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PROCEEDINGS OF CONFERENCE ON LARGE LIQUID ROCKET SYSTEMS JULY 9-10, 1963

[redacted]

NASA Headquarters
Washington, D.C.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JUNE 1964

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PROCEEDINGS OF

CONFERENCE ON LARGE LIQUID ROCKET SYSTEMS

JULY 9-10, 1963

NASA Headquarters
Washington, D.C.

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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CONFERENCE ON LARGE LIQUID ROCKET ENGINES

Main Auditorium, Room F62012 (FOB 6)
NASA Headquarters

July 9-10, 1963

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OPENING REMARKS

By John L. Sloop

Director, Propulsion and Power Generation,
Office of Advanced Research and Technology, NASA

There has been considerable NASA interest in large liquid rocket engines. In 1961 design studies were undertaken by NASA under the direction of Henry Burlage at Rocketdyne and Aerojet on contracts NAS-5-1025 and NAS-5-1026. This has led to a series of technology programs which we hope will be summarized and discussed in this meeting. In addition I am sure you are familiar with the studies at Marshall Space Flight Center on Nova and post-Nova. These have shown the needs for more advanced propulsion concepts.

The purpose of our meeting is to present and critically discuss this available information, to identify major problem areas, and to point out specific problems where in-house and contract effort is needed. The first two sessions will contain presentations by contractor personnel. After these two sessions this will be an all-government meeting for planning purposes and for discussion.

We have divided the discussions into several sessions. These will be headed and run by the chairman for each session. I will act as general chairman.

At the end of the meeting the chairman of each of these sessions will give a résumé of the major problems where in-house and contract effort is needed. I expect that these chairmen will summarize their own particular session. After all the papers are presented the chairmen will make general comments and the floor will be open for individual comments.

SESSION I

PROPULSION REQUIREMENTS FROM A VEHICLE VIEWPOINT

Chairman: Francis Williams,
Future Projects Office,
Marshall Space Flight Center, NASA

1. INTRODUCTION

By Francis Williams

Future Projects Office
Marshall Space Flight Center, NASA

Before we consider requirements from a vehicle standpoint, perhaps we should first define the vehicle.

Nova is the name which has been attached to the propulsion requirements that we are to discuss, so we might start out by defining Nova.

There are two definitions that come to mind right at first. The first one, which I will quote from the dictionary of Rockets and Guided Missiles, goes like this:

> "Nova, appears suddenly at unpredicted times and places, sometimes becoming the brightest object for a few days. Then they fade and pass away."

There is another definition that we, working in the area of Nova, prefer: that is:

> "Nova, the next large launch vehicle after Saturn V."

I think you can see why we prefer the latter definition. We sometimes believe, looking at the past of Nova, some of the exercises we have gone through, that it flares up and does fade away. We hope that the present flare will be a continuous one and will maintain itself for a long time to come.

What are the requirements for Nova? These are not well established to date. We have, as part of the studies that we are undertaking, a market analysis, if you will, or assessment of what future and more ambitious missions might require Nova. The man-Mars exploration mission is one of the predominant ones, and a rather demanding mission. Other missions which might require Nova are: a lunar base; various orbital missions, both civilian and military types; very deep space probes; and scientific missions that have very high velocity requirements and require vehicles even bigger than Saturn V. There is also the possibility of such things as global logistics which may fit into the picture in the Nova time frame.

All these missions are being assessed by the groups that are working on and worrying about Nova, not only just trying to assess the market but to investigate each of these mission requirements or, if you will, "desirements," to determine their effect on the basic launch vehicle system. When we say "system" in our discussions, system is not just a single-stage or two-stage launch vehicle. This involves the overall facilities, logistics, the training, the whole program.

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Next we might ask the question: What does Nova look like? This also is not well established to date. We have numerous designs and configurations under consideration. Some of these and some of the trade-offs will be discussed subsequently.

We have various operational considerations that we are investigating - and of course several propulsion system concepts that we are trying to weigh, one against the other, to find out which is the best. We have several major studies underway at the present time. We have parallel or two systems studies that are being conducted on a parallel basis, one with General Dynamics/Astronautics, headed by Andrew Kalitinsky, and the other with Martin Marietta, Baltimore, headed by John Youngquist.

In support of these two systems studies we have three separate studies. All three of these happen to be at Martin Marietta's Baltimore division. One is launch operations and facilities; the other, test operation and facilities; and the third, transportation and logistics.

There are numerous other studies being done in support of advanced technology that have application in the Nova area. Many of these will be discussed in subsequent presentations.

I might also mention that in support of the basic Nova systems studies we have obtained a considerable wealth of data from the three engine contractors who will give presentations at this conference. Unfortunately, for them at least, it is not funded support at this time. They have, I think, very cooperatively supported both of the system study efforts to date and I think we have a very good working relationship there.

The Nova systems work is directed by a Nova management team. This Nova management team is made up of personnel from various organizations within NASA: NASA Headquarters, Marshall Space Flight Center, Manned Spacecraft Center, and Lewis Research Center. We hope in the very near future to get the other NASA organizations involved. We also have as members of the Nova management team, the Air Force. Howard Barfield is the senior Air Force representative.

I would like as head of the Nova management team to solicit your support and comments or criticisms on the Nova study. We have several published documents which have been distributed throughout NASA. If you do not have copies of these, they can be made available to you. We will have another major review in a couple of months. You are cordially invited to participate in these reviews.

What we are attempting to do, very briefly, with the Nova studies is to define the most desirable launch vehicle to succeed Saturn V. We hope, in the following presentations by contractors, that we will somewhat set the stage for this two-day session which is addressing itself to propulsion requirements for Nova-class vehicles. We plan to present some of the results, some of the very latest data that are available, some of the trends that are coming out of these studies, and some conclusions that we feel can be drawn at this time.

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2. PROPULSION SYSTEM REQUIREMENTS WITH RESPECT TO VEHICLE SYSTEMS

By John Wamser

General Dynamics/Astronautics
General Dynamics Corporation

This paper is concerned with a discussion of a portion of our Nova Part II study effort. I will discuss recovery and reuse and the selection of first- and second-stage propellants. We have reexamined the use of hydrogen versus RP as the first-stage propellants, and we have looked at the use of fluorine as the second-stage propellant. I will also discuss the use of forced deflection nozzles in the first- and second-stage applications and present some data on the thrust vector control requirements for these vehicles.

Figure 1 shows three typical Nova vehicle systems: A class II stage-and-a-half vehicle and a two-stage vehicle using advanced propulsion, and a class III single-stage vehicle also using advanced propulsion. In the case of the class III single-stage vehicle, the design is also based on some assumed improvements in the state of the art in materials and construction methods.

From the results of the study, it appears that all stages should be recoverable and reusable and that long lifetime and reusability are very important criteria for the propulsion systems. It also appears that the stage-and-a-half vehicle is somewhat less efficient than the two-stage vehicle.

In all cases the recoverable stage design is based on the Nexus concept, and needs a low W/C_DA to minimize heating during reentry.

In the case of orbital recovery an ablation material is required to prevent the tank from reentry heating damage. This is not the case necessarily in suborbital recovery or suborbital reentry.

An attitude control system is required to provide the proper attitude for the orbit and reentry. At subsonic speeds a parachute system is deployed to reduce the velocity and just prior to impact retrorockets are fired to reduce the velocity still further to something less than 30 feet per second.

In the case of the single stage, the design is somewhat different in that the stage is designed to enter from a rearward direction. This has a number of advantages. It provides more readily a better c.g.-c.p. relationship and gives more freedom in stage design. The heat protection system for the plug, for the propulsion system, also doubles as the heat protection system for reentry. In addition, we feel that this particular configuration with its plug arrangement is compatible with some of the air augmentation systems which we are considering.

The first comparison I would like to make is shown in figure 2. Here we have the use of hydrogen versus RP as the first-stage propellants. The vehicle on the left has lox/hydrogen in both stages. The one on the right has lox/RP in the first stage. In both cases the second stage is recoverable. Expansion deflection engines are used throughout.

The height of the RP stage is considerably less than the hydrogen stage by about 56 feet. Half of this is due to the use of RP. The other half is due to the fact that the area ratios on the vehicle on the right were selected at 40; the vehicle on the left at 100. However, 95 percent of ideal altitude compensation was assumed in both cases and for this condition it turns out that the propulsion performance is essentially the same, so this does not really have any effect on the comparison except that it makes the vehicle on the right still smaller.

The diameter of the RP vehicle is 100 compared with 134. This results in a lower launch drag and somewhat simplifies the manufacturing transport problems. Most important, because of the smaller size, it greatly reduces the cost of the first stage. The payload ratio, 18.8 compared with 14.4, is not as good for the RP vehicle. However, most of this weight is propellant itself, the RP. The total cost effectiveness is $91 per pound in the 225-kilometer orbit compared with $102. So we see that the RP vehicle is more efficient. The recurring cost effectiveness is $51.50 compared with $58. This includes the vehicle cost, propellant cost, and operational cost.

There are many other considerations in selecting a first-stage propellant. One of the more important here is the fact that RP is noncryogenic and has certain operational and logistic advantages. There are other advantages having to do with safety and so on.

We have made comparisons of quite a number of vehicles with different propulsion systems in the first stage and different propellants. These results are characteristic. The relationships hold. On the basis of this it would be our conclusion that RP would be the preferred first-stage propellant in a two-stage vehicle.

The next comparison is the use of fluorine versus oxygen as the second-stage oxidizer. (See fig. 3.)

Both vehicles use, in this case, RP. Lox/RP is the first-stage propellant. The second stages are again recoverable. The vehicle on the right has a fluorine/hydrogen second stage. The vehicle on the left has a lox/hydrogen second stage. What we are trading off here is the higher specific impulse and higher density of the fluorine at its higher cost. The fluorine vehicle is smaller in length, somewhat smaller in diameter, and has an improved payload ratio of 16.8 compared with 18.8. However, its cost effectiveness is not so good, $91 per pound compared with $107, and $51.5 compared with $60.

These relationships we have also examined in other configurations and these are characteristic.

Figure 4 shows a comparison of using forced deflection nozzles versus bell nozzles in the second stage of a two-stage vehicle. The vehicle on the left has two M-1 engines in the second stage; the one on the right has two expansion deflection engines in the second stage. The M-1 has a chamber pressure of 300 psi whereas the expansion deflection engines have a chamber pressure of 3,000 psi and an expansion ratio of 150. The length of the ED vehicle is 322 feet compared with 332 feet for the M-1 vehicle. The diameter is the same. The payload ratio is improved for the vehicle on the right. The total cost effectiveness is greatly improved, \$102 compared with \$110, as is the recurring cost effectiveness, \$58 versus \$65. This improvement is due almost entirely to the fact that the high area ratio was used with the 3,000-lb/sq in. chamber pressure ED engines.

There is a secondary effect in that the vehicle itself is somewhat shorter since the interstage adaptor is shorter. What we are seeing here is the effect of the area ratio in the second stage.

We have examined the same comparison, a similar comparison in the first stage and, assuming that we would have about 95 percent of ideal altitude compensation with the forced deflection engine, there is very little performance gain in the first-stage application. However, the forced deflection engine would be much smaller and we would then still have an improvement in vehicle size and facilities and so on.

Our conclusion here is that we would recommend forced deflection nozzles in both stages. I should say that, although we made this comparison here with expansion deflection engines, the same thing would hold true for any forced deflection engine, plug, or other configuration.

Figure 5 shows some typical thrust vector control requirements for the type of vehicles we have been considering. Shown here is the effective gimbal angle and control moments for a number of vehicles. Vehicles B, E, F, and H are cylindrical vehicles from the part I study. The vehicle represented on the right is a class II two-stage vehicle with a high drag shape. The effective gimbal angle for vehicle B is just a little greater than 2°. Shown here is the increment added to this for engine-out. This happens to be a 16 F-1 first-stage configuration and was designed for engine-out.

Vehicle E has the solid first stage. The large increment here is due to the staging increment. In this case the vehicle was parallel staged and this results in a large gimbal angle requirement.

Vehicle H is a stage-and-a-half vehicle and shows the difference between lox forward and lox aft on gimbal angle.

For vehicle 2-B-1, it is obvious that the gimbal angle is quite a sensitive function of the nose radius. Each nose radius here has associated with it a particular flare angle. The 120-foot nose radius would have a small flare angle, the 60 foot would have a large flare angle. We have selected for design in this case a nose flare angle of about 28°. We see that the gimbal angle requirements for this vehicle are quite similar to those of the other vehicles.

From these comparisons we would conclude that both stages of the two-stage vehicle should be reusable and that the first stage preferred propellant would be lox/RP, the second stage lox/hydrogen, and that forced deflection nozzles

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have a significant payoff in both stages, the first stage because of the height advantage and the second stage because of the advantage that comes from the high area ratio with its high rise.

