today cost as much as they do, no single private contractor has the facilities actually to test and develop a 24-million-pound-thrust engine. Therefore, other means must be devised in order to do the maximum amount of work on an engine and then go to a government site to complete PFRT and Qualification testing.

First, we design and fabricate the components for a 2-million-pound-thrust engine module. We combine these and come up with a module, a segment of the complete engine. At this stage we make a decision. Are we going to use modular engines (plug or forced-deflection nozzle) or are we going to develop a 24-million-pound-thrust pump for use as a single pump with the modules clustered around it? Then, we take either one of these three configurations, and we go to a government facility and conduct tests to complete PFRT and Qualification testing. This is the method we propose to develop a large-sized engine.

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The development plan for the advanced engine is shown in figure 20. Here we see the advanced technology work that should be started now. I am not showing a chart on the modular clustered concept, but this typifies a particular plan. This plan is for a single-pump configuration.

The advanced technology work is performed on the 170,000-pound-thrust system discussed previously. We do tests on combustion development, heat transfer, performance, and so forth using this breadboard engine. Concurrently, we are developing a 2-million-pound-thrust pumping system. This is used in the demonstration engine. The demonstration engine is a single-pump, two-chamber engine having 2 million pounds of thrust. Integrated engine characteristics, as well as sea-level performance and thrust vector control gas injection characteristics, are demonstrated and verified with this demonstration engine. We also use the 2-million-pound-thrust pumping system for developing the 2-million-pound-thrust primary and secondary combustors that are used on the 24-million-pound-thrust engine. We have indicated here the approximate cost to do these particular programs; we have about 800 tests for full-scale combustors development to work out all the critical areas.

A portion of the development must be done off site at a government facility. We can do all component development at our facility in Sacramento. Complete TPA testing must be done off site. The engine development is off site. The total cost of this is $1,374,000,000.

One method which would allow development of the pump in-house is to use our present M-1 facilities; we can get 25 seconds duration on the hydrogen pump. (See fig. 21.) The gas generator supplies approximately half the required 3.2 million horsepower for the hydrogen pump system. We augment the horsepower by the hydraulically driven turbine in this configuration and get an extended duration by bootstrapping some of the hydrogen into the gas generator, since we are limited on tank capacity. On the oxidizer pump, since we also have a hydrogen-rich gas generator, we are limited to 5 seconds. If we develop an oxidizer-rich gas generator, we can get about 25 seconds. So, we can do the major portion of the pump development work in our Sacramento facility.

The development plan features are as follow:

(1) Advanced technology approach to solve critical problems
(2) Scale model testing
(3) Multiple combustor ring
(4) Use of pumps for suppling high pressure for thrust chamber testing rather than high-pressure tank facilities
(5) Pumping system test setup for big pump testing
(6) All component testing in Sacramento
(7) Overall program cost savings of 40 to 50 percent over a conventional development approach

We have already discussed the advanced technology that needs to be started. We would go into scale model testing to define and eliminate the critical areas of development. The use of multiple combustors permits a great many tests for a minimum cost. We can do the component testing in Sacramento. If we start advanced technology today with the scale-model testing, we estimate we can save from 40 to 50 percent of the overall program cost required by the conventional developmental approach.

The following table summarizes performance, costs, and time. We have talked about specific impulse. The forced-deflection cluster is the five 6-million-pound engines we showed. The plug is the modular plug configuration. Next are the clustered modules with forced deflection.

ENGINE SYSTEM COMPARISON

	F-D cluster	Plug	F-D modular	Single engine
Specific impulse, sec				
Sea level	385	354	385	383
Vacuum	448	414	448	450
Installed weight, lb[a]	361,000	358,000	423,000	312,000
Cost, millions of dollars				
Development	1,083	817	817	1,374
Production (200 systems) . . .	2,230	3,390	3,340	1,180
Reliability				
Qualification	0.991	0.991	0.991	0.992
Qualification + 5 years	0.996	0.996	0.996	0.997
Availability (year)				
PFRT	1970	1970	1970	1970
Qualification	1972	1972	1972	1973

[a]Includes: Engine, frame, suction lines.

Finally, there is the single-pump engine. Our studies show that this single-pump engine is a more reliable engine; it does not require engine-out. Engine-out requirements complicate the system tremendously. Note that the development cost of the single engine is the highest: $1.3 billion. But production costs for this engine, based on delivery of 200 systems, are lowest. Thus, on the basis of overall cost, weight, specific impulse, and reliability, the single-pump engine is best.

We are not only working on advanced high-pressure pump-fed engines, but we are also doing work on low-chamber-pressure-fed engines. Figure 22 shows a two-stage vehicle that can carry the payload indicated. We use in our configuration lox/RP-1 for the first stage and lox/hydrogen in the second stage.

We believe there ought to be advanced technology, particularly in the area of the combustion chamber itself. (See fig. 23.) When we are talking about a thrust of 20 million pounds in a single combustion chamber, we need to know more about performance and we need to know more about stability. This figure depicts a water-launch platform of two T-2 tankers welded together, in which development work can be done.

QUESTION PERIOD

MR. GINSBURG: Would you care to quote the weight of the turbopump that was a part of the 312,000-pound total engine system?

MR. GIBB: About 122,000 pounds.

MR. GINSBURG: Would you care to talk a little more about how you arrived at 2,500 psi as the optimum chamber pressure?

MR. STIFF: We made optimization studies based upon payload in orbit. These studies showed a slight increase in payload for higher chamber pressure, but the curves were flattening at chamber pressures of around 2,500 to 2,800 psia.

MR. BARTZ: Can you comment on your selection of propellants, which seems to be contrary to the Convair selection of lox/RP-1 for the first stage?

MR. STIFF: Our study shows that even with a bigger tank, performance is increased by using lox/hydrogen.

MR. BARTZ: There wasn't any question about the performance. Cost, I think, was their basis.

MR. STIFF: On the basis of the experience that we have had with Titan I and our lox/hydrogen work, we believe that lox/hydrogen is by an order of magnitude cheaper to develop. We have had very few cases of high-frequency instability with lox/hydrogen. Of course, these are two cryogenics, but there is also a cryogenic with the lox/RP-1.

MR. BARTZ: Then it boils down to a difference of how you look at the costs apparently.

MR. STIFF: No.

MR. BARTZ: Again your conclusions and Convair's?

MR. STIFF: Probably so.

MR. WILLIAMS: I would like to comment on that. First of all you mentioned, specifically, development costs. All the development that we have to date, or at least that we have been able to analyze, does not really verify the point that you made, that lox/hydrogen is so much cheaper to develop; that is, engines of this category are not necessarily cheaper to develop. We would go into numbers, if you like. The conclusions which have been drawn, specifically in the GD/A results, do not look solely at the development costs per se but the operational costs which, for a program of the size that we are talking about, are really the influencing parameters when you get down to cost. And here is where the RP really begins to take over in the operational cost area. The cost of propellants is one thing. The size of the vehicle which must be manufactured, checked out, transported, logistically supported, et cetera, adds to a degree to the advantage of the more compact lox/RP system. The hazard, the launch facility separation, the test program, and the separations required there are other important parameters that fit into the overall cost of the systems.