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NOVA VEHICLE SYSTEMS

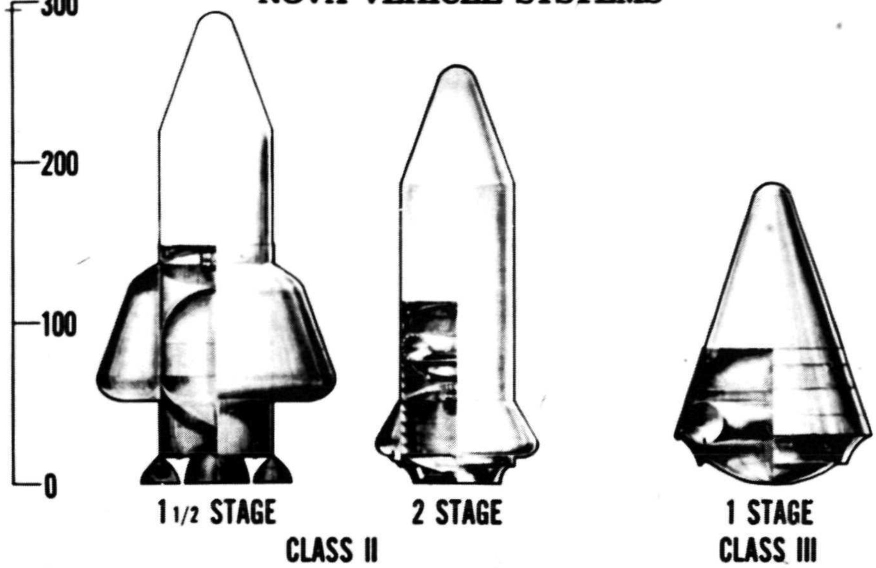

Figure 1

H_2 vs RP-1
FIRST-STAGE PROPELLANT

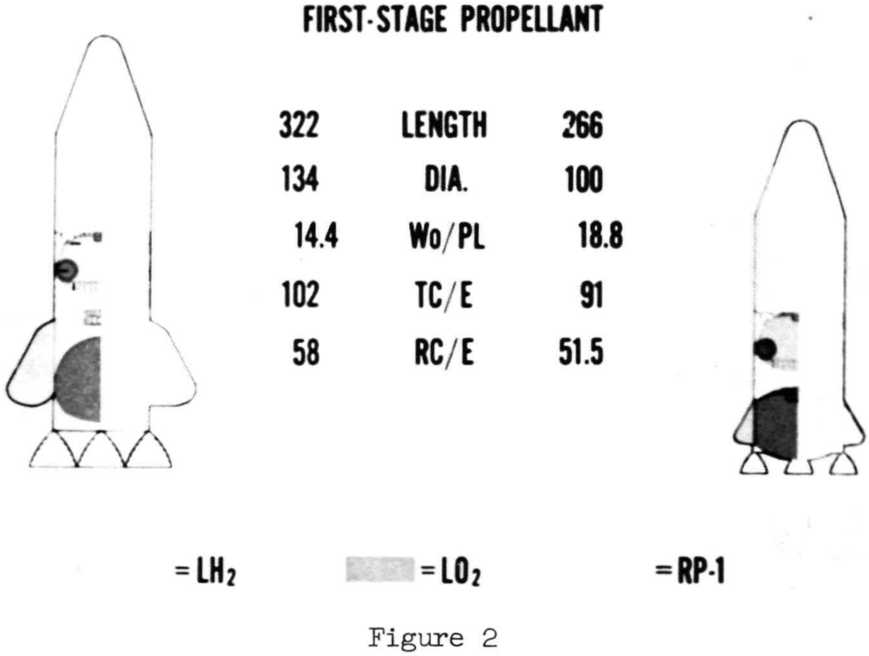

Figure 2

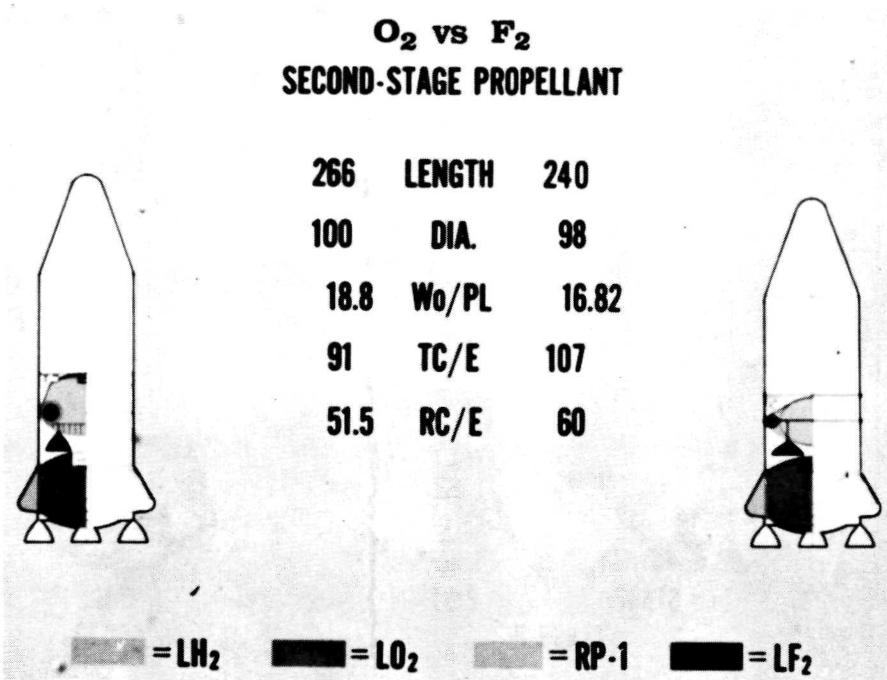

Figure 3

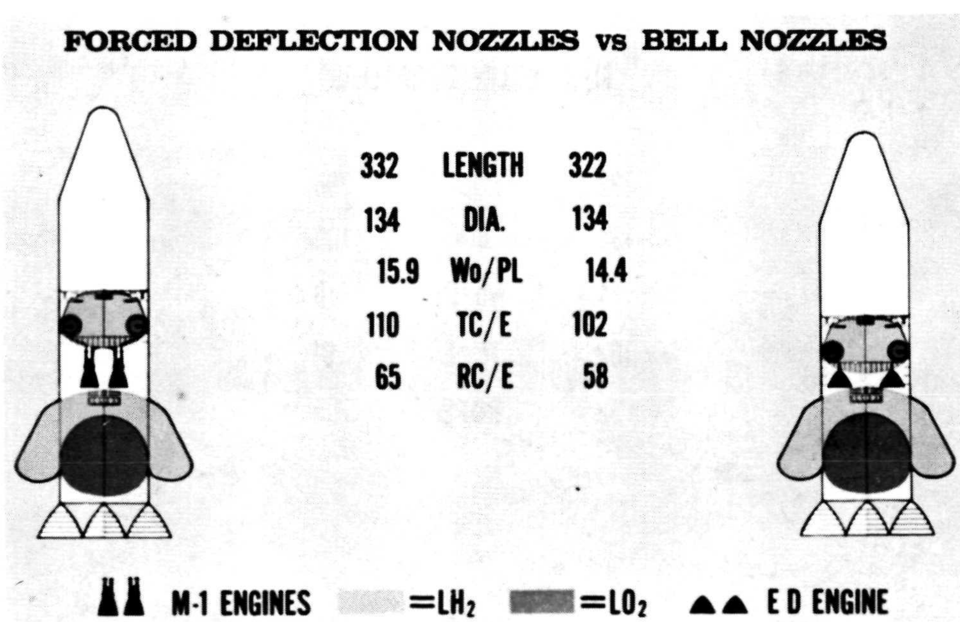

Figure 4

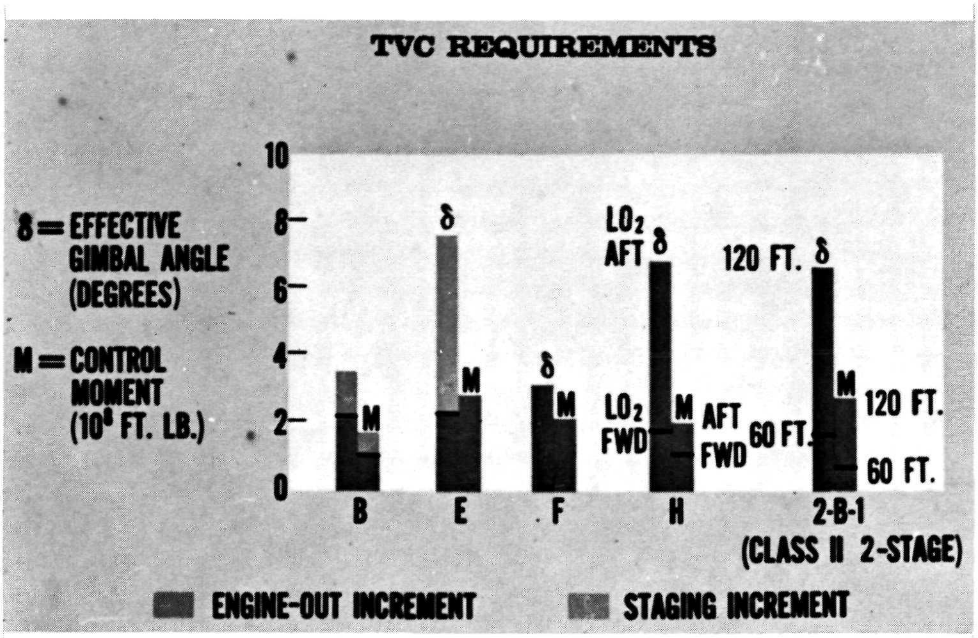
TVC REQUIREMENTS
δ = EFFECTIVE GIMBAL ANGLE (DEGREES)
M = CONTROL MOMENT (10^8 FT. LB.)
0
2
4
6
8
10
δ
M
LO2 AFT
LO2 FWD
AFT
FWD
120 FT.
60 FT.
120 FT.
60 FT.
B
E
F
H
2-B-1 (CLASS II 2-STAGE)
ENGINE-OUT INCREMENT
STAGING INCREMENT

Figure 5

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3. RESULTS OF PROPULSION SYSTEM DESIGN

AND TRADE-OFF STUDIES

By Joseph Streetman

Nova Propulsion Group, General Dynamics/Astronautics
General Dynamics Corporation

The major portion of our Nova part II study effort has been devoted to trade-off studies in the area of advanced propulsion. We have had, in general, two objectives in this trade-off study. The first was to identify the areas of advantage or disadvantage for forced deflection engines compared with bell engines. The second was to compare the various types of forced deflection engines that we know of today with the objective of determining if possible the most optimum type.

The first of these trade-off studies that I will discuss is a generalized comparison of bell versus expansion deflection engines, a part of which Mr. Wamser has already covered. When you compare expansion deflection (ED) engines or forced deflection engines with bell engines you immediately discover two quite obvious characteristics. (See fig. 1.)

First, the forced deflection engine is quite a bit shorter than the bell engine at a given thrust and area ratio as can be seen in figure 1. The particular configuration of the forced deflection engine is about 50 percent of the bell length. We can get much shorter length with some of the other forced deflection engine configurations. Also given in figure 1 is one of the important dimensional characteristics of the forced deflection engine, the D_P/D_T ratio, in this case 3.5 to 1. This is the ratio of the diameter of the plug engine if a plug is used or the center body if an expansion deflection engine is used to the diameter of the equivalent bell engine of the same thrust.

The next characteristic of the forced deflection engine is that it presents the possibility of altitude compensation. In figure 2 the delivered I_{SP} is plotted against altitude; the dashed line shows I_{SP} for the bell engine at the normal compromise area ratio ϵ for a first-stage application, in this case 35 to 1 for a 3,000-psi engine, and altitude compensations of 90, 95, and 100 percent. The performance of the bell engines at low altitudes is considerably better than for the 90- to 95-percent altitude compensation cases, although these are somewhat lower than the 100-percent altitude compensating engine. This might indicate that there is some sort of a crossover point below which the bell engine may be the most desirable from the performance standpoint and above which the forced deflection engine would be.

We have a somewhat clearer way of presenting these data. If we check typical vehicle trajectories and obtain altitude versus time histories, and plot specific impulse versus time rather than altitude, and if we integrate these curves and obtain the integrated average I_{sp} versus time, we come up with the plot shown in figure 3.

In this figure the integrated average I_{sp} is plotted against flight time for a bell engine at the compromise area ratio of 35. For the 90- and 95-percent altitude compensation cases, we see that the crossover points are approximately 222 and 160 seconds, respectively. Times longer than these would favor the forced deflection engine because of a higher delivered average I_{sp}. Times shorter would favor the bell engines.

So with 90- or 95-percent compensation there would be very little payoff for the use of the forced deflection engine from a pure performance standpoint. However, for the 100-percent altitude compensation case, there is quite an advantage throughout the entire flight range.

We made a study to see what would happen if we did achieve 100-percent altitude compensation, to see what the maximum possible payoff would be. The results are shown in figure 4. The ratio of gross weight to payload is plotted against expansion ratio. In this case it is a lox/RP first stage of the two-stage vehicle using 2,000-psi chamber pressure. We show the performance for bell engines if 100-percent altitude compensation were achieved. As can be seen, there is not much of a payoff. This payoff represents about 3.5 percent out to the 100 area ratio. If a more practical compensation were assumed, the payoff would be considerably less.

The characteristic of the forced deflection engine that makes for short length, however, makes possible the use of a very high area ratio in the second stage, as Mr. Wamser pointed out. We have used area ratios from 40 to 150 in the vehicles we studied in part II. Over this range we found an improvement in vehicle performance of about 10 percent, and a comparable, almost equal, and in some cases greater, improvement in vehicle cost effectiveness.

So there is quite a payoff in the use of the forced deflection engine in the second stage if good efficiency is achieved.

Figure 5 shows a comparison of the single large external diffusion engine with a four-external-diffusion installation at the same total thrust. In this case the thrust level is in the 30 million to 20 million pound class.

First we might compare the two single installations, one with a D_P/D_T ratio of 10, which makes it quite large diameter, and the other with a low D_P/D_T ratio of 3.5, which appears to be about optimum from the weight standpoint. In the configuration on the right there is quite a bit of difficulty in mating the engine to the vehicle structure, and quite a bit of extra length required. Both of these features cut into the advantage of low engine weight with low D_P/D_T ratio.

The configuration having a D_P/D_T ratio of 10 has been selected for all our detailed single-engine installation studies in part II.

Next we might compare the single ED installation with the four ED engine configuration shown on the left in figure 5. First, there is a height advantage over the single ED engine installation. The single-engine configuration has a distributed load application to the vehicle rather than the point load application that you get with the four-engine configuration. This normally would make for some savings in vehicle structure weight. However, we also note that with the cluster four configuration it is quite easy to build the engine with the D_P/D_T ratio for low engine weight.

I might also discuss thrust vector control. If you look at the cluster four configuration on the left in figure 5, you see that almost any of the known schemes today can be used. Gimbaling appears to be about as applicable as it would be for a bell engine.

Secondary fluid injection also looks okay. However, if you look at the single ED configuration, gimbaling certainly doesn't appear applicable at least in this D_P/D_T range. Secondary fluid injection doesn't look very good either. In fact, the only thing that looks promising at all is throttling, either of quadrant or modules, if modules were used in this configuration. And even here we have very little experimental evidence as to how effective this would be.

Some of the data in this comparison are summarized in table I. Included in the table are the thrust vector control types just discussed, and their applicability.

With regard to engine weight, there appears to be about a 40-percent penalty with the use of the single ED engine compared with the multiple, primarily because of the D_P/D_T ratio advantage of the multiple installation. The cost advantage for the single installation looks like about a 10-percent improvement over the four performance, although you can get a higher expansion ratio with the single ED installation because of more effective utilization of the base area, the performance is the same.

To summarize this portion of the study, the single ED installation certainly looks better than the multiple installation and does have some installation advantages. But because of better flexibility, if the choice had to be made at the moment, we would choose the cluster four configuration.

A comparison of the ED versus the plug engine is shown in figure 6. We will do this on the basis of the single large ED engine versus the plug engine.

This comparison, between the ED and plug is quite difficult. It is one in which there are a lot of really important questions for which at the moment we have very few experimental answers. This discussion will be partly intuitive.

First, looking at the ED, in either the bell module configuration or the toroidal combustion configuration, it appears that the expansion sections are more or less protected from the effects of base flow, low base pressure, and so forth, which is not the case with the plug engine. In the right-hand figures it is apparent that we might get some significant performance losses through overexpansion due to flow around the vehicle base as shown in the flight case compared with static no-flow case. It appears that this loss might be less with the bell cluster plug configuration than with the toroid because you do control the expansion down to a fairly high area ratio.

Second, thrust vector control appears quite difficult in either configuration compared with the cluster four ED engines. However, it looks like it probably would be less of a problem with the plug configuration than with the ED. For the plug configuration, it seems throttling or perhaps gimbaling of the modules, if they are used, offer promise of success.

The efficiency we don't know at the moment. It is quite important, I might point out. Here we refer primarily to high-altitude efficiency because that is where most of the payoff would seem to be, and where uncertainties at the moment might cost us the most performance, although low-altitude efficiency is also important but to a lesser degree.

With regard to base heating with the two configurations, it appears that the problem is probably less with the ED engine than with the plug engine because at least up to a very high altitude the exhaust jets diverge rather than converge as they do with the plug engine; with the plug engine, the exhaust jets probably impinge and jet back up to the vehicle base, which probably would increase the base heating to a large degree.

The next comparison is of the type of combustion chambers to use (fig. 7). Two leading contenders today seem to be some version of the bell module and the toroidal combustion chamber. In weight, we feel that there is a slight, though not overwhelming advantage for the bell cluster. This we feel is probably because, at least for engines having high D_P/D_T ratios which are appropriate for plug and some versions of the single ED, the bell module is more efficient volumetrically than the toroid would be, and there would probably be some advantage in weight for the combustion chamber itself.

With the ED engine, the external expansion section has to withstand the effects of the high-pressure hot jet directly from the toroidal combustion chamber in a ratio of something like 2 to 1, around 300 psi. In the case of the bell cluster, the pressure could be atmospheric or less, if desired.

In terms of controlled expansion, we have already indicated that the bell module might have the advantage here because with base flow the area ratio is controlled down to a fairly high value with the bell modules.

For thrust vector control, it appears that probably the problems are less with the bell module than with the toroid, as mentioned earlier. The bell modules might be either gimbaled or throttled.

As for efficiency, we don't know at the moment. On the one hand the internal expansion portion of the bell cluster is as good as we know how to make expansion sections today. However, the discontinuities you get due to the relatively small numbers of exits to the external expansion section might cause some appreciable losses whereas with the toroidal system, the exit is almost a continuous annular throat, which might make for system advantage for the toroid.

With respect to flexibility, there is quite an advantage in the toroidal combustion chamber. For one thing the area ratio that can be obtained is not a function of the number of modules used as is the case with the bell cluster. This means that you don't necessarily have to use a large number of engines. You can select an engine size or a module size which is convenient and economical to develop, produce, and so forth, whereas with the bell cluster you are more or less driven to a fairly large number of engines. This, of course, means that engine-out capability is probably necessary with the bell cluster, where it is not necessarily required with the toroidal system, depending on the number of segments you choose to develop.

Figure 8 shows the plug-engine area-ratio amplification, that is, the area ratio of the plug engine to the area ratio of the bell module used plotted against the number of engines. It indicates that to get an appreciable amplification, a fairly large number of engines has to be used. It is only on the order of 1.5 with 10 engines. To get good amplification you have to go out to the 20 region or even higher, which is desirable in some of the larger vehicle diameters.

Perhaps it would be well to summarize the plug versus ED engines and the comparison of the bell module with the toroid system.

We pointed out that there are a lot of important key questions that would affect the decision in either of these areas, which we believe require experimental answers. Other factors include:

(1) Efficiency. Here we refer again primarily to high-altitude efficiency

(2) The effects of base flow and pressure need to be known

(3) Base heating

(4) Sensitivity to engine out in both configurations

(5) Side-force generation

(6) Thrust vector control, both in the types that are applicable and the effectiveness and efficiency of the types

(7) Degree of altitude compensation that can be obtained with the various types

Figure 9 shows a comparison of single versus multiple pumps for use with a multichamber bell cluster plug engine. In this case there are 24 bell modules clustered around a plug.

In the configuration using multipumps, the pumps are designed with the shortest possible structure that we can build. In the configuration with four pumps, or with single pumps, they require quite a bit of additional structure. This is of the order of 10 or 15 feet for the single-pump configuration, and considerably less for the multiple-pump configuration.

Also we have shown all these configurations assuming the use of the topping cycle which is the most efficient one we commonly talk about. However, with the topping cycle a high pressure is required, at least one high-pressure hot gas line between each pump package and the chambers. In the case of the multichamber pump installation this is no particular problem. The lines can be made quite short. However, with the single-pump configuration, the lines are quite long. They radiate out from the pump package to the chambers. And they are fairly complex because expansion joints would be required to avoid heating in the base area. You would have to have quite a bit of insulation of these lines. For this reason, we don't believe that the topping cycle would be the best one to use with this configuration. We think the gas generator (GG) cycle should be used with the loss in efficiency of something like 1.5 percent, which means about 7 seconds in specific impulse.

For the four-pump configuration, the question is not quite so clear. Probably there the GG cycle would be feasible and with the same loss you would get with the single-pump configuration.

In weight, there appears to be a significant advantage for the multipump configuration compared with the single pump, and somewhat lesser advantage, perhaps none at all, for the multipump compared with the four-pump configuration.

Figure 10 shows the reliability and economic considerations associated with these configurations. Engine reliability shows the advantages we can get with pump-out if 16 pumps were used, and with two engines out at 32. Note that the reliability is higher at 32 with a couple of modules out than it is with the single-pump configuration.

This figure also shows an improvement in cost effectiveness with an increase in the number of pump packages used. Actually, we drew a smooth curve through the points for the single, the four, and the 32-pump configuration; we realized later that in reality there should be some breaks in this curve. In fact, there should be two curves, one for the use of the GG cycle which would probably look something like the curve shown and one for the use of the topping cycle which would go through the 32-pump configuration, and which also would have a shape similar to this curve. There should be some breaks for reliability.

To summarize, considering all the installations, weight advantages, plus reliability and costs, the multipump configuration appears optimum.

The question of the optimum number of pumps to use in a multichamber plug configuration leads logically to the more general question of the optimum number of engines for use in the Nova class vehicle.

Figure 11 shows the reliability and cost data which we believe apply, along with the assumed values for α, the probability that a good engine will be shut down mistakenly and β, the probability that a bad engine won't be detected and shut down, which leads to catastrophic failure. The assumption for α and β makes quite a difference in the engine-out reliabilities shown, especially for the very large number of engines.

Figure 12 shows the rest of the cost data that we believe apply. The first unit cost includes both the first and the second stage as do recurring and total cost effectiveness. There is very little to choose from. The total-cost-effectiveness curve is quite flat between 4 and about 24 engines, but starts to increase quite rapidly at 24. We don't consider it particularly advisable to go beyond that point. In fact, this curve would probably increase quite significantly at the single engine point. It seems that a good choice of numbers of engines to use might be the maximum number that you could use without requiring engine out, which is something of the order of five or six. Perhaps the minimum number that is practical to use with only one engine out is on the order of 12 or 14 engines.

The next trade-off that we will discuss is chamber pressure optimization. Figure 13 shows vehicle size effects with chamber pressure. The vehicle on the left has 1,000 psi in both first- and second-stage engines. It happens that they are in ones; the second stage has an area ratio of 40. The vehicle on the right has the same number of modules and plug clusters. Two of the modules are used in the second stage with an area ratio of 100. Even with this configuration there is quite an area ratio advantage for the use of high pressure in second stage.

Notice the ratios of gross weight to payload for the two configurations, about 19 versus about 14, a significant advantage for the higher chamber pressure. However, most of this is due to the second stage. Note also the lengths: 173 feet for the vehicle on the right versus 196 feet for the vehicle on the left. The widths are, respectively, 135 feet and 146 feet. These are significant but not overwhelming advantages.

Figure 14 shows the cost considerations and performances for the first stage only, using lox/RP with an expansion ratio of 40, of four ED engines. Performance, here shown as the inverse of the ratio of gross weight to payload, increases by about 10 percent between 1,000 and 3,000 psi. However, the engine cost increases at an even faster rate, so that the total vehicle cost increases somewhat between 1,000 and 3,000 psi. However, this is only about 2 percent, which is almost within the accuracy of the study, so we have had to look at some more intangible considerations - less tangible than cost performance and so forth.

In terms of engine size, there is a significant advantage in going from 1,000 to 2,000 psi, and some advantage in going from 2,000 to 3,000 psi, but not nearly as much as the step from 1,000 to 2,000 psi.

In the study that we show here we assumed a constant allowance for chamber cooling over the range. We believe that we were somewhat conservative at the low pressure end and somewhat optimistic at higher pressures. A more detailed study probably would shift this curve somewhat more in favor of the low pressure chamber end. We have some indication that you could get by just with regenerative cooling, no fin cooling, up to 2,000 psi.

So considering all these things, we feel that 2,000 psi is the best compromise chamber pressure for this particular configuration.

Now we come to augmentation, which is quite important for Class III vehicles. About the first thing we found when we began to study air augmentation some months ago was that there was very little data available for ejector performance in which the primary is the high energy source. In fact we found no data at all on performance when the primary was a lox/hydrogen rocket. For this reason, we started a relatively simple test shown schematically in figure 15 a short time ago with General Dynamics funds. It is now supported by NASA.

We use a lox/hydrogen rocket with a thrust of 120 pounds, chamber pressure of 450, optimally expanded. We are using tubes of three different area ratios, three different diameters, several lengths to get several different length-to-diameter ratios. We are measuring the parameters shown in figure 15, the rocket thrust, tube thrust, total pressure distribution at the tube exit, and static pressure distribution along the tube wall.

We are now getting data and are in the process of reducing and evaluating and so forth.

Figure 16 is a picture of the test setup. We have shown the rocket, tube, instrumentation, and so forth.

Several times during the presentation we have mentioned the importance of certain questions for which as yet there are no experimental answers. We believe most of these answers should be obtained from the NASA Advanced Technology Program. The more important of these and the ones that we believe would be applicable for the Advanced Technology Program may be summarized as follows:

(1) Nozzle efficiencies at high expansion ratios, primarily at high altitude

(2) Efficiency loss with engine out for the various chamber configurations and the various engine configurations possible

(3) Side-force generation or thrust vector control, type effectiveness, and so forth

(4) Degree of altitude compensation, or low-altitude efficiency

(5) High-pressure technology in the areas of turbines, pumps, bearings, chambers, chamber cooling, and so forth

To conclude, we have covered quite a lot of territory. The more important conclusions may be summarized as follows:

(1) Altitude compensation is not essential for two-stage vehicles. The advantage for the forced deflection engines is primarily in installation, vehicle height, and in the second stage, in the higher area ratios that can be obtained.

(2) The single ED engine has installation advantages, but the multiple ED engine installation is the more flexible and at the moment we feel that would be the optimum choice.

(3) There is no decisive advantage for either plug or ED engines that we can see at the moment.

(4) The toroidal combustion chamber appears to be more flexible than the bell module, but weight and performance comparisons are uncertain. More answers are required.

(5) For the multichamber bell plug cluster, the multipump configuration is optimum.

(6) At a given thrust, a small number of engines is preferable to a very large number.

(7) For a lox/RP first-stage, 2,000-psi chamber pressure appears to be the best compromise.

TABLE I

SINGLE vs MULTIPLE E D ENGINES	SINGLE	MULTIPLE
TVC-GIMBALING	NO	YES
TVC-SFI	NO	YES
TVC-THROTTLING	MAYBE	YES
ENGINE WEIGHT	140%	100%
EXPANSION RATIO	NOT LIMITED	100*
COST	90% (1)	100% (4)
WT./PL RATIO	15.3	15.3

*P_C = 3,000 PSIA MOUNTING DIAMETER = 70 FT.

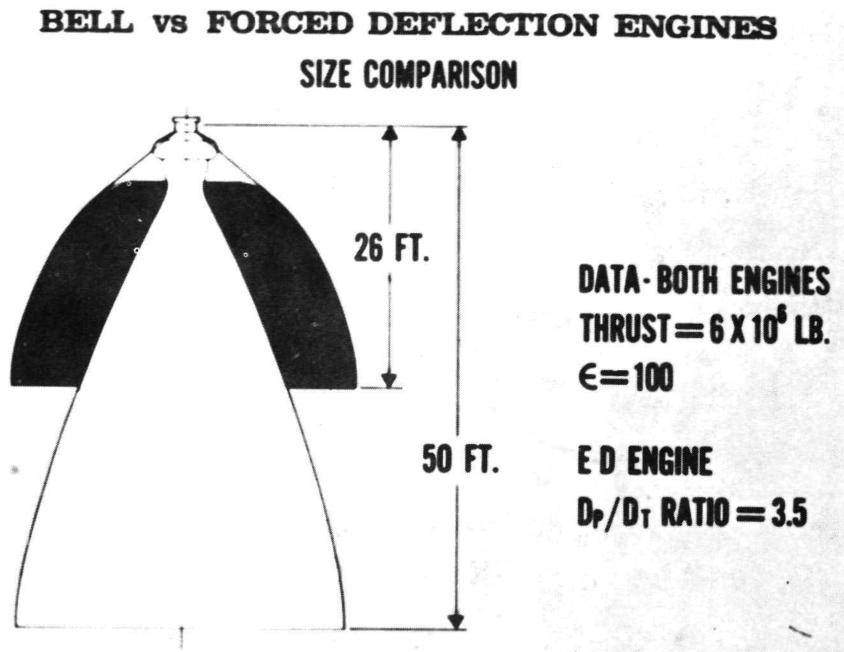

Figure 1

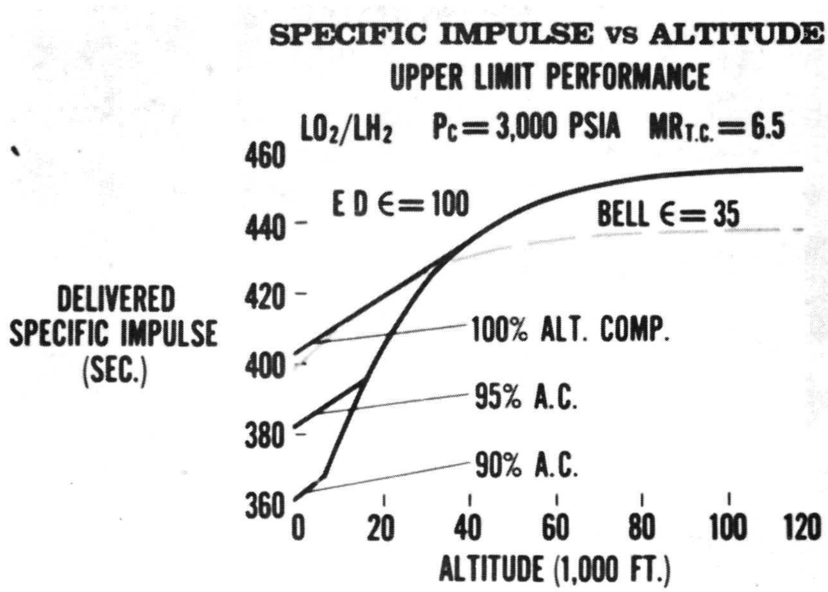

Figure 2

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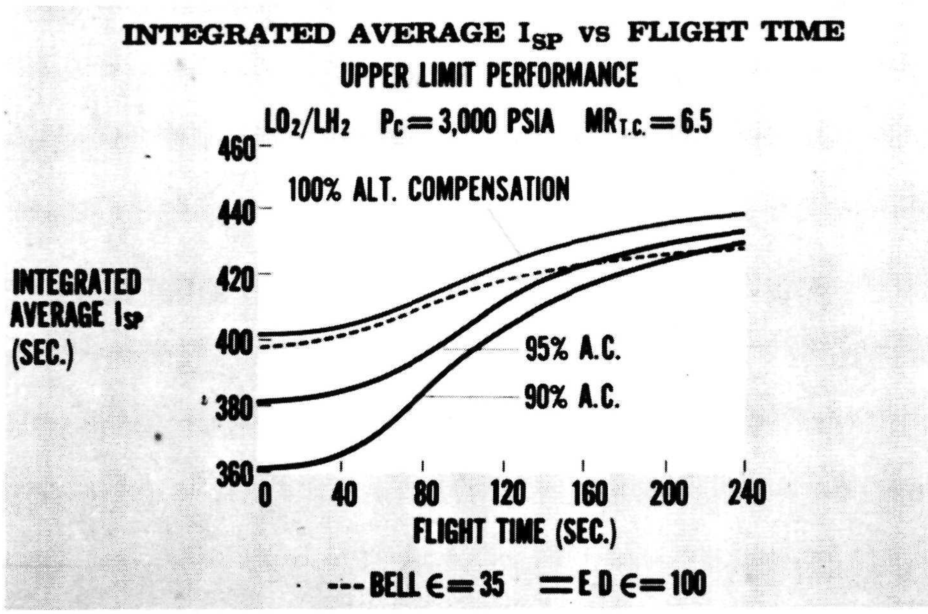

Figure 3

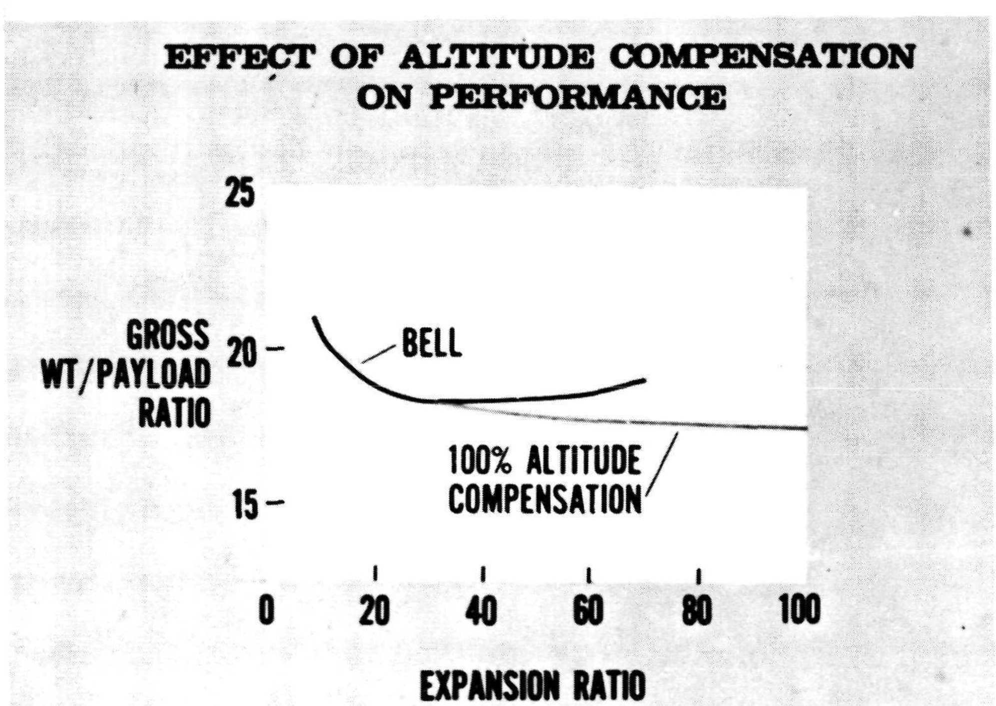

Figure 4

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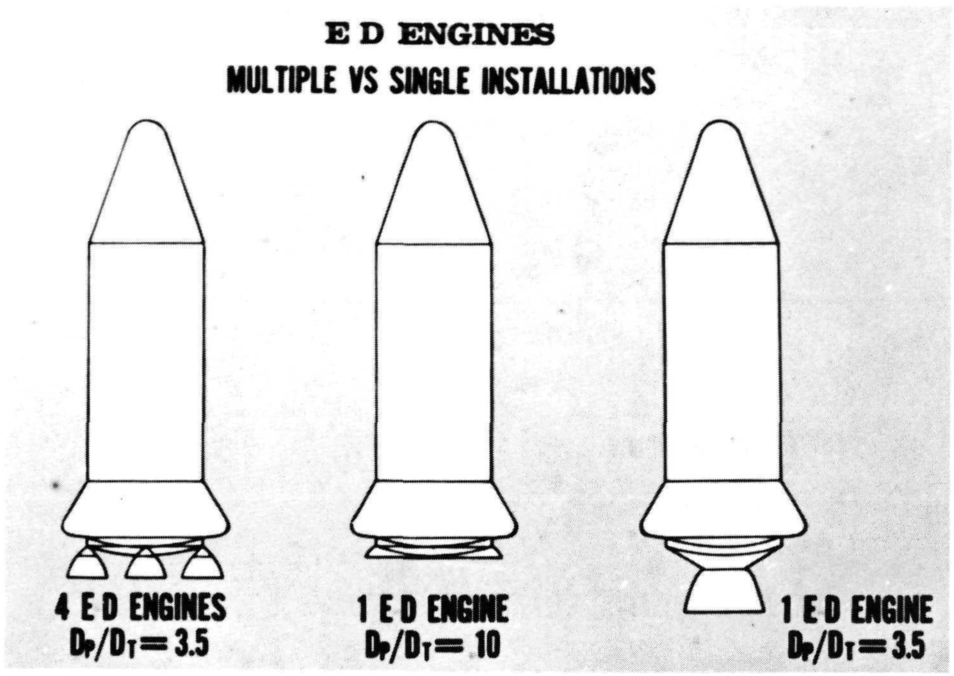