The costs of the propellants themselves actually contribute, particularly when you get to recoverable systems. Since we use the systems over and over, the expended propellants do become a contributing factor. In recoverable engines, the RP systems are more compact, do lend themselves more readily to recovery, and hence provide economical advantages there. Even if recovery is not considered for RP versus hydrogen in the first stage of the two-stage system, the RP still has a slight economical advantage. I mention economics quite a few times. It is not that we have concluded that economics should in all cases be the overriding criteria, but it is one yardstick which we are using in evaluating all the systems.

In fact, I would be interested in the comparison you made that can be strongly influenced, as we have found, by the choice of configurations: the one in the optimization of combustion chamber pressures, for example, whether to use a single-stage, a stage-and-a-half, or a two-stage system. This one simple choice there can have a rather influential effect on combustion chamber pressure optimization. What vehicle did you use in arriving at certain engine configuration selections of criteria?

MR. STIFF: Most of our studies to date have been based on a single-stage-to-orbit configuration. I think if single-stage-to-orbit is of importance - and we believe that it is, from an economical standpoint - I think you would immediately eliminate lox/RP for this type of operation, because it doesn't have the performance.

MR. WILLIAMS: This is correct.

MR. STIFF: This was a primary consideration in the choice of lox/hydrogen in our studies. Most of our work has been performed on the basis of a single-stage-to-orbit mission for which we optimized our system. We have looked at stage-and-a-half vehicles and some of them looked very interesting. We have been working with Martin-GD/A on various configurations.

MR. BEICHEL: In studies with Boeing and Martin-Marietta, we found that hydrogen was the most economical for the single-stage-to-orbit configuration. We also studied the two-stage vehicle; if the same velocity increments are taken, lox/hydrogen is cheaper. We optimized the hydrogen for the first stage. That is where the big cost differential comes in.

We must also look into the future growth potential that is still there with the higher performance of the hydrogen. We checked back again and again and we cannot find the justification for the RP-1 as a superior system because of cost.

MR. CONNORS: You prefaced your remarks by the observation that we could include the M-1 engine in an unconventional configuration. Would you care to amplify that?

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MR. STIFF: M-1, as you know, operates at a chamber pressure of 1,000 pounds, and it is of course bigger than the more advanced type engine at 2,500 pounds. My remarks were primarily to the effect that we could incorporate a forced-deflection type nozzle with the M-1 configuration. This would not be a staged-combustion cycle as we have in the advanced engine, but the ingredients are there. A forced-deflection nozzle can be placed on the combustion chamber for altitude compensation, which would increase the performance. The combustion chambers of the M-1 can be clustered around a forced-deflection nozzle.

MR. CONNORS: In listing problems in advanced technology, you didn't include a great deal of wind-tunnel work. When you are talking about other than sea load compensation, there are a lot of stream effects that can affect your results.

MR. STIFF: That's true. If I didn't make that clear, it was an error on my part.

MR. NELSON: I would like to ask something in connection with the argument of lox/hydrogen versus RP. It seems to me that you can't talk about the two in the same breath without varying the staging. RP can't get to orbit in one stage, period. Lox/hydrogen maybe can, but you might pay a penalty. It seems to me that you are overweighting your staging problem when you say you favor one-stage-to-orbit. Or are you considering recovery?

MR. STIFF: If you consider recovery, I think it is quite important to consider a single stage from an economical standpoint. Of course you can stage.

MR. NELSON: This, then, is the parameter. I don't consider the number of stages. I don't think we have had a severe problem in staging.

MR. STIFF: Have you been keeping up with Titan II?

MR. NELSON: It is in the development phase. We have a lot of problems.

MR. STIFF: Isn't it axiomatic that with more stages the inherent reliability is lower?

MR. NELSON: In general, yes. I think the point here is that maybe the staging parameter is not the primary one, but maybe the recovery factor. If you assume you have to recover, maybe this is the predominant parameter, rather than the number of stages. When you argue RP versus lox/hydrogen in the first stage, it is where are you going to recover, not whether it is one stage or two stages. You get up to three or four stages, yes. But between one and two stages I don't think there is much of an argument.

MR. STIFF: I think there is more cost associated with the two-stage or three-stage system. I don't think you can get around that. If you can do the job with a single stage I think you should. There are tremendous costs associated with each stage you have to check out in the system. I suspect when you check out the Saturn right now, it is a tremendous job.

MR. NELSON: Multiple staging makes the system more compact. If you compare Titan III with the three-stage Saturn, it is very spectacular, the compactness of the Titan III. Here you have a four-stage system versus three-stage. If you are going to recover I think you have an argument for one-stage-to-orbit. If you are not going to recover, I don't think you have an argument.

MR. STIFF: I think we do.

MR. NELSON: I am giving another point.

MR. STIFF: In our studies of large boosters and the costs, considering the amount of money that we have already spent, if we don't include recovery, is an error in our analysis.

MR. NELSON: This is right. Compactness, I think, is another factor maybe of importance, and introduces whether you should stick with lox/hydrogen or really get into a new high-density, high-energy propellant.

I don't think we have defined the ground rules to discuss whether you put lox/hydrogen, RP, or some other propellant in the first stage. I think we have to define the parameters. You haven't a discussion until somebody in the meeting sets down the criteria we are going to argue to. That is one point.

MR. WEIDNER: As was indicated, we have in the past worked with GE and Martin in a Nova-type study. One of the ground rules of the Nova study certainly was reusability and recovery. I think that was one of the strong desires there. Without mentioning it here, this ground rule is one of the accepted ones or adopted ones of the Nova.

MR. WILLIAMS: This is correct. Cost has been a major yardstick that we have used in all of our studies, when we compare RP versus hydrogen in the first stage of a two-stage vehicle. We optimized each system from cost standpoint. When we compared stage-and-a-half with two-stage and with single-stage, whether it be recoverable or expendable, again we optimized from a systems cost effectiveness standpoint. And the results were on the basis of this optimization to the best of our ability of all systems on a consistent set of ground rules.

MR. MORRELL: I am a little intrigued by your breadboard scaling procedure, where you are going to go up by a factor of 10 in one step, from 170,000 to 2 million. Did I understand you properly there?

MR. STIFF: If we go the single-pump route, which is the way we believe we should go, a single pump for 24 million pounds, we are going from 170,000 to 2 million pounds. Keep in mind, though, that we are already developing a 1.5-million-pound system pump right now.

MR. MORRELL: I am not concerned about the pump because I don't know much about the pumps. Maybe you can do it. I am thinking in terms of scaling either your dual combustion system or single combustion system, both for performance and stability, by a factor of 10. I am not sure I see how you do this. I am sure that you are going to have a reasonable development program.