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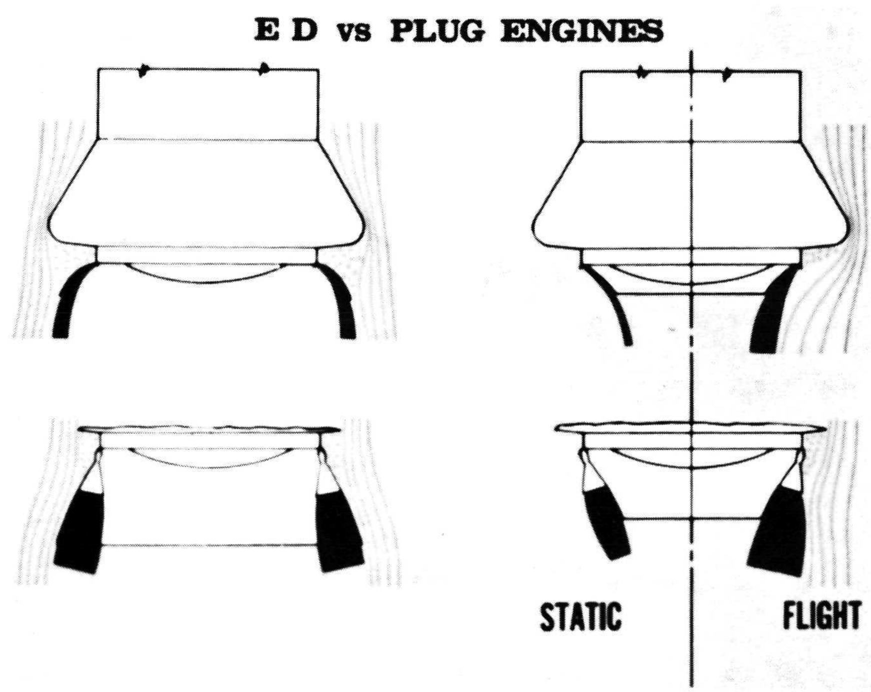

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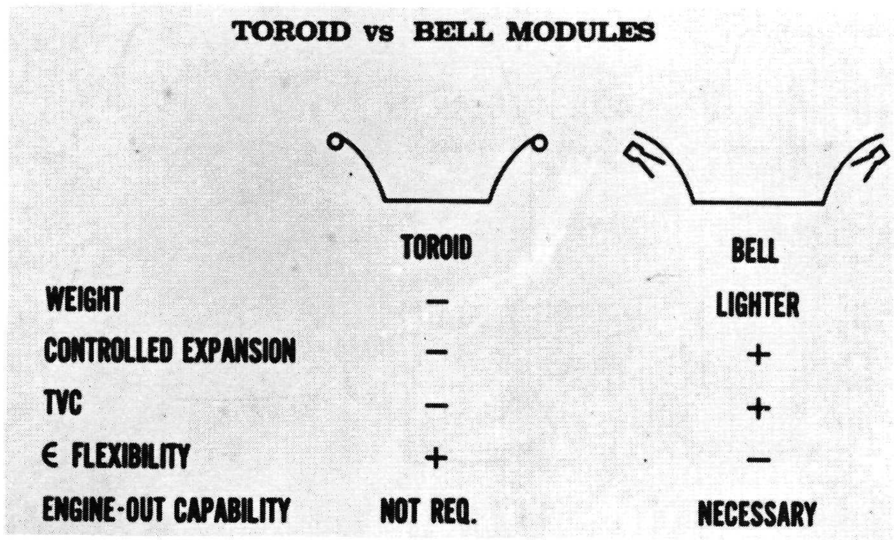

Figure 7

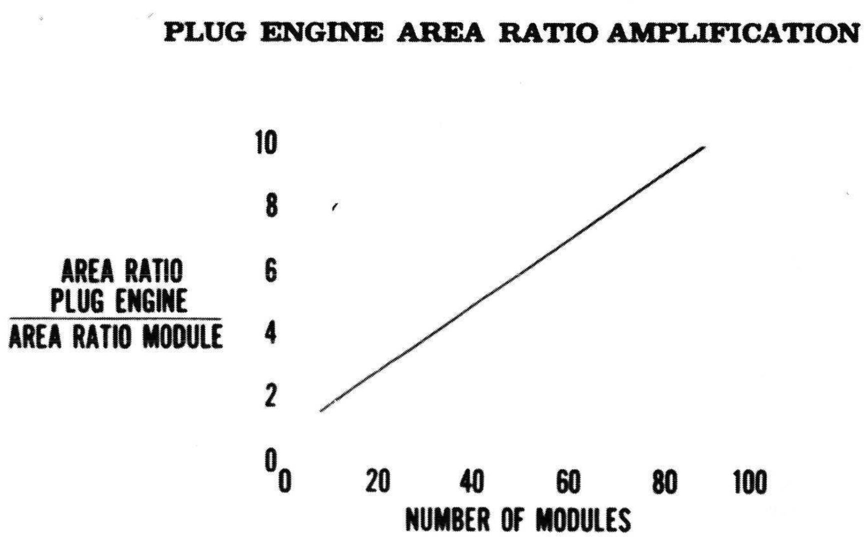

Figure 8

SINGLE vs MULTIPLE PUMPS

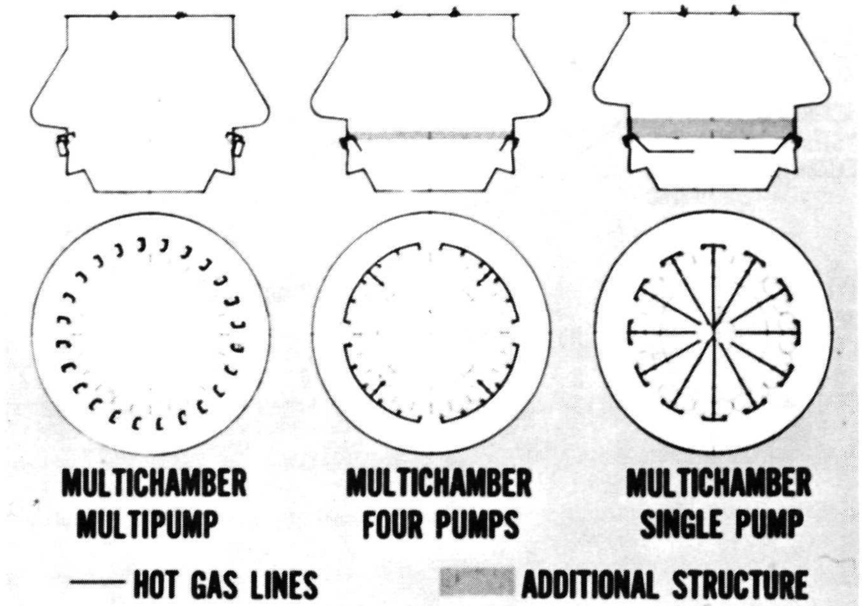

Figure 9

SINGLE vs MULTIPLE PUMPS

THRUST = 32×10^6 LB.

32 THRUST CHAMBER MODULES WITH TWO-OUT CAPABILITY

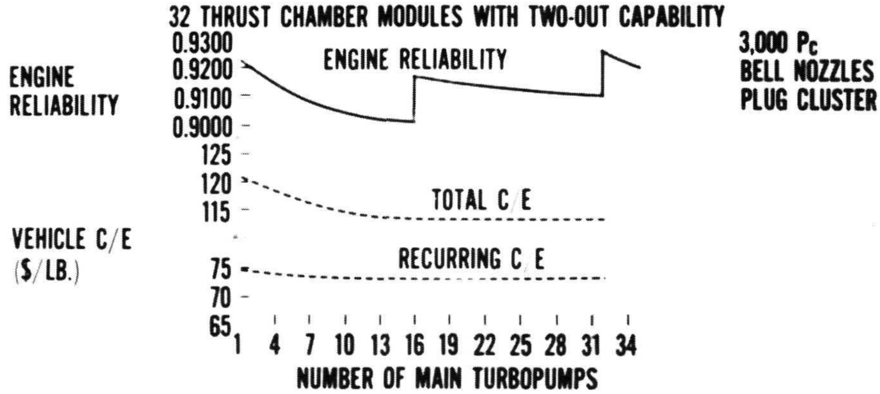

Figure 10

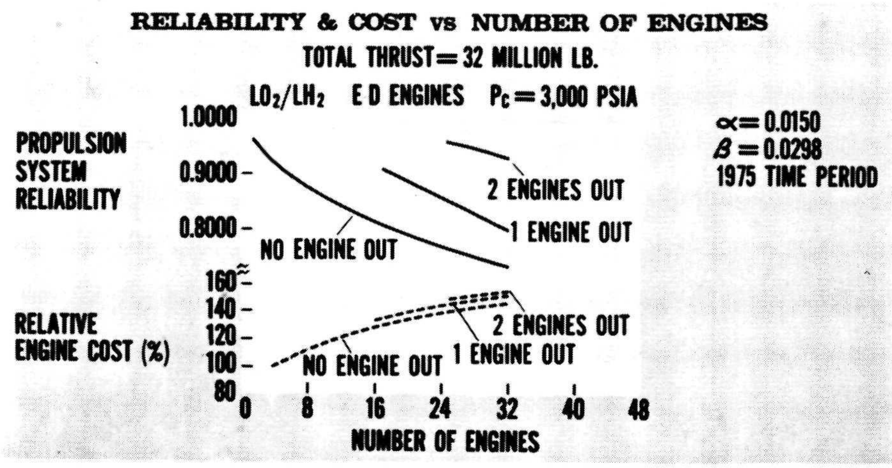

Figure 11

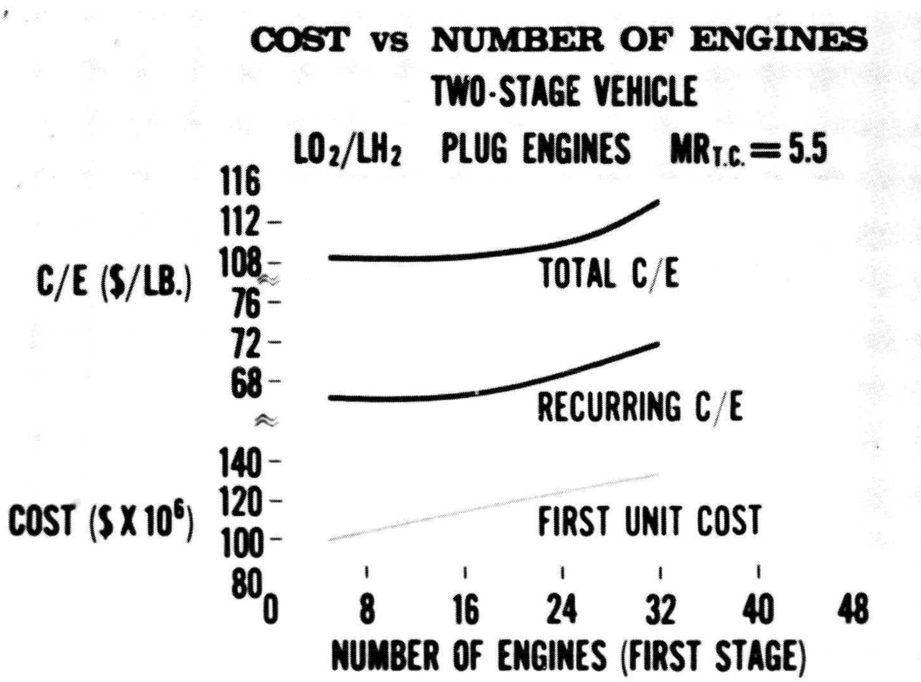

Figure 12

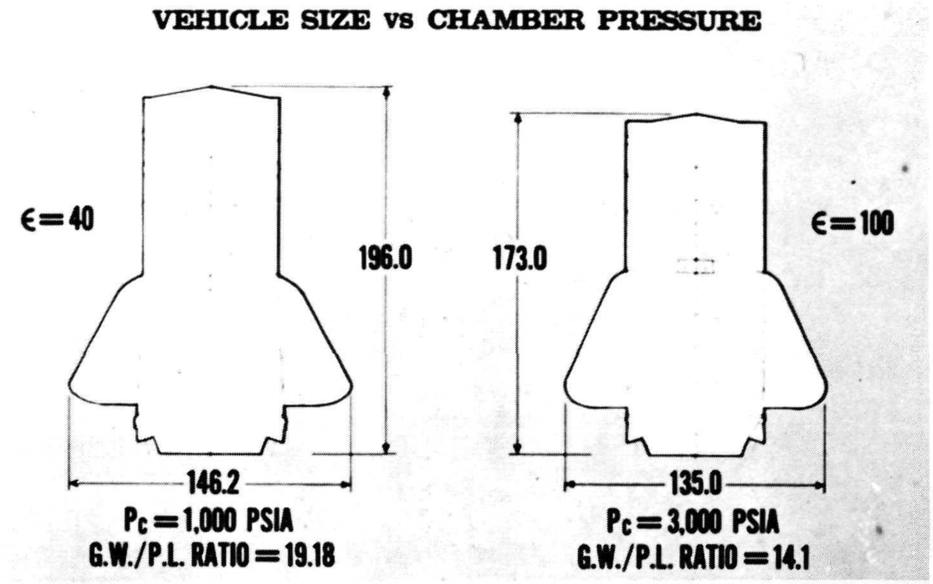

Figure 13

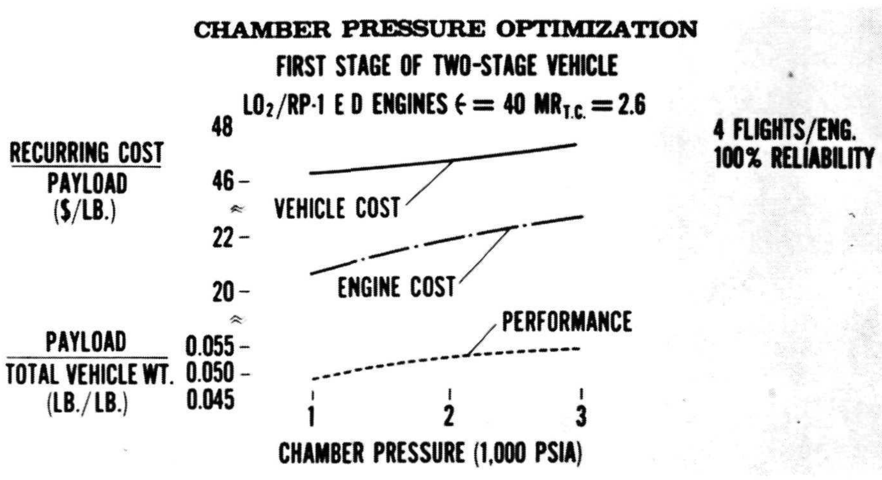

Figure 14

STATIC AIR AUGMENTATION TEST

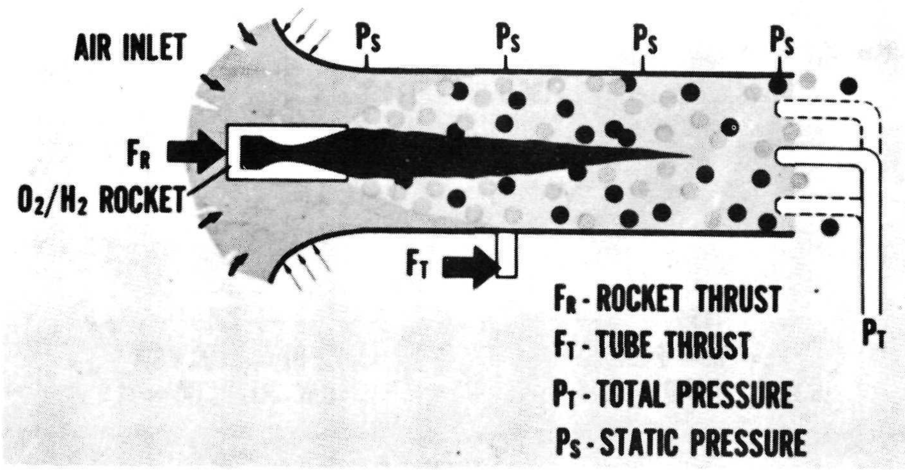

Figure 15

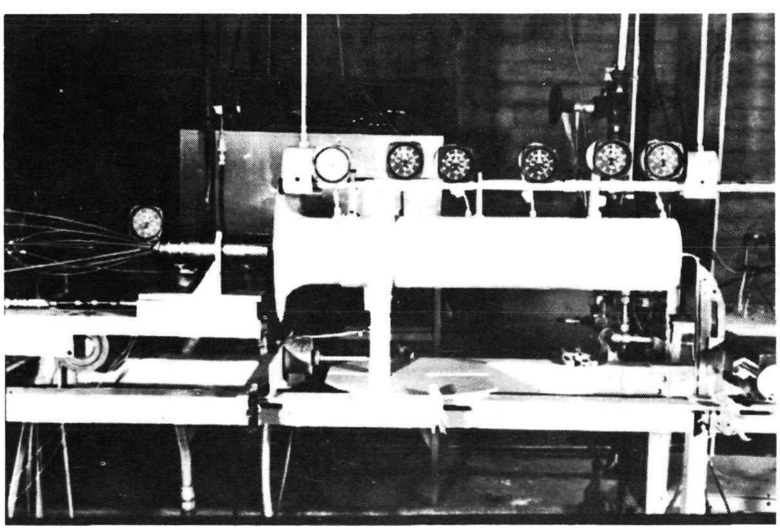

Figure 16.- Test setup.

4. PROPULSION SYSTEMS RELATING TO CLASS II AND CLASS III VEHICLES

By John Youngquist

Baltimore Division, Martin Company

We have classified our engines, our vehicles, in three classes, and essentially these three classes are determined by the propulsion technology. I would like to discuss the class II and class III vehicles and the propulsion systems relating to them, with emphasis on the performance, trade-offs, control requirements, and reliability. I would then like to begin discussing class III by considering air augmentation.

By way of review, figure 1 shows our class I vehicles on the left which were of one M-1 and solid M-1 and class II vehicles on the right. The third vehicle from the left is a tandem vehicle. The fourth vehicle shown is a single-stage vehicle. The last vehicle is a partial stage vehicle with engine staging.

These vehicles are all characterized by advanced propulsion systems with hydrogen and oxygen. Vehicles three and four from the left have a multiengine plug arrangement and the vehicle on the right-hand side is a large advanced altitude-compensated engine.

The thrust levels of the three class II vehicles, and we will be talking more of this, are about 18 million psi, and the two class I vehicles each have a thrust level of about 30 million psi.

Figure 2 shows in summary the cost effectiveness picture. We actually developed a concept of the uncertainty of the cost effectiveness for each of these vehicles. Although I won't dwell on this, the third vehicle from the left is the single-stage vehicle. Second from left is the two-stage hydrogen vehicle and the left-hand vehicle is a partial stage. These latter two vehicles are in the range of $0.95 to $1.10 per pound cost effectiveness. Note, however, that guess of cost spread for the single-stage vehicle is, as expected because of its sensitivity, higher than that for the other two class II vehicles.

Relative to the geometry compression, figure 3 shows a number of advanced rocket engine configurations.

This goes from one extreme to another at a constant chamber pressure of 1,000 psi. An F-1 chamber is shown at left. You can go through the various ED and RF nozzles or conceive of, say, a ring nozzle (this is similar to the Rocketdyne torus. You can even conceive finally of the maximum compression at the same chamber pressure with the same total throat area with the same total exit area, the thrust plate. Thus, in our advanced engines we are tending toward rearranging the chamber geometry to get this maximum compression.

Figure 4 shows the effect of increasing chamber pressure. Here, against a constant diameter we are making an improvement in area ratio of about 2.5. Unfortunately this is a log scale. It goes from about 1,000 to 3,000 at 2.5. So these two effects are shrinking our power plant compartments and we hope to achieve performance improvements here.

Figure 5 shows a classical portrayal of what we can achieve with area ratio. This is for lox/hydrogen as against mixture ratio. The dashed line is for chamber pressure of 1,000 psi, and the solid line is for 3,000 psi. We see that the chamber pressure per se is not responsible for performance; rather, it is increasing the area ratio. So the importance of the geometry compression is clear.

Figure 6 shows the effect of altitude compensation. The upper two curves are at 3,000 psi. The lower two curves are at 1,000 psi. The dashed curve in each case represents a nozzle that is designed for 3 million pounds of thrust against an 80-foot-diameter vehicle, not to overexpand at sea level. It is a bell nozzle. The solid line represents 100-percent augmentation at the two chamber pressures. Against this given vehicle diameter we can get a higher chamber pressure against a higher expansion ratio. Thus, at a given diameter, we certainly can achieve more integrated total impulse with the higher chamber pressure. Note the area that relates to the efficiency of altitude compensation.

We have expressed this in a comparison in table I. What we have done here is try to break apart the performance contributions that relate to altitude compensation and mixture ratio change, again from hydrogen/oxygen vehicles. In a two-stage vehicle - and we have expressed these by exchange ratios in effective specific impulse - you would add all these effects up and optimize the vehicle; however, here the effects are separated so that we can understand the problem a little better.

On a two-stage vehicle, the effective altitude compensation, including the high altitude area ratio, results in about 7 seconds of specific impulse. In a single stage, which has more sensitive performance relationships, it is about 10 seconds.

The area between 100-percent augmentation and the nozzle that was designed so that it wouldn't overexpand at sea level in each case has resulted in an integrated average of about 1 second extra specific impulse. Changing the mixture ratio, while we optimize our vehicles to any particular mixture ratio, produces specific impulses of 6.5 to 7 seconds. Changing from 5 to 7 we see a 5-second improvement - this is converting the weight to specific impulse - and a second improvement on the single stage. If you did not have altitude compensation but raised the chamber pressure from 1,000 to 3,000, and changed the mixture ratio from 5 to 7, that effect alone gives these values.

We can conclude from this chart that what we are desirous of knowing about altitude compensation is how it works and how we can best exploit the idea of getting a high area ratio to get the full performance benefits.

Figure 7 shows these benefits turned back into payload increase against chamber pressure, where no compensation on a single-stage vehicle is compared with no compensation and compensation on a two-stage vehicle. Again we can conclude that for the two-stage vehicle the improvement with chamber pressure is not very substantial. However, it is vital to the concept of a single-stage vehicle.

Next, let us discuss control requirements, which are shown in table II. Consider the bottom three configurations. The maximum δ requirement is for hydrogen/oxygen vehicles which we have configured with the lox load tank forward. We are getting relatively low peak gimbal or peak thrust vector requirements. We think this offers an opportunity to look at new methods of control beyond gimbaling, certainly secondary fluid injection, but also throttling.

Figure 8 shows a duty cycle which at 2°, the average is more like 1°, is characteristic of these low length-to-diameter vehicles.

Figure 9 shows this requirement translated into secondary injection fluid requirements. This particular vehicle is a two-stage hydrogen/oxygen vehicle which has a propellant load of about 13.5 million pounds. We can see that the total injection fluid is about 0.6 of 1 percent, which is quite economical. We have also analyzed, however, the throttling requirements for these low thrust vector requirements; figure 10 shows the level of throttling that is required to satisfy the control requirements on the two-stage lox/hydrogen vehicle. Essentially we are saying that we have about a ± 15-percent control requirement, if you can postulate working around a mean thrust. If you can say it is 100 percent down, the requirement then becomes about 100 percent to 70 percent, which particularly in the advanced or high-pressure engines we think the response rate that we need is about 5 percent thrust per second; we think this is possible to achieve in these advanced engines.

There are some other advantages that we might accrue from throttling, if we would accept it as an approach to controlling the vehicle. These advantages may be summarized as follows:

(1) Low maximum acceleration
(2) Compensate thrust variations
(3) Control guidance capabilities
(4) Shape thrust-time history

In the single-stage vehicle particularly we could use throttling to minimize the cutoff burnout load factor. In turn, the low maximum acceleration we anticipate would be very desirable for the space payloads that the Nova will carry. In the event that we go to modular type thrust units, the throttling can be used to compensate for variations or flight variations.

There are possibilities with guidance control systems, particularly for rendezvousing approach near orbit or cutoff of the engines where the low thrust capability looks interesting.

In turn, we feel that we can make some improvements relative to heating and loading on the vehicles by shaping the trajectory using throttling.

Figure 11 shows a composition of reliability budget for a two-stage hydrogen vehicle and a single-stage hydrogen vehicle. The goal we have selected for the two-stage vehicle is 92 percent; the general goal we have selected is 90 percent or higher above the target for the Nova vehicles. The goal for the single-stage vehicle is 96 percent. The breakdown shows a lower requirement for the engine cluster of a two-stage vehicle than for a single-stage vehicle. The two-stage vehicle has 18 module engines in the first stage and two engines in the second stage. Most of this number, 0.975 compared with 0.992, is related to the altitude start of the two upper-stage engines.

In similar fashion the number is lower in pressurization and feed, essentially half the operating hardware. This is reflected also in the staging which is included in all other systems which is the lower number than the single stage.

Figure 12 shows the engine cluster reliability for different operating concepts. We are postulating the previous numbers, engine out with no holddown; most vehicles flying now, except through the Saturn S-1A, are holddown only. In this figure we see how the single engine out supports the reliability of the multiclustered engine arrangements.

Let us look at the effect of single-engine module reliability shown in figure 13. The two-stage vehicle has approximately 20 engines; the single-stage vehicle has about 24 engines. Thus the number of engines is significant here. What is significant are the altitude start requirement and the holddown and no altitude start in the single stage. The single-stage reliability, with about a 99-engine individual module, is getting up to very respectable total vehicle reliabilities.

Figure 14, although somewhat misleading, is intended to present the approximate thrust levels of all the 150K engines that the country has developed in this area, and show what we believe to be the reliability retention. Actually, we see historically an improvement in time against the number of engines flight tested. We can see that the S-1A essentially has a perfect flight test record. If we include its ground test record we are, however, ahead of any previous experience in this class engine module.

We feel that this demonstrates the effort that has gone into this class of engine. We actually have Aerojet engines in here, too, but we are talking about the entire industry technology, which demonstrates the improvement we have seen over the years.

Figure 15 presents another look at the reliability situation wherein we have taken the failure history data on three different engine programs (Pratt & Whitney, Rocketdyne, and Aerojet). We have taken the failure history. Plotted along the abscissa is the planned number of engine firings through PFRT. If we had entered the history of all these engines at about 400 firings, we would have expected a spread like this. Thus, looking at the composite of the industry, at any number of tests, we can see that certainly for a low number of tests through PFRT would have a wide spread. It is our judgment that more than 1,000 tests, where the curves flatten out, will give some certainty of what we require.

This goes certainly into the modular versus large thrust engine development. Also shown in the figure are the approximate prices for this kind of a PFRT program on a 6-million-pound engine. A half-billion-dollar program will support about 370 tests. A 6-million-pound-thrust module for a billion-dollar development will mean about 750 tests.

There is one other point concerning the part the engine will play in the recovery systems, and this is reuse. We think that the reuse and engine life history are going to have to be developed on the engine we select. We, in turn, feel that a substantial number of firings are also going to be required to demonstrate the actual reuse capability or the engine lifetime for recoverable vehicles. Therefore we conclude that we would prefer a module lower than the 6 million pounds; we would prefer a module in the range of half a million to up to 2 million pounds, depending on whether we are dealing with single-stage or two-stage vehicles, somewhere in that range.

Figure 16 is a schematic drawing of the air augmentation system. RENE means rocket engine nozzle ejector. We in the Martin Company have been looking at air augmentation for several years. The principle, basically, is we are introducing secondary air, mixing it with the rocket engine exhaust, extracting its energy, transferring momentum, integrating the momentum or thrust on the outer shroud. We are not postulating burning other than what the fuel-rich gases may yield. So that in concept we have selected the simplest system which we hope to be able to reduce to the lightest weight and not get into a reliability problem right from the start.

Table III shows another aspect of this augmentation. The ducted rocket experiments or analysis were made over a decade ago. The differences between the approach we have taken in our ejector system are that the secondary air flow to the propellant flow is in the region of, say, 2 to 5, and the ducted rocket was in the region of 20 pounds air to 1 pound of rocket exhaust. In turn, our test data to date have shown that we are getting supersonic exit velocities, whereas the ducted rocket was essentially subsonic exit. We are utilizing a divergent shroud for mixing, whereas the ducted rockets were essentially analyzed on constant-area basis.

Our test model is shown in figure 17. This test model was tested more than 2 years ago. Test model design is in progress right now and tests will be made in the near future. This model is shown installed in the Tullahoma Tunnel. It was a 3,000-pound lox/RP engine. The tabs seen in the figure are vortex generators and were one part of the test objectives.

Figure 18 shows some of the results of these early tests. Effectively the problem, as expected, lies with the length of the ejector and thus the efficiency of mixing. The test data shown here are for an augmentation factor. The total thrust out of the augmenter and the rocket divided by the rocket thrust gives the augmentation factor. In this figure the thrust ratio is plotted against the secondary air flow divided by the rocket propellant flow, the mass ratio.

Ejectors having three different length-to-diameter (L/D) ratios have been tested, two with and without vortex generation. The nozzle ejector with L/D = 1.75 has essentially little augmentation.

The vehicles we are concerned with incorporate L/D ratios of about 5 for a single engine; these data are for Mach 2 at 40,000 feet, and are corrected for inlet momentum losses. We are getting an augmentation factor of about 1.5- to 50-percent improvement.

Figure 19 shows the variation caused by a change in ejector L/D ratio with respect to the modular engine concept. Here there is a geometric transformation of the single engine to multiple engines; the ring engine gives the shortest ejector. It is clear that with modular engines we can shorten the duct length.

Figure 20 shows the vehicle concept incorporating this augmenter; we call this Renova. This is a hydrogen/lox vehicle. The rocket engine is plotted. This configuration has 750-pound-thrust engines, with about 36 of these in the ring. For a single engine, the L/D ratio through this augmentation system is 5.

We are postulating the nose fairing over the payload and looking at the improvement we can get in normal shock recovery with body compression.

Figure 21 gives an idea of how long the augmenter can reasonably work in the atmosphere. This is a plot of altitude against time; we are operating the augmenter nearly to 150,000 feet. Note that we have bent the trajectory so that the maximum q condition is about 1,700 psf. This is about twice the usual value for rocket vehicles; the penalties for air loads were introduced into the Renova vehicle.

The remainder of the trajectory past approximately 130,000 feet is all rocket. We have postulated shutting down engines and utilizing the duct area for high area ratio and high specific impulse after we have ceased using the ejector.

Figure 22 shows a performance map of the ejector operation. This is a plot of the specific impulse and thrust against altitude. We have used a reference module having an area ratio of 300 to 1. (See fig. 22.) If it were altitude compensated the area ratio would go from about 23 to 300. This would be comparable, say, to all-rocket vehicle thrust starting at about 30 million pounds.

After we lose static augmentation beyond a Mach number of 3.5, we then are traversing lines of constant Mach number on the trajectory shown in figure 21.

The peak specific impulse is of the order of about 850. Burning time was about 100 seconds to approximately 130,000 feet.

Let's look at the performance effect on the Renova vehicle. Figure 23 shows the specific impulse plotted against burning time for the Renova and the all-rocket single-stage vehicle. The basic reason that the vacuum portion is a little higher for the Renova is that it does have a larger diameter and larger base. The effect is of an integrated specific impulse. Without going into details, weight statements, and so forth, I will say that the weight of the ejector is about balanced off by the gain in performance in the vehicles that we have analyzed so far. We do see weight improvements in the ejector system, both in inlet design and perhaps in additional burning. The ejector I have just described is about paying for itself in a single-stage vehicle. However, we think that we have an improvement and that the air augmentation scheme should be further pursued in an advanced technology program.

Figure 24 compares the summary cost effectiveness of the Renova vehicle against the single-stage and two-stage vehicles. In general, the Renova vehicle requires more development. A varying refurbishing factor on the Renova due to the acoustic and heating problem gives rise to the slant lines; we are studying this now and have not settled on a number. Essentially Renova is competitive with the single-stage vehicle as it stands. We feel that with improvements we can extract more performance and bring the air augmentation scheme ahead of the all-rocket single-stage vehicle.

Our conclusions may be summarized as follows:

We feel that high-chamber-pressure technology should be pursued. Our reason for wanting high chamber pressure is to get high area ratio at high altitude. High chamber pressure and high area ratio are vital to any concept of the single-stage lox/hydrogen vehicle. Without technology in this area we feel we can't contemplate single-stage configurations. The high chamber pressure or the high area ratio is less of a requirement in the two-stage vehicle.

Relative to air augmentation we feel that the approach we have presented should continue to be pursued and that investigations of inlet matching and burning should be made to improve the performance of air augmentation.

We feel that, in addition to gimbaling or secondary injection, throttling should be investigated both for vehicle control and for the benefits of low load factor that we may require for payloads and certainly for single-stage vehicles.

Relative to reuse and reliability, we have proposed utilizing a high number of modules in order to get a single module so that we can get a high number of tests in the development program. This high number of tests can give us very good data on reusability of engines.

Altitude compensation is a problem of how it works and that it will work, and not so much its efficiency. We require it, however, to get high area ratios at high altitude to get this performance. The actual efficiency contribution in the atmosphere we believe is small.