MR. STIFF: You might visualize the module for clustering as a vacuum-cleaner-shaped system. The primary reason for that particular system is to use a pumping system instead of a high-pressure feed system, to develop seals and bearings, to get hydraulic characteristics for the pump and so forth, and also to develop the combustor itself. This permits advanced technology work at a small enough scale, but not so small that it doesn't mean anything. We believe 170,000 to 200,000 pounds is a reasonable size. Of course then you have to go from there up to your 2-million-pound thrust size. Each one of the primary-secondary combustors has to be able to handle a thrust of 2 million pounds in a 24-million-pound engine. You have the normal problems of going from a thrust of 170,000 up to 2 million pounds.

MR. MORRELL: They are hardly normal. I think there is one thing in your favor, as long as you are sticking with hydrogen/oxygen; I agree you probably have less problems with dynamic instability, except perhaps in the feed system type of instability. I would think that the scaling methods for maintaining performance and dynamic stability are completely different. I don't see how you reconcile jumping from 170,000 to 2 million. You really have a problem on your hands if you are going to maintain both similarities.

MR. STIFF: I agree with you that you can't maintain the similarities exactly. I think rocket development today indicates that the larger the size, the more problems encountered, especially in high-frequency instability.

MR. MORRELL: And low frequency.

MR. STIFF: I am not too concerned with low frequency, especially with a high-pressure engine, if there is sufficient discharge pressure and a hard system is maintained. I am not too particularly concerned with the low frequency, but rather with the high frequency.

MR. MORRELL: I think the point is that you are getting up to sizes where the two modes are now able to interact. You can't interact. You can't decouple. Your acoustic modes are almost the same frequency as your hydrodynamic modes. So you have multiple problems of coupling. I think you won't have these at the 170,000 level.

MR. PAUL: You can't solve all the problems by your scheme but you can solve a number of them. And you can study a lot of starting techniques and so forth; it should be a great help if you proceed the way you propose.

MR. STIFF: I don't pretend we are going to solve all problems at a scale of 170,000 pounds.

MR. MORRELL: I don't see what you are going to solve at all. The dynamics are completely different.

MR. STIFF: We will solve a lot of problems. We can get heat-transfer data.

MR. MORRELL: You can get heat-transfer data on a heated tube.

MR. STIFF: I disagree.

MR. MORRELL: You can if you want to scale enough. The point is that the large systems you are eventually going to build will be completely different dynamically from the breadboard kind of thing you will build.

MR. STIFF: There will be some difference. I won't say completely different. The cycle is the same.

MR. MORRELL: Using a bunsen burner in a tube is the same cycle as a rocket engine.

QUESTION: What thrust level and validity do you hold in the tests that you made in your Air Force Contract, in which you compare the conventional cycle with the two-stage cycle? I believe you showed considerable evidence of a lot better combustion, indicating better stability. In other words, how much credence do you give to this, and do you think this could be extrapolated to larger sizes because of the inherent nature of the cycle itself being gas instead of liquid?

MR. STIFF: I am kind of getting into trouble here by talking about scaling. With this system, we have a gas; essentially the fuel is a gas going into the combustion chamber, and the oxidizer is, of course, the liquid. In our experience to date we have had very, very good stability with this system, and we have tested over a large number of different injector configurations. It appears to us that this is a superior method of injecting propellants into the combustion chamber to give stability. Whether you can scale this thing up to 2 million pounds remains to be seen.

DECLASSIFIED
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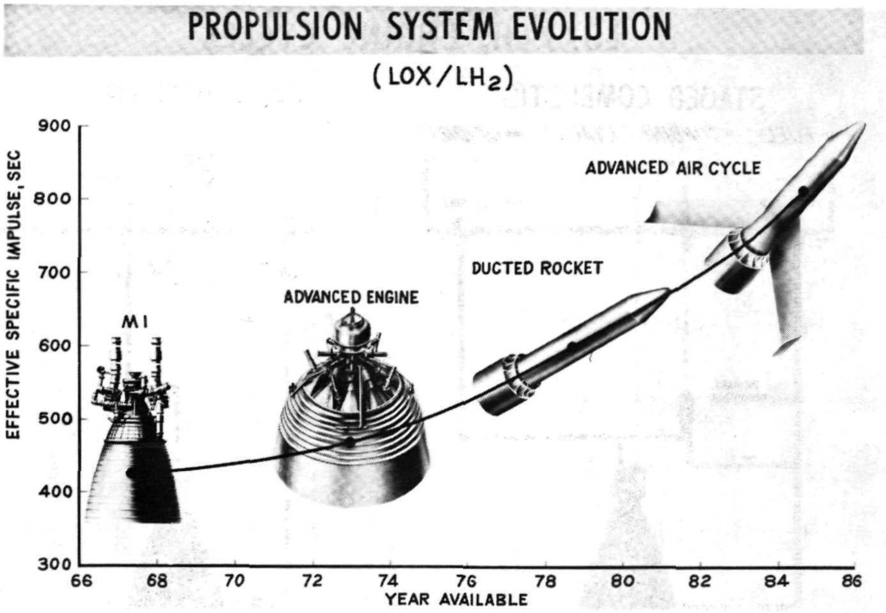