TABLE I.- EQUIVALENT VALUES - LOX/LH_2

	TWO STAGE	SINGLE STAGE
ALTITUDE COMPENSATION (INCL ϵ CHANGE)	7 SEC I_{sp}	10 SEC I_{sp}
(OVEREXPANSION EFFECT)	1 SEC I_{sp}	1 SEC I_{sp}
MIXTURE RATIO (5 TO 7) STRUCTURAL ONLY	5 SEC I_{sp}	9 SEC I_{sp}
COMBUSTION CHAMBER PRESSURE & MIXTURE RATIO (1000 TO 3000, 5 TO 7)	10 SEC I_{sp}	19 SEC I_{sp}
1 SEC I_{sp}	17,000 LB ΔW_1 3,460 LB ΔW_2	15,200 ΔW

TABLE II.- JET DEFLECTION REQUIREMENTS

CONFIGURATION	STAGE I PROPELLANTS	STAGE I ENGINES	MAXIMUM δ (DEG)
T 10 E E	LOX/RP-1	F-1A	3.97
T 10 E E	SOLID	SOLID	2.92
T 10 E E	LOX/LH_2	HP PLUG	2.14
S 10 E	LOX/LH_2	HP PLUG	2.29
P 10 E E	LOX/LH_2	L-6H	1.25

TABLE III.- SYSTEM CHARACTERISTICS

1. MAX. $W_s/W_p \approx 4$	1. MAX. $W_s/W_p \approx 20$
2. SUPERSONIC EXIT	2. SUBSONIC EXIT
3. DIVERGENT SHROUD	3. CONSTANT AREA SHROUD

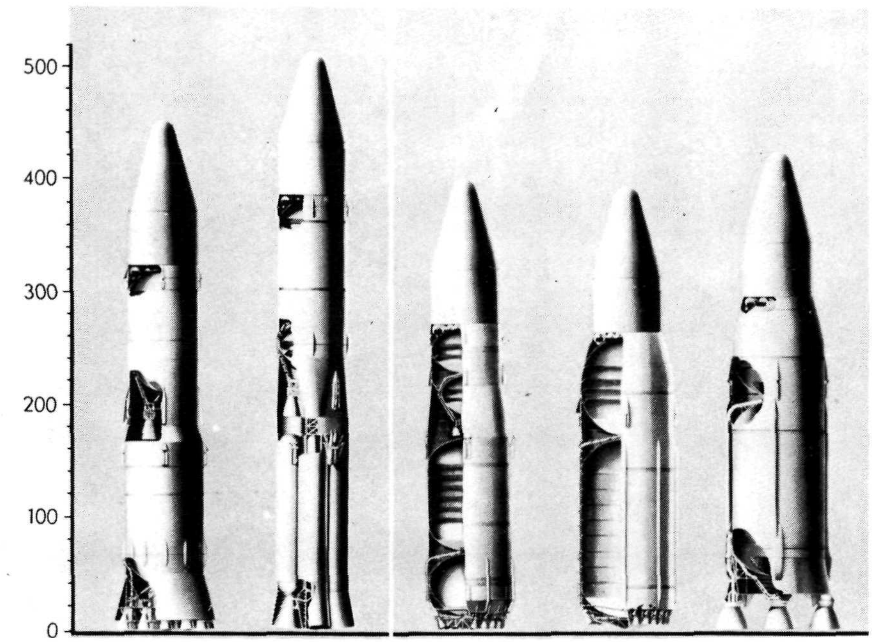

Figure 1.- Selected configurations.

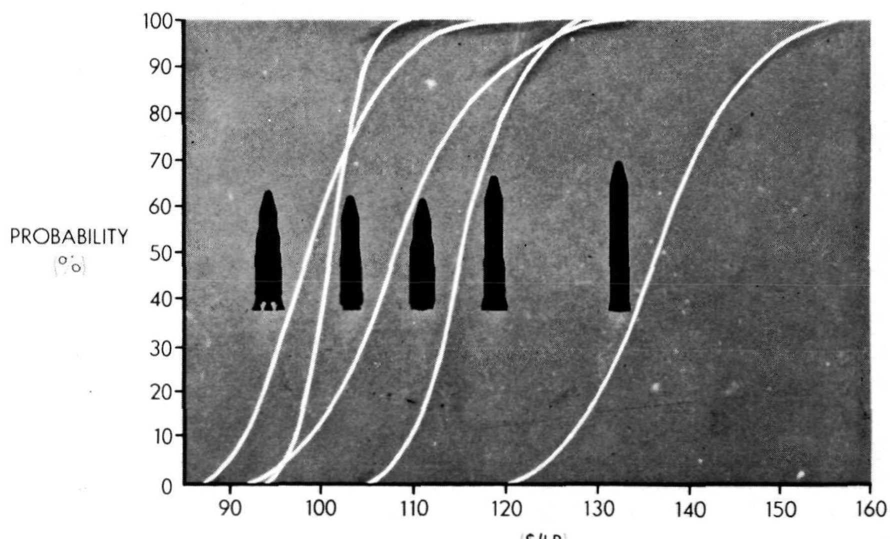

Figure 2.- Cost effectiveness - high mission level.

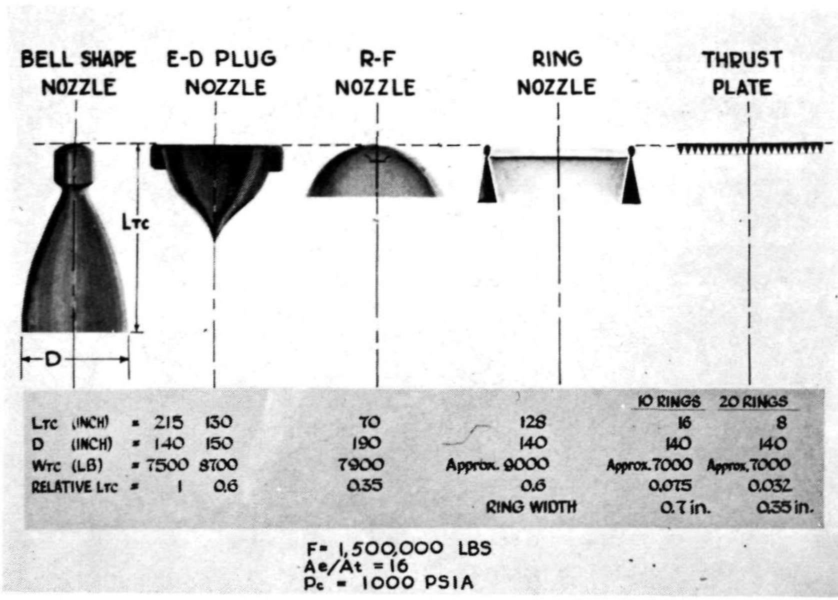

Figure 3.- Advanced rocket engines.

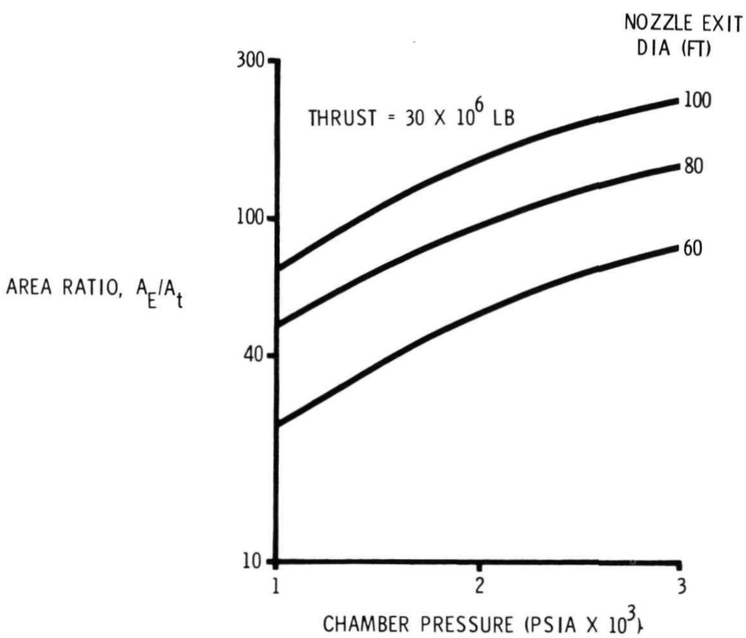

Figure 4.- Area ratio.

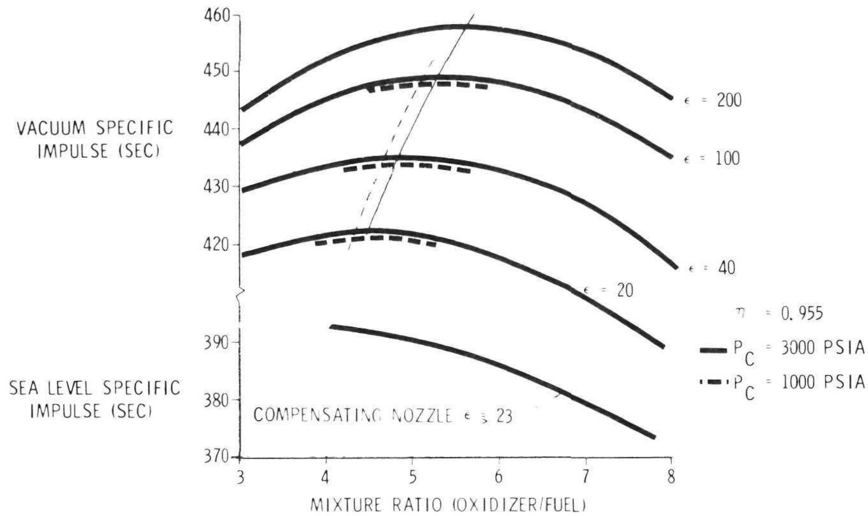

Figure 5.- Lox/LH_2 specific impulse plotted against mixture ratio.

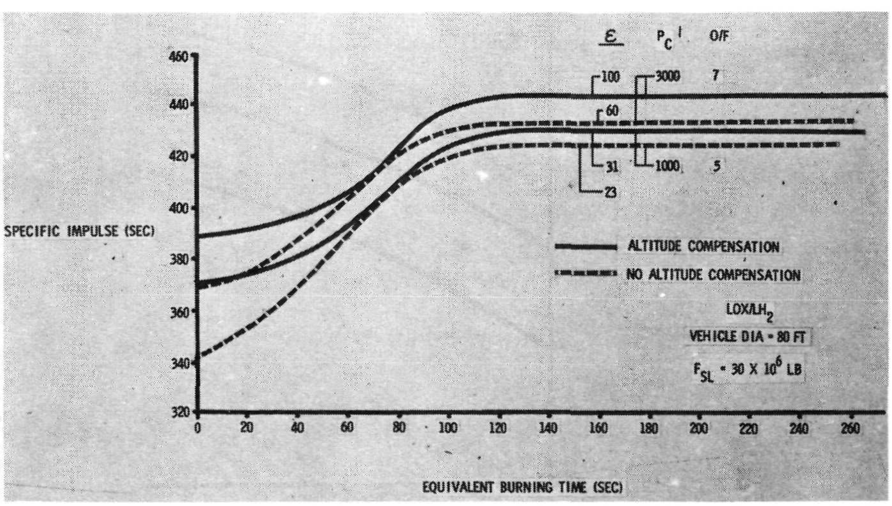

Figure 6.- Specific impulse.

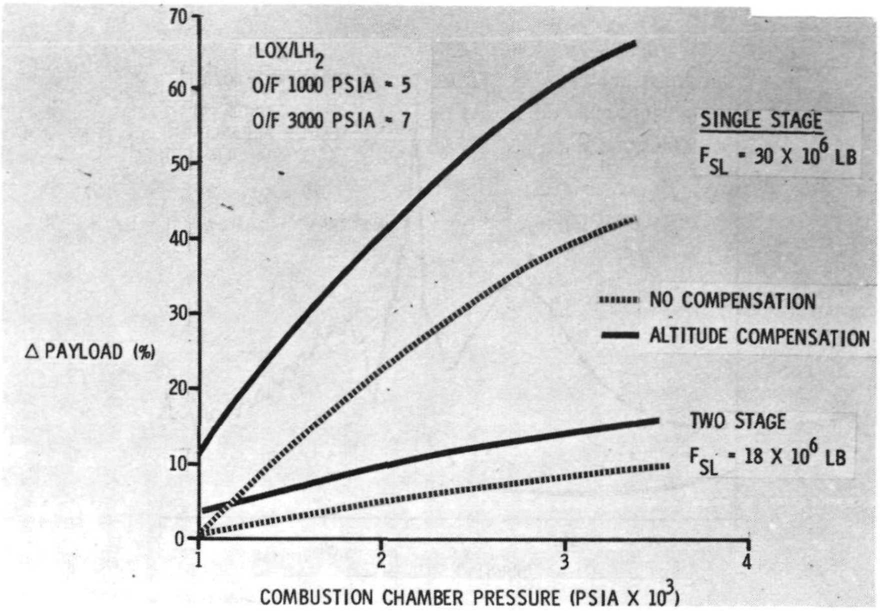

Figure 7.- Payload plotted against chamber pressure.

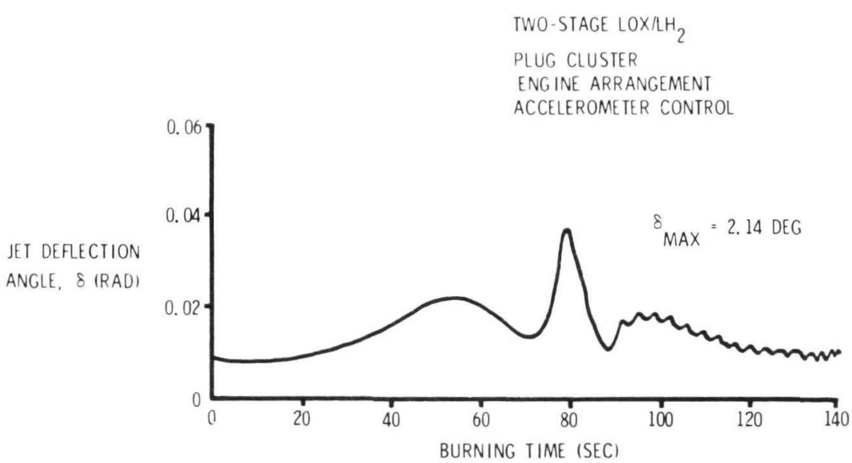

Figure 8.- Typical TVC duty cycle.

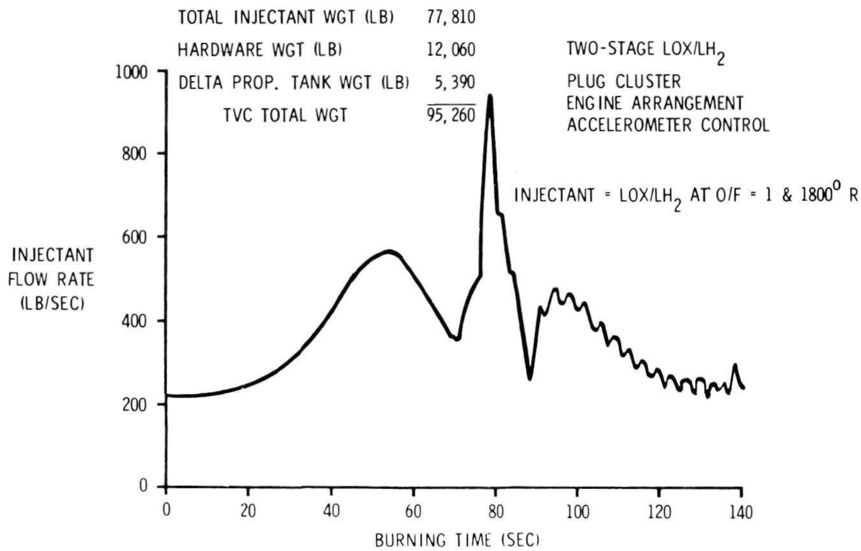

Figure 9.- Secondary fluid injection.

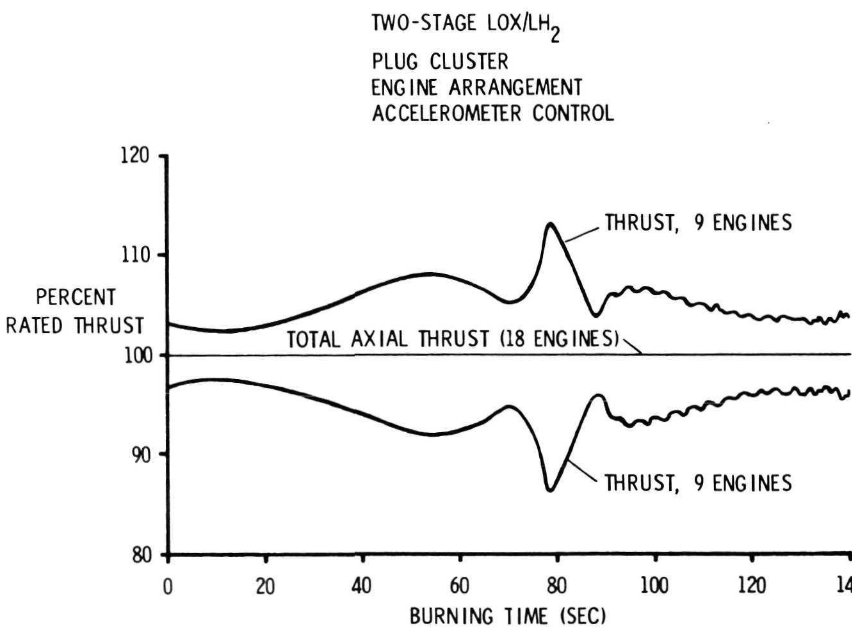

Figure 10.- Differential throttling.