Figure 1

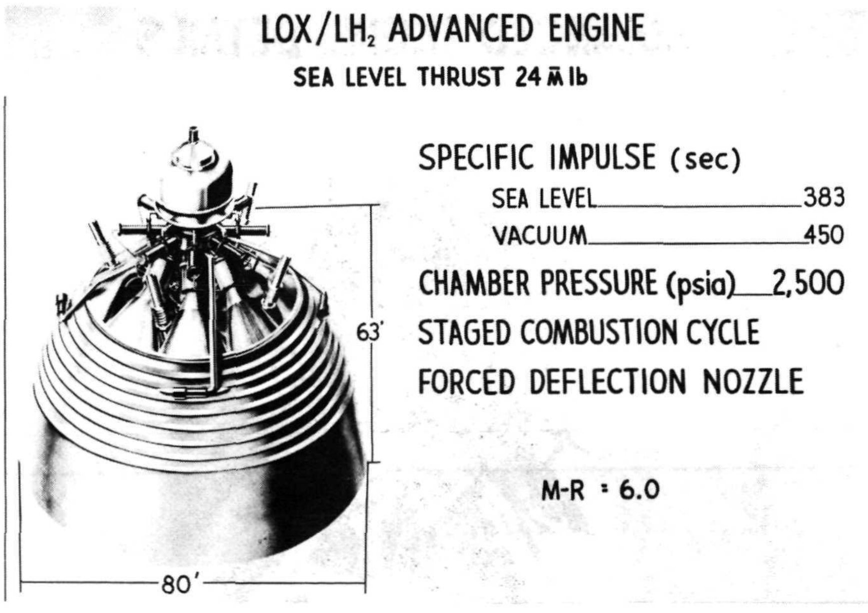

Figure 2

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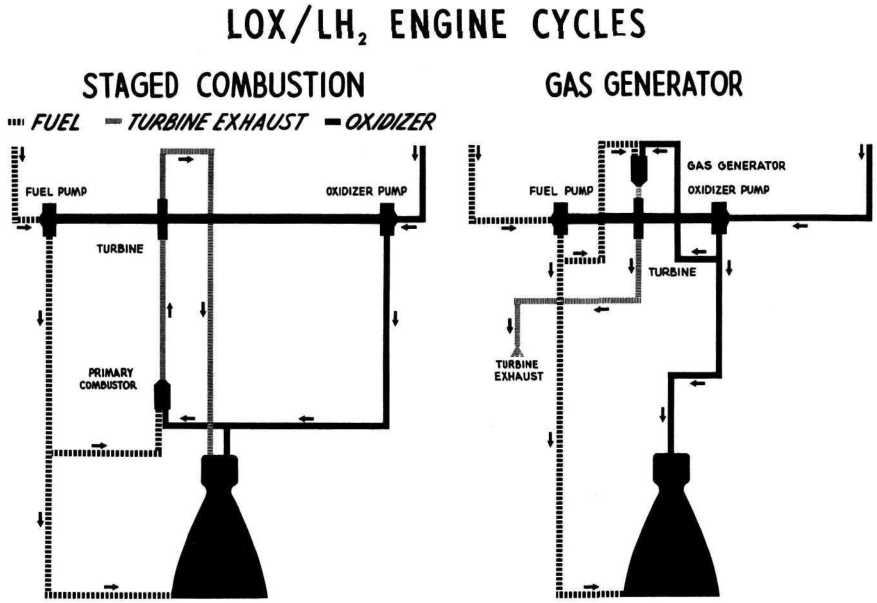

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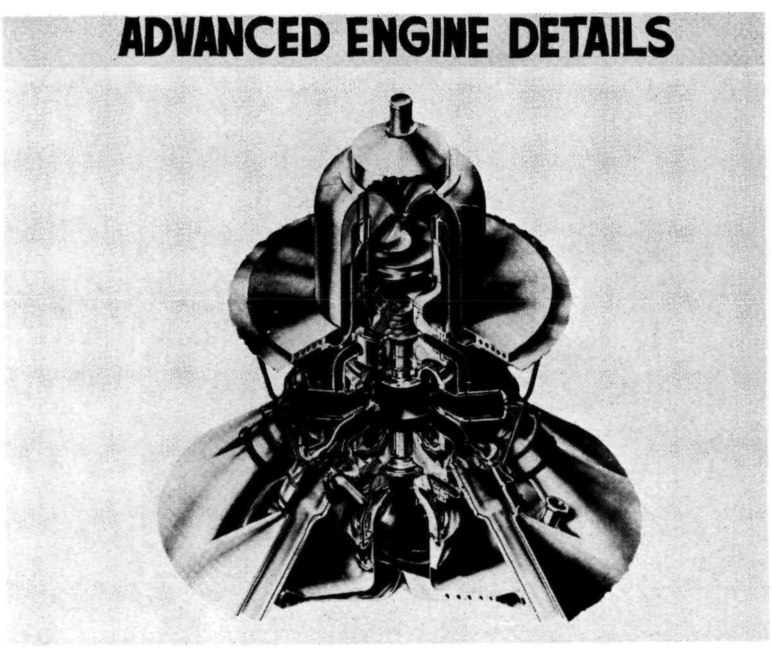

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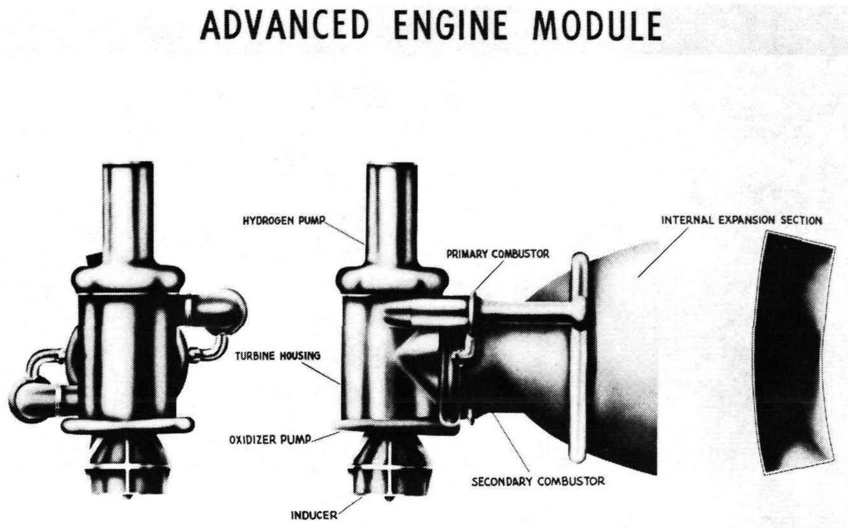

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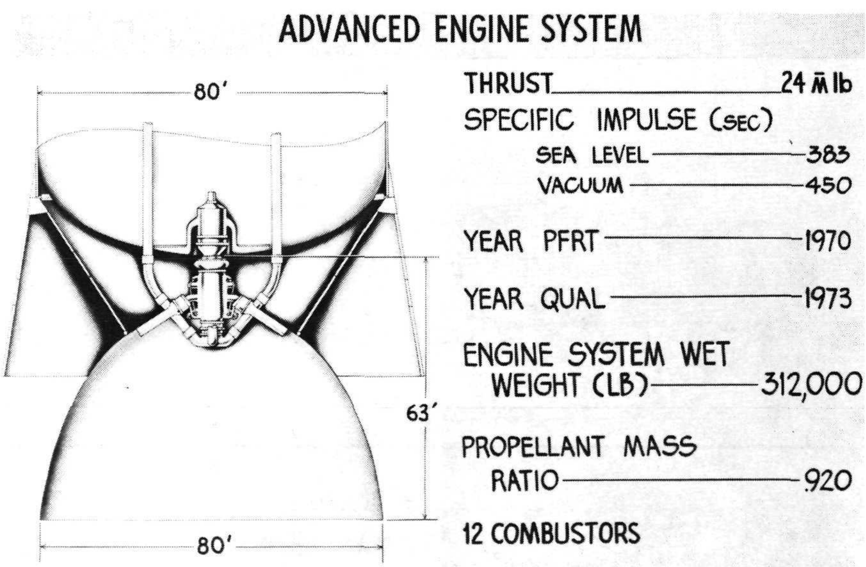

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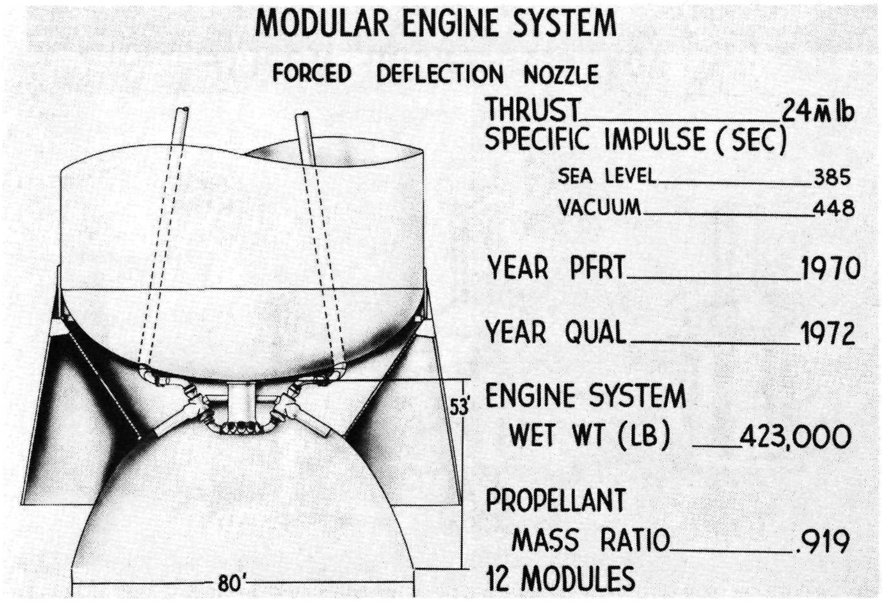

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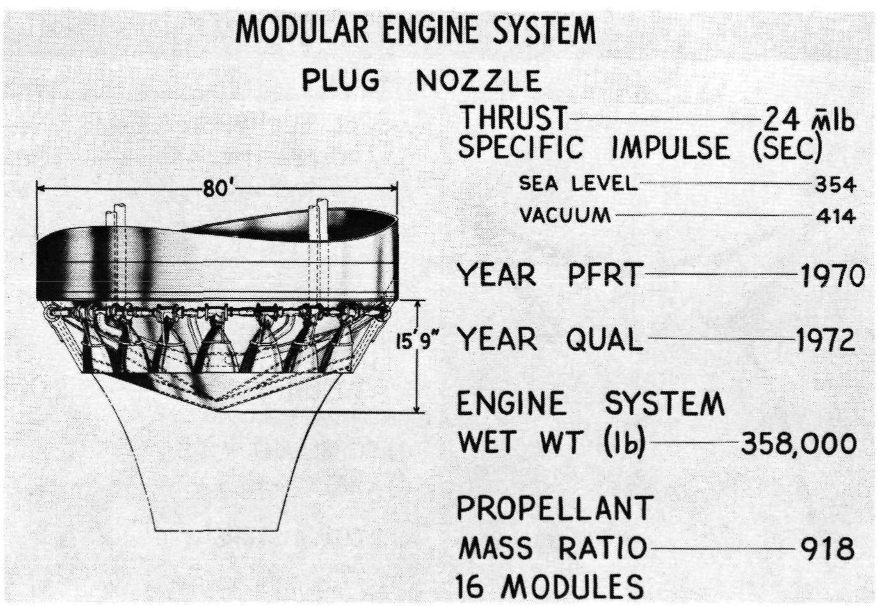

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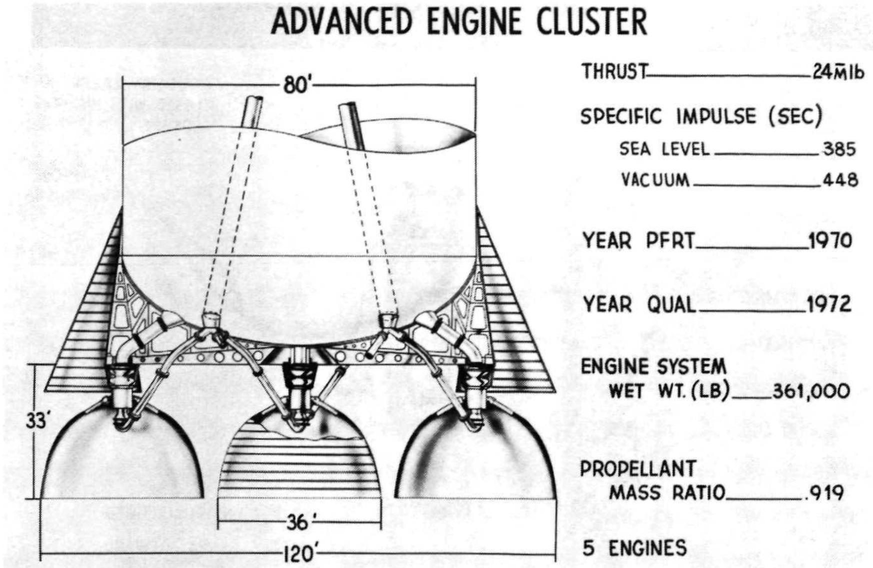

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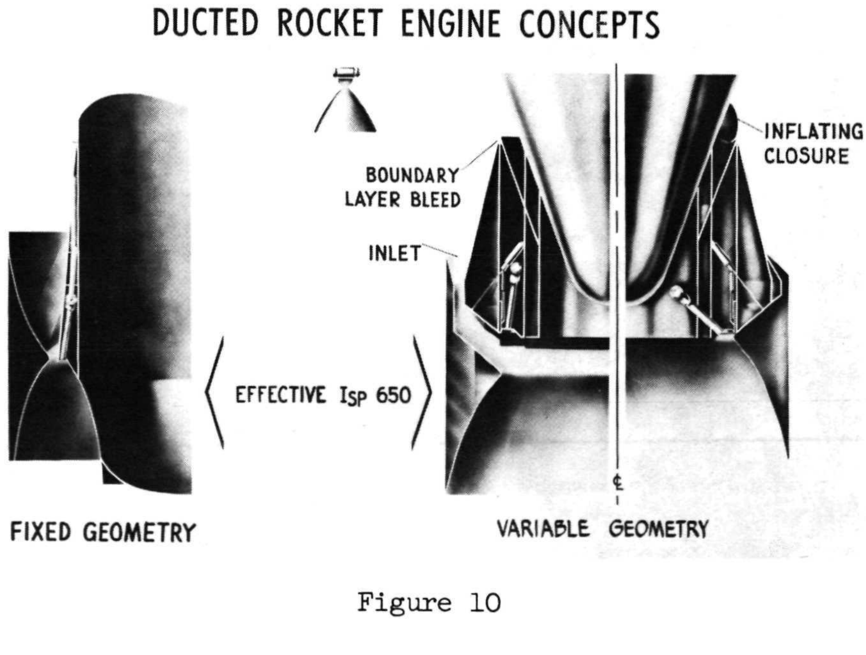