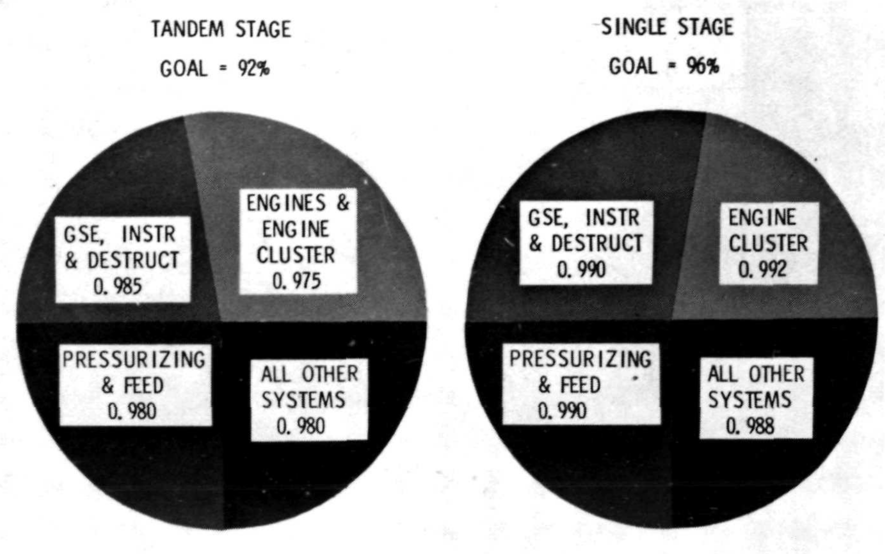

Figure 11.- Reliability requirements for Nova vehicle systems.

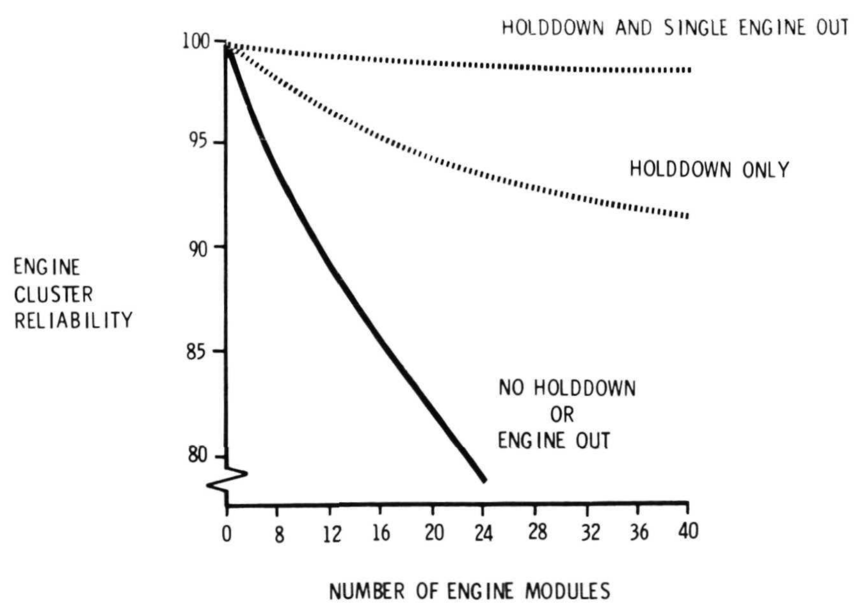

Figure 12.- Engine cluster reliability for different operating concepts.

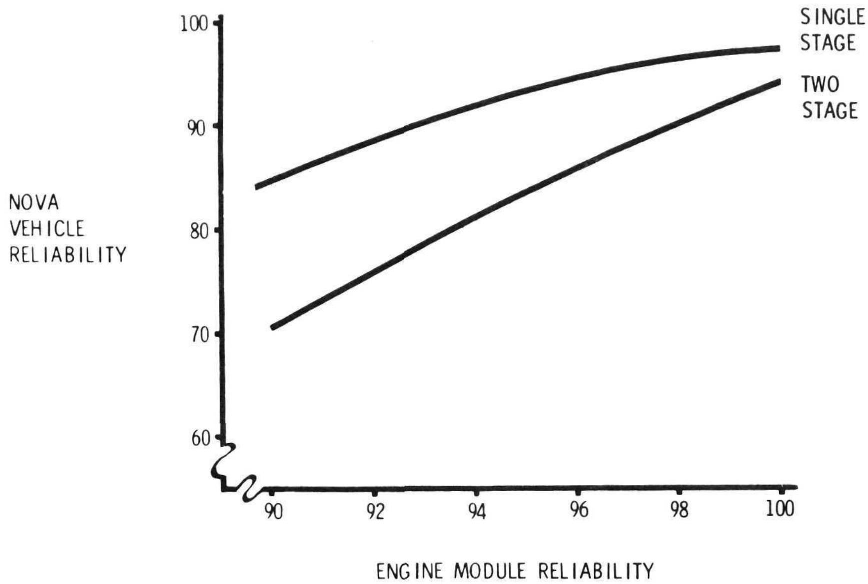

Figure 13.- Sensitivity of vehicle reliability to engine module reliability.

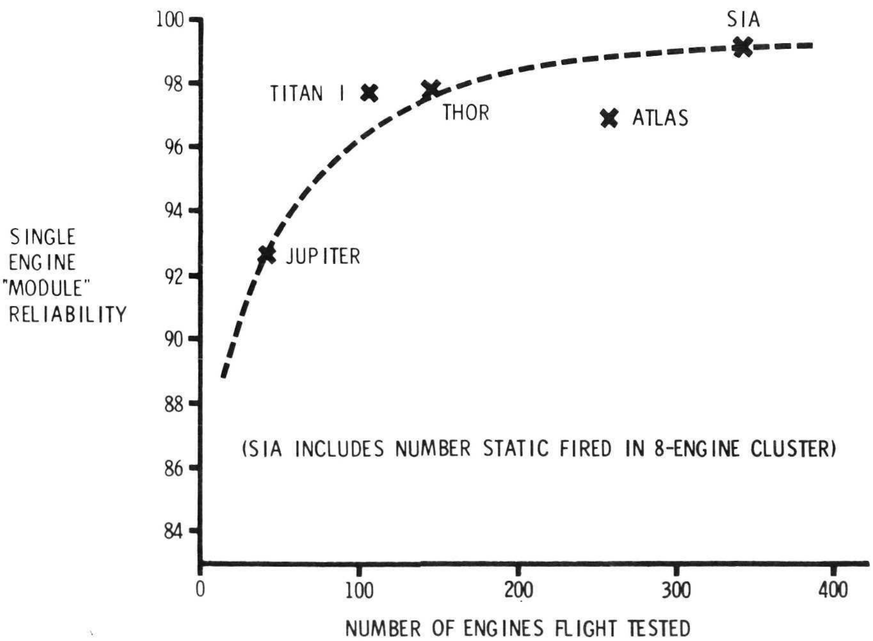

Figure 14.- Flight reliability history for 150K engine "modules."

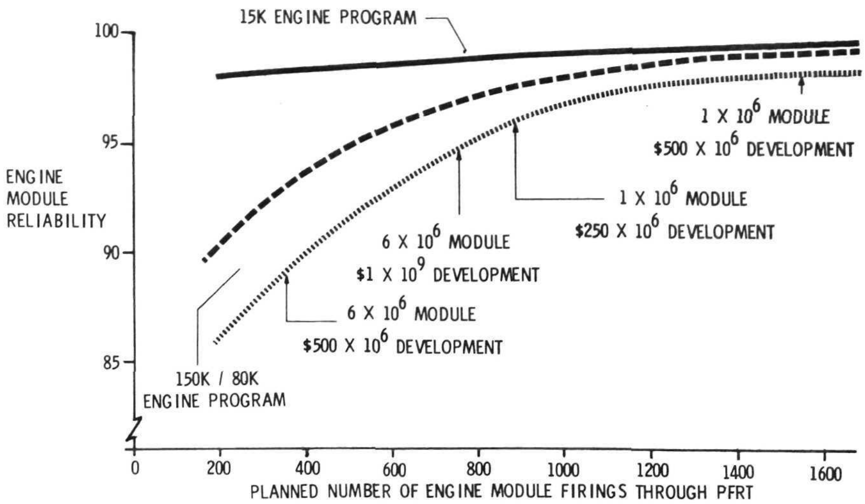

Figure 15.- Engine module reliability growth during development.

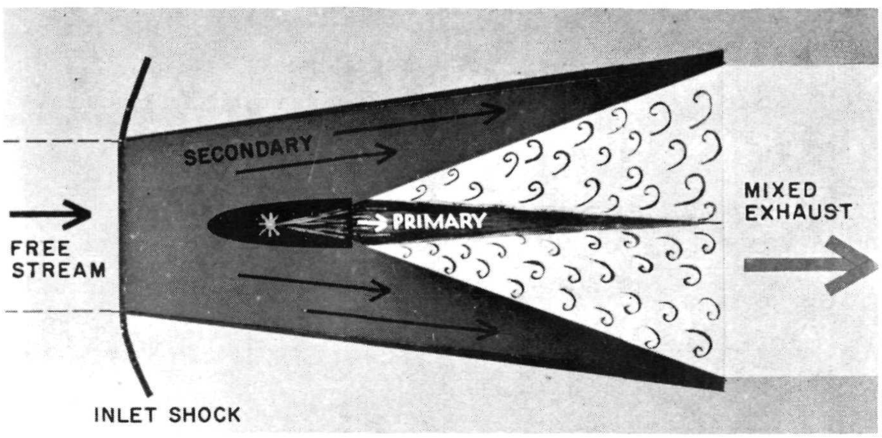

Figure 16.- Rene system.

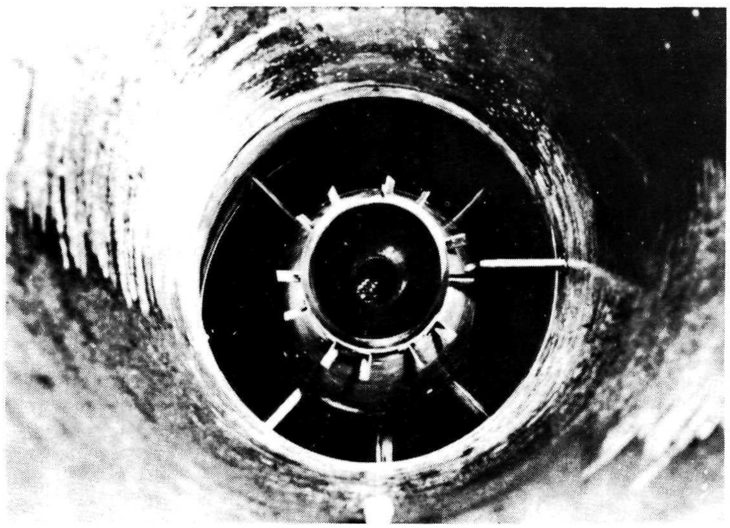

Figure 17.- Test model.

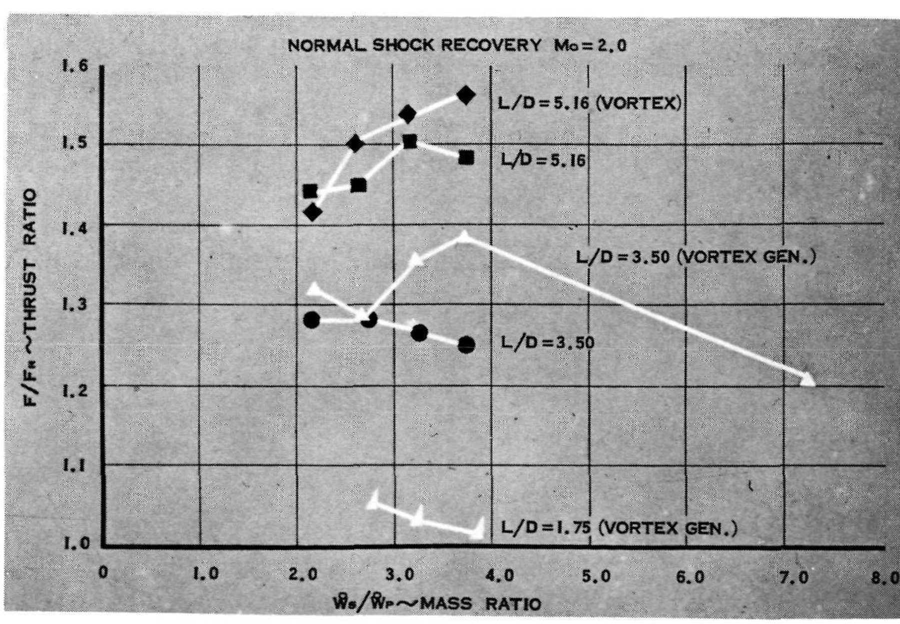

Figure 18.- Experimental nozzle ejector.

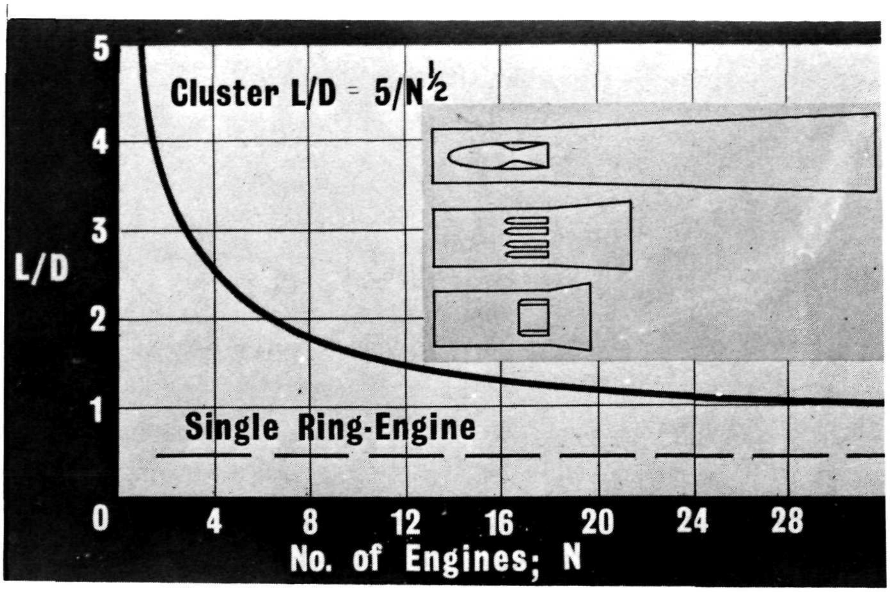

Figure 19.- Ejector L/D variation.

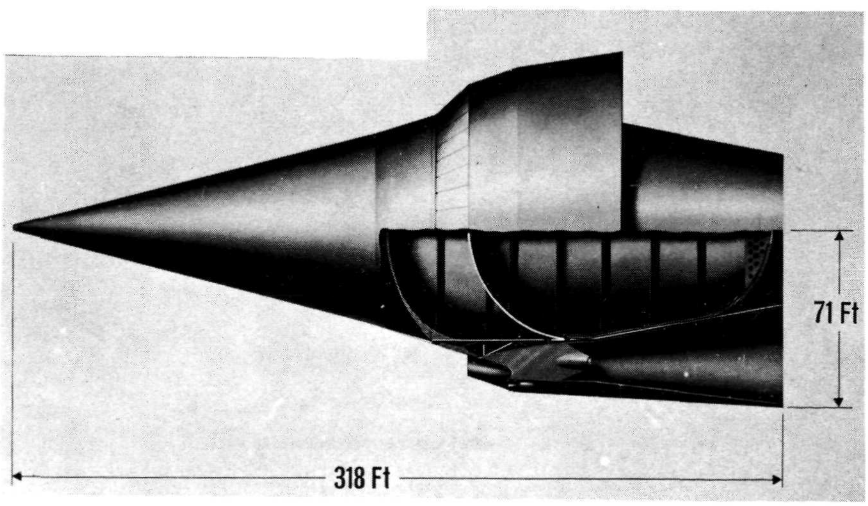

Figure 20.- Renova.

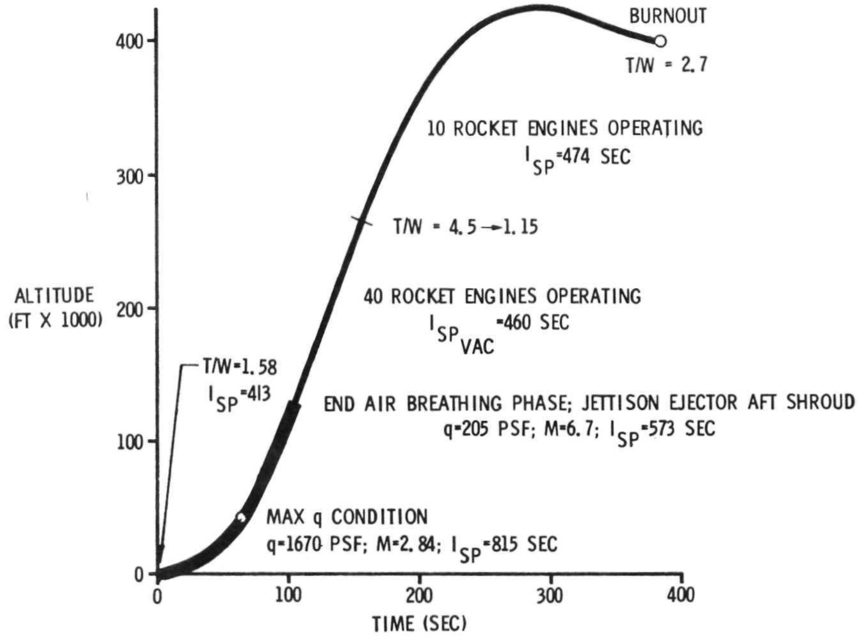

Figure 21.- Typical ascent trajectory.