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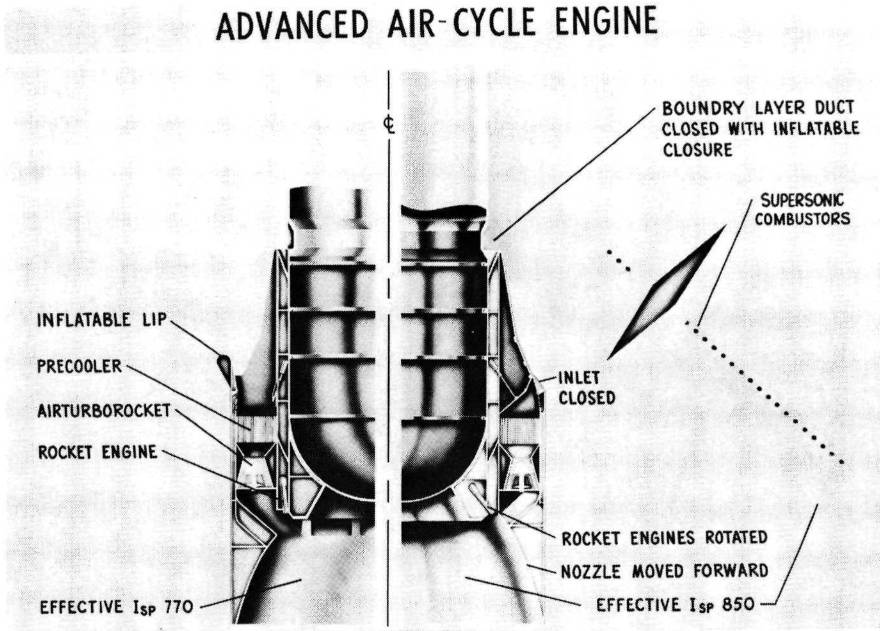

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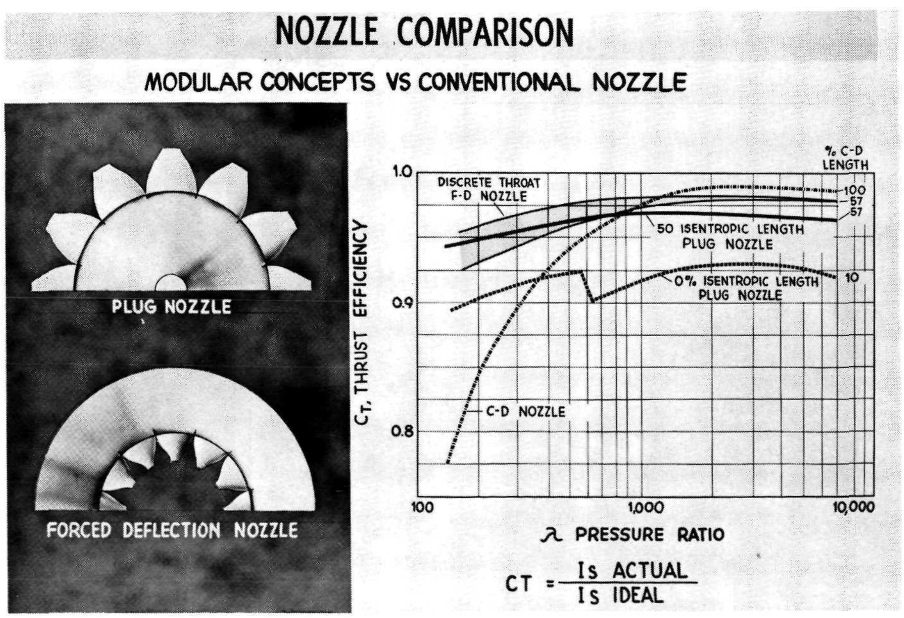

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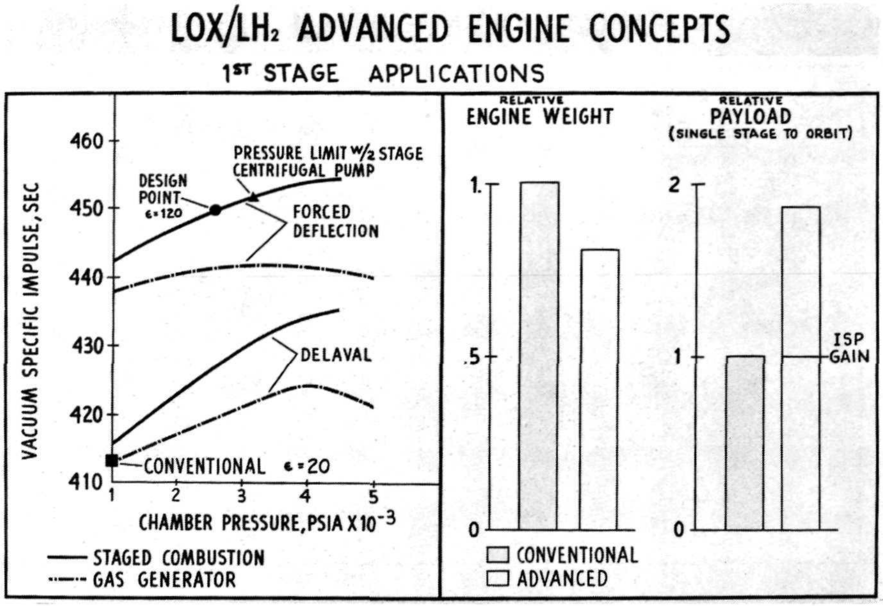

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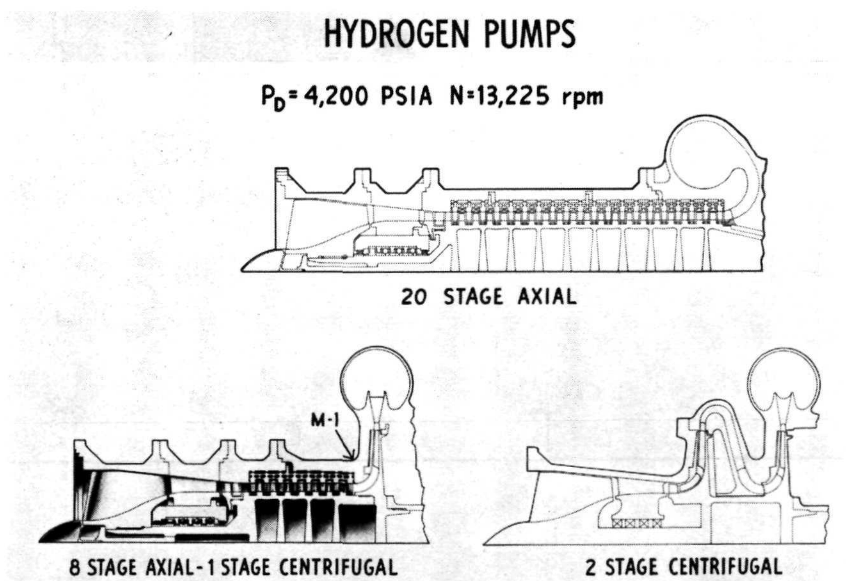


Figure 14

ADVANCED ENGINE INSTALLATIONS

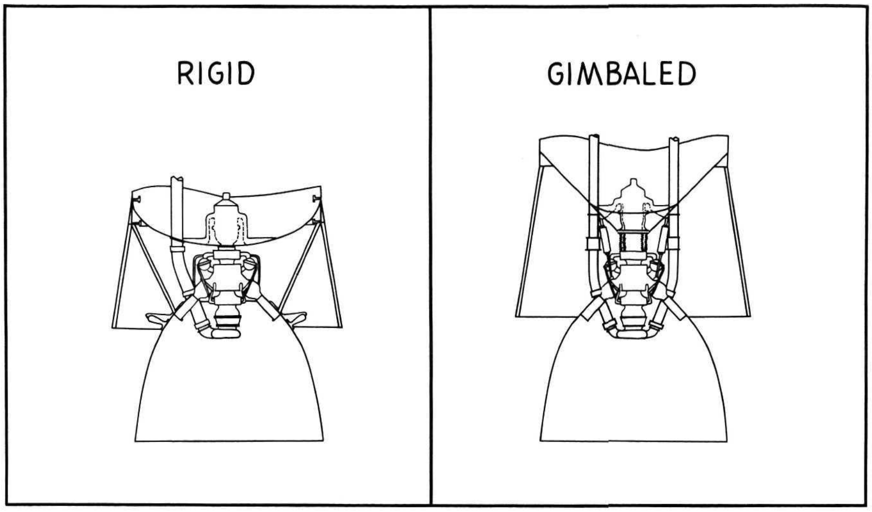
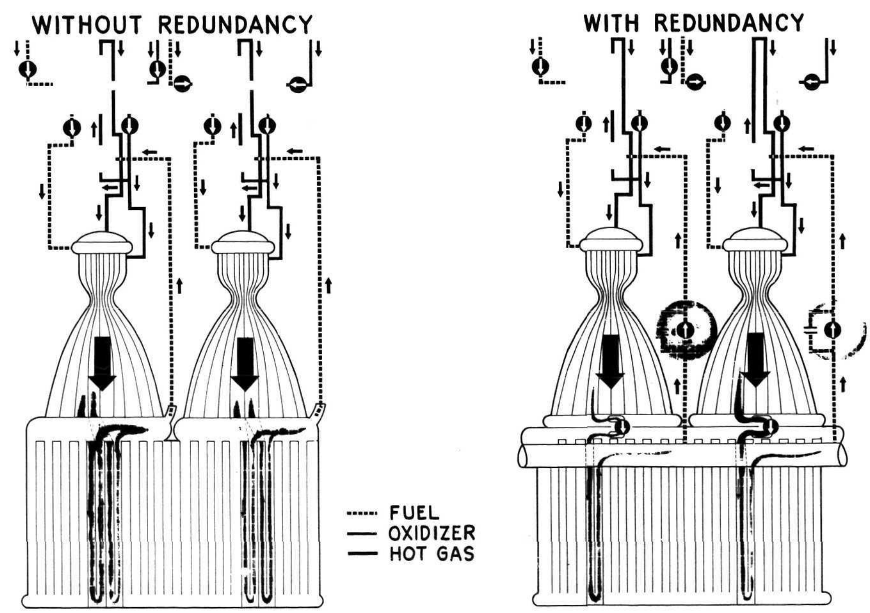


Figure 15

MULTIPLE ENGINE INSTALLATIONS



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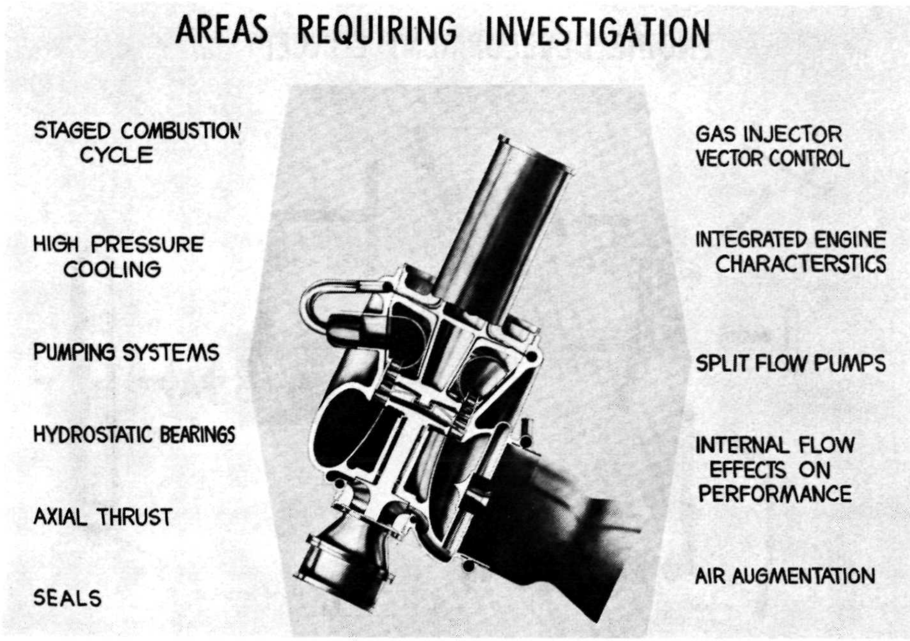

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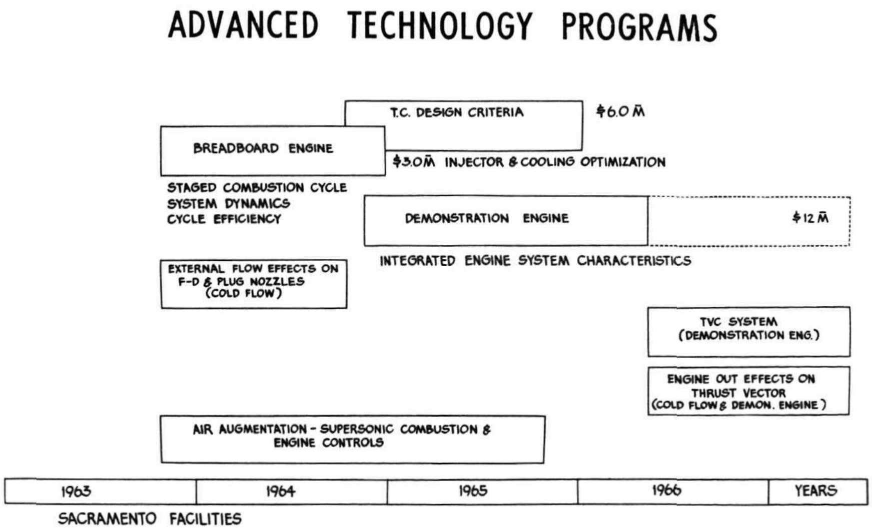

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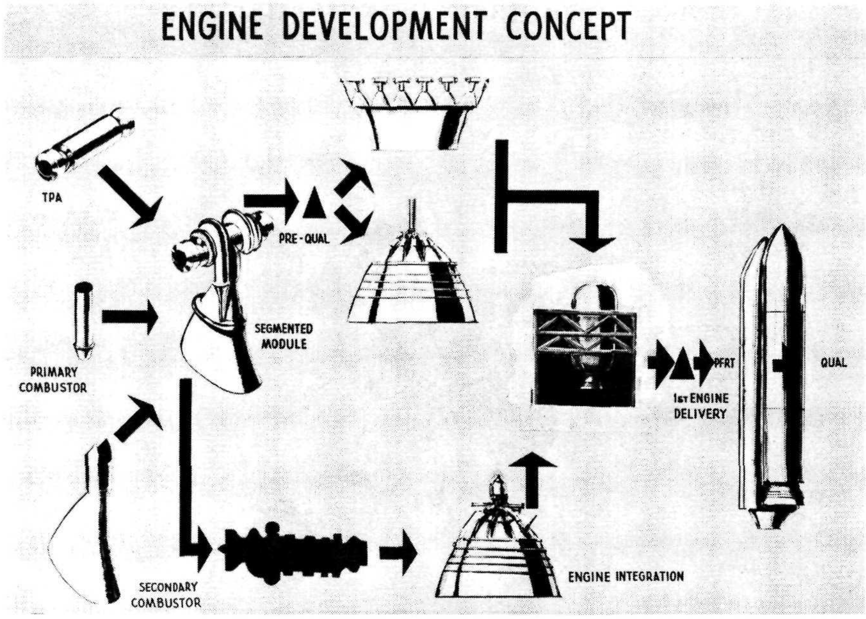

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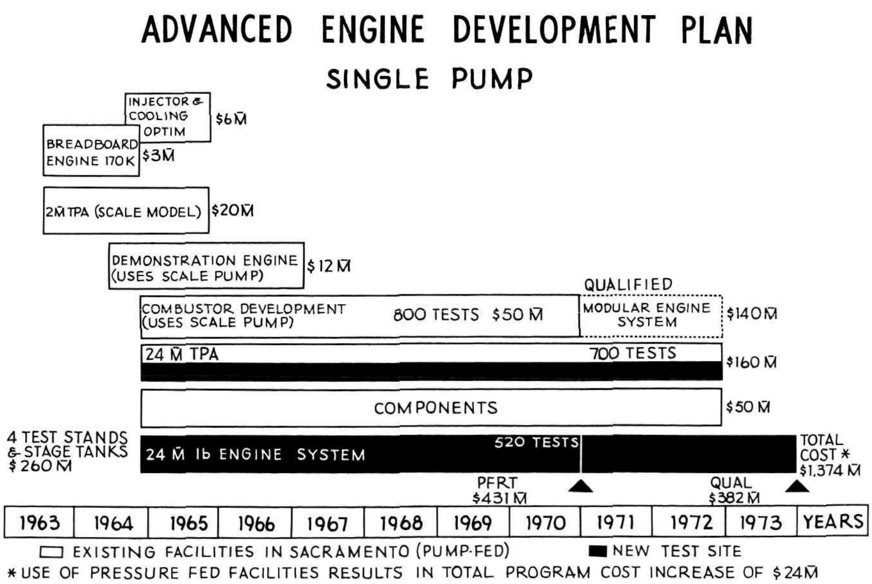

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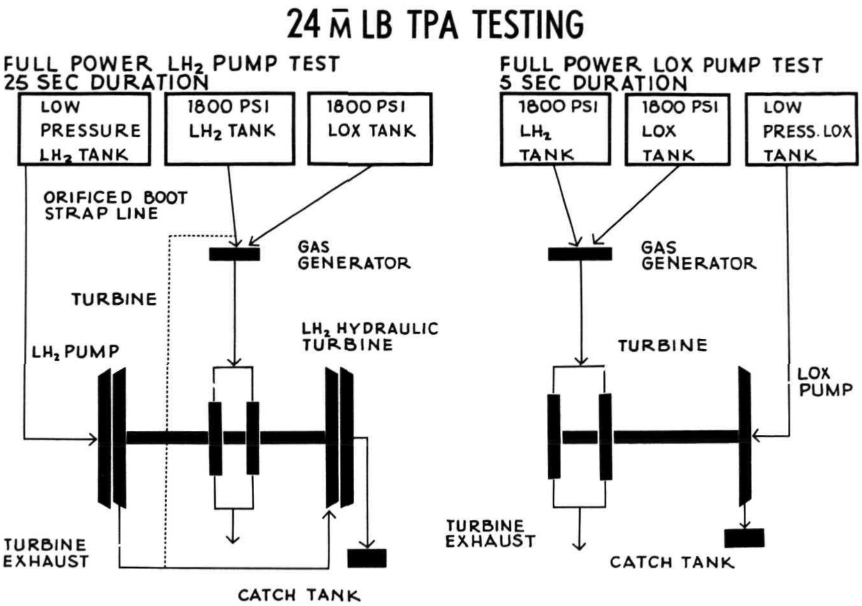
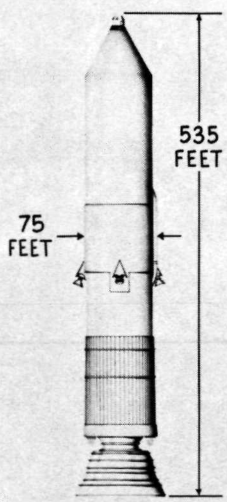


Figure 21

PRESSURE FED BOOST VEHICLE

LOW CHAMBER PRESSURE ENGINE

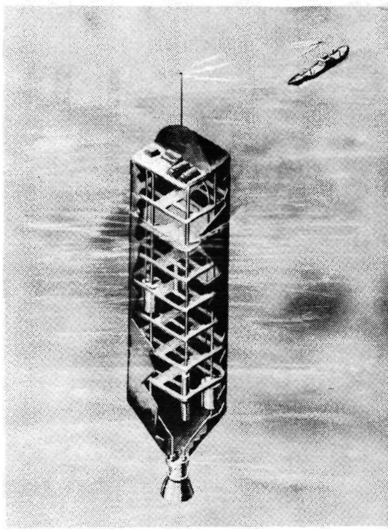


PAYLOAD	1.727 MILLION LBS 225 KM ORBIT	
LAUNCH WEIGHT	40.23 MILLION LBS	
	STAGE I	STAGE II
LAUNCH WEIGHT	27.557 MIL LBS	10.973 MIL LBS
THRUST	80 MIL LBS(S.L.)	11.3 MIL LBS(VAC)
PROPELLANTS	LO_2 /RP-1	LO_2 / LH_2
CHAMBER PRESS.	300 PSIA	60 PSIA
NOZZLE AREA RATIO	5	23
MASS FRACTION	0.915	0.905

Figure 22

LARGE PRESSURE FED ENGINES

ADVANCED TECHNOLOGY



OBJECTIVES

DEVELOP 20 MILLION LB THRUST TEST CAPABILITY

- DETERMINE COMBUSTION STABILITY, INJECTOR PERFORMANCE, START & HEAT TRANSFER CHARACTERISTICS OF LARGE THRUST, LIQUID PROPELLANT, WATER LAUNCHED SYSTEMS

LARGE ENGINE SCALING FEASIBILITY DEMONSTRATION

PHASE I

- DESIGN, FABRICATE, TEST FLOAT & STATIC FIRE FULL SCALE 20M LB THRUST WORKHORSE CHAMBER

▲18 MONTHS ▲$5.9 MILLION

PHASE II

- FULLSCALE-HOT FIRING TEST PROGRAM-5 INJECTOR DESIGNS, TWO THRUST LEVELS, 30 SHORT DURATION FIRINGS

▲12 MONTHS ▲$1.4 MILLION

Figure 23