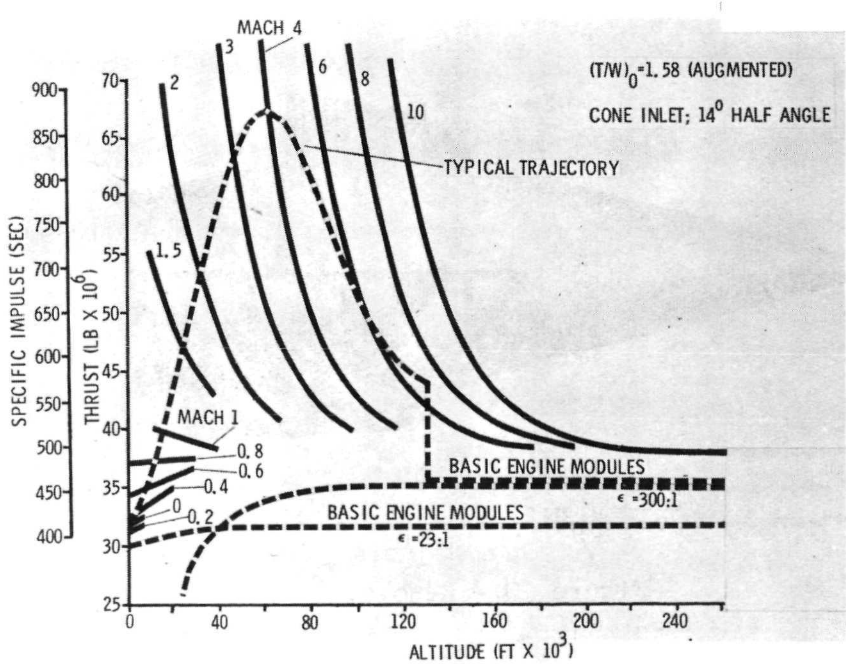

Figure 22.- Renova - estimated Rene performance.

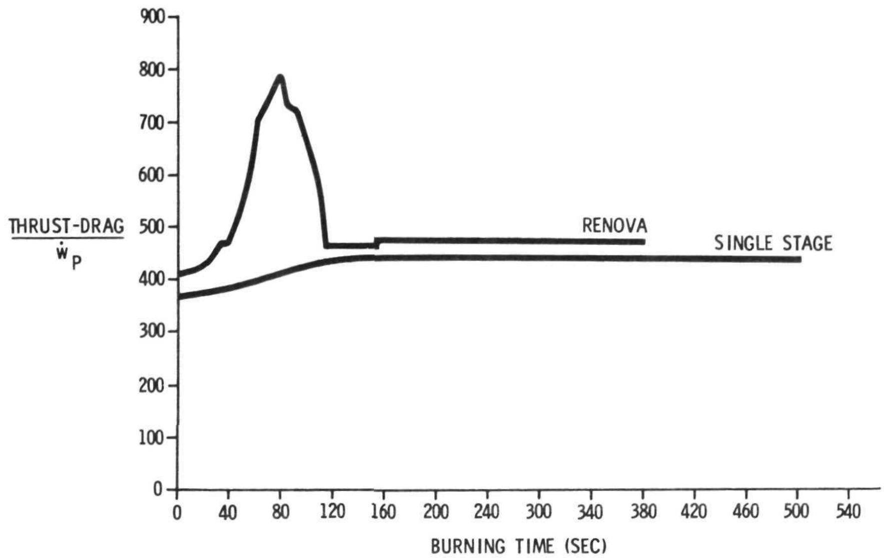

Figure 23.- Effective specific impulse.

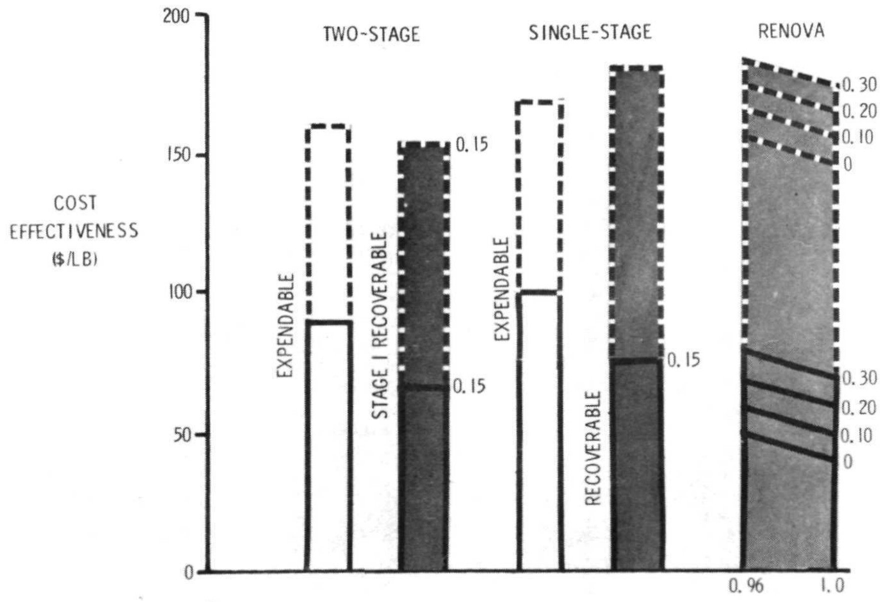

Figure 24.- Maximum Nova mission.

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5. SUMMARY DISCUSSION

MR. WILLIAMS: One additional point I want to make. As you have seen, we have concentrated our discussion of Nova in the propulsion area on chemical propulsion. Lest we be misunderstood, we are in our Nova studies considering nuclear propulsion. Within the Nova studies that are underway at General Dynamics and Martin, we are, however, excluding nuclear propulsion for, if you will, orbital transportation, for hauling cargo to orbit and those missions which originate or at least start from orbit.

For quite a few missions or configurations nuclear propulsion definitely has tremendous applications. For space probes, where you would want very high velocity increments, all configurations on which we are doing design or trade-off work are configured so that they are sized to accept a nuclear third stage of the heat-exchanger variety and hydrogen as a working fluid.

The man-Mars mission, the payload, the planning, and the programing of this, is based on nuclear propulsion, at least for the manned landing activities; the vehicles that we have under consideration, are sized so that they can carry the very low density hydrogen into orbit that might be needed for the working fluid for nuclear-Mars ships leaving an orbit and going to or in the vicinity of Mars.

So we do have nuclear propulsion included in our Nova studies. I might also add in our past Nova studies, that is, vehicles after Nova, or at least in a time frame after the Novas that we have under consideration, we are looking at a little bit more ambitious application of nuclear propulsion. But it is not only chemical propulsion that we have under consideration. I want to make that point clear.

QUESTION: It wasn't obvious to me from the Martin paper why we should pursue this air augmentation. A number of people, including myself, spent quite a bit of time a decade ago looking at this thing. The results from the Martin Company don't convince me that I should go back into this rocket business. Would the Martin people like to comment on this?

MR. YOUNGQUIST: I hope I made myself clear that I wasn't making large claims for it. I feel that relative to the technology propulsion program we have in the country we can afford to take a better look at this. Relative to the engine concepts we have today, which in effect are the geometric effects, it looks like we might have the possibility of configuring a good augmentation scheme. I believe there are some serious unknowns about the theory and how air augmentation works. I think we should understand this. I am not making a vehicle recommendation. What I am doing is making a technology recommendation for more exploration in this area.

MR. WILLIAMS: Maybe I could make a comment along these lines. The Nova studies have been going on now in-house and with smaller contracts than those we have presently in progress for about 3 to 4 years. We initiated activities which finally led to the present GD/A-Martin contracts a little over a year ago. Then the task which was defined to Martin and GD/A was, basically, at least the one they bid on when the request for proposals went out, that of a preliminary design of an F-1/M-1 Nova which would provide the U.S. with the capability of direct manned lunar landing and return. As evident from the material just presented, we have gone a little bit past designing or defining a system which would do just that job. We have gone through an evolution over the last year. We are looking a little bit farther downstream time wise. We are looking at greater advances in technology, if you will. In fact, during the last 4 or 5 months, and probably for some months to come, we are going to be looking at systems which are not conventional by any stretch of the imagination as we know rocket systems today. We want to make sure that we are not overlooking any major advancements, not necessarily breakthroughs, in the area of advanced space systems before we continue what we hope will be a narrowing-down process ultimately leading to the definition of one Nova system.

So we feel at this time that air augmentation may hold certain promise and deserves some investigation. Again, the degree I think is one of the subjects to which we need to address ourselves.

MR. THOMPSON: At the present time we have two studies going, one which is a continuation of the Martin effort, in conjunction with the Air Force at Edwards Air Force Base, which will be done at Tulahoma. Initial firings of this configuration will be soon. It is a 12-engine module configuration, annular, lox/RP, each having 500 pounds of thrust. The objective is to find a minimum L/D to get sufficient augmentation.

The other study that we have is a technical study which was recently awarded to The Boeing Company. In this study we are going to be delving into the heart of the matter, mixing phenomena, in particular, to look at the viscosity coefficients required to calculate adequate performance and other aspects of it.

I think that perhaps Mr. Williams could qualify the degree of thought that is being put on the single-stage orbit vehicle.

MR. WILLIAMS: We have, of course, looked at advanced systems. There are certain definite advantages to a single-stage system, particularly when we talk about reuse: the turnaround time, recovery operation, the logistics, the development itself. One of the figures in the paper by Mr. Youngquist showed the cast sensitivity. This I think can be derived back to the weight - how well we can design, how much we can reduce mass fraction. The I_{SP} sensitivity must be taken into account, in case we cannot squeeze out of an engine, an advance engine, all that we want. It is very sensitive. It is not quite, but almost, too sensitive to put too much effort in at this time.

It is hoped that this air augmentation might be a way to minimize, if you will, the sensitivity and get a steeper slope, at least as plotted on the curve that was presented. Mr. Youngquist indicated that right now the air augmentation seems to be carrying its own weight, but that is about all. Maybe with certain new applications or modifications, we might be able to improve this. We feel that before we throw it out once and forever, a certain amount of investigation needs to go into it. In propellants, for example, we have looked at fluorines, at storables, at quite a few things. We want to continue this narrowing-down process but we do not want to overlook any promising elements that we might integrate into Nova.

MR. PAUL: I am wondering whether the two-stage contractors have looked at the problem of sloshing, the suppressions of force, because the diameters of our tanks are increasing up to around 80 or 100 feet. I am wondering whether the stationary and tie-slosh baffles are still a good thing to use, or whether the weight might increase excessively.

MR. WILLIAMS: This has been taken in consideration - we hope adequately. The weights, the cost, the performance, and so forth, have been quoted on the basis of an attempt to assess the slosh problem in the control, as well as the structural loads, moments, and so forth.

MR. KALITINSKY: We have attempted to take slosh into account in some of these large-diameter vehicles as well as some of the odd shapes. I must say that it requires a good deal more analysis and some testing before we can be quite sure. All we can do now is simply make an estimate, a guess, as to what it is.

MR. WILLIAMS: We have attempted to put in penalties, if you will, from a weight standpoint, for slosh suppression. What precisely this weight looks like, whether it is mesh, ping pong balls, or what have you, we aren't quite certain. We feel we have penalized the system sufficiently to solve this problem once we arrive at what the best solution is.

MR. YOUNGQUIST: More particularly, we have looked at the multicell versus single-cell tanks, multicell tanks being orange-peel configurations which in effect might be installed baffles as compared with cylindrical tanks or ellipsoidal tanks which have ring baffles around the inner surface. I think what we have looked at, and it depends on the type of control system you use, can solve the problem in any given case. As a matter of fact, the solutions, even though diverse approaches on how you install baffling, still seem to be a toss up. We think we can handle the slosh problem.

MR. CONNORS: In talking about some of the advanced nozzles we seem to overlook the fact that we are going to require venting for ED-HFO nozzles. This will be an important consideration in the structural concept of the vehicle.

I think also that probably in the Nova study it might have been well to have done differently and included a freedom in configuring the vehicle, perhaps using the advanced nozzle concept with current engine cooling technology - F-1/M-1 engines configured to fit into a conventional nozzle. This would have solved the base heating problem as well as the altitude compensation.

MR. KALITINSKY: We have looked specifically at M-1 as a module and we are also looking at F-1 as a module. There are considerable gains to be obtained from high chamber pressures. Since the present studies are examining the avenues which will provide the greatest economic pay-off, we have to keep showing the gains that can be achieved by going to higher chamber pressures.

[redacted]

COMMENT: High chamber pressures are way down the street.

MR. WILLIAMS: Nova, maybe. This is, if you will, the dilemma that we are in.

COMMENT: The figure used at the recent Lewis conference showed a cost of three categories of vehicles on some time scale. Class II was a number of years downstream of Class I.

MR. WILLIAMS: If you allow the advanced nozzle to fit into the same time bracket as category 1, I think you may have picked up a significant gain. Also some evaluation of what the heat shield requirements would be when you are talking about clustering 16, 18, 20 engines should be made.

For the sake of those who didn't participate in the Lewis conference perhaps these classes, as we have called them, should be clarified. Class I are state-of-the-art vehicles. Class I vehicles utilize existing or in-development elements of systems as basically they are designed or planned at this time. Class I, we feel, could be available were a decision made in the next year or two to pursue such a program into the early 1970 time period, that is, operationally available through a flight development program by, say, probably not earlier than 1973, maybe 1974.

The Class II vehicles are typified by most of the material previously presented. We feel it provides us time for certain advancements in the state of the art of propulsion as well as structures, control, and so forth, and would be vehicles that would be available in, say, the 1976 through 1978 time period.

Class III vehicles are very advanced, with quite sophisticated air augmentation and would fall in the late 1970 or early 1980 time period. Air augmentation falls into that category.

In the Class I, we feel that somewhere in the order of about $5 billion to $6 billion will be required to get through a developmental program and get supplies in the pipeline for an operational program. One has to then go back and see at what rate this $5 billion to $6 billion can be accumulated.

Knowing as we do the budget situation, we are not going to accumulate much in the next 2 to perhaps 6 years. So this doesn't give us much leeway and adaptation of current hardware or planned hardware. It does, however, give us some.

We have looked at the adaptation of F-1's and M-1's which definitely will cost us money, will require certain feasibility demonstrations, and at the somewhat elaborate development program over and above the elements themselves, the F-1 or up-rated F-1 as we have used, or M-1's, or a rubberized M-1, to see how we might modify these basic elements into a more sophisticated or exotic system. This we felt from a timing standpoint primarily, and timing in that the requirement to accumulate the money or wealth of resources to do this, really pushed it into a class II time period, and hence we compared the system with those other systems which could also be available at that time.

 [redacted]

To us class I is something that we can start on, maybe not today but in the very near future with a very positive definition or set of milestones to advance toward. No particular proof of feasibility would be required to pursue either of the Class I systems.

QUESTION: Would someone comment on the venting of the expansion deflection nozzles?

MR. STREETMAN: This is something I meant to point out in the presentation but did not. We believe bleeding or venting of the center body on the ED would seem to be required. It seems to be indicated by all the available test data. However, we have not seen any definite data pro or con on the plugs. We really can't make any firm statements about that.

MR. GILMORE: There is very little test data available on the plug nozzle which we have spent more time on than the ED nozzle. However, as far as we can determine, there should not be any venting requirement. This still has to be verified by real test data, however.

MR. WILLIAMS: You are speaking specifically of the plug?

MR. GILMORE: Just of the plug. The ED nozzle, as far as we are concerned, will require venting of some kind.

MR. DE MARS: I believe we are presently funding a study to give us the answers which will back up Mr. Gilmore's statement regarding venting to the base plug nozzle. On the ED, I think we can be fairly well assured that we have to have some type of venting. Whether it be air venting or turbine exhaust gas supplied from the gas generator or tapoff system is still in question. But I do think we have to have some kind of air replenishment.

MR. THOMPSON: We all agree that venting is required on ED. What would be the best type of venting to accomplish the highest percent of base pressure recovery?

MR. CONNERS: I will touch on this in my paper. I think the requirement is that you provide a near ambient pressure environment at the point where you start the external deflection. The question is how you choose to do this; there can be a lot of flexibility. You can do it with a hot gas as long as you achieve the near ambient pressure environment. But this is what will control the off-design expansion in the performance of the nozzle. I think it is a significant structural problem and you can't just say, "Yes, we will vent." It seems to me that it has to be an integral part of the design. You have to account for it. If you vent to the external screen, you have to provide scoops, and you have to provide the structural components. If you use hot gas, also structural problems exist as to how you are going to duct it out, how you are going to cool it.

All I am pointing out is that in any superficial treatment of the propulsion system we should look at the whole picture and look at the details of how we would propose to vent it. I think when we run comparisons, too, with the existing technology, we ought to look and try to assess what will be required in the way of a base heat shield. This, too, is a performance loss if we have to go to heavy weights. So there are many components to the argument for advanced nozzles. Performance is only one of them. Base heating, development, the whole gamut, we have to go into these.

MR. WILLIAMS: I think one of the considerations, in the GD/A work and the selection of RP and the expansion ratio, was to minimize this particular problem.

MR. KALITINSKY: That is correct.

MR. SLOOP: Many questions have been raised here which we hope will be covered further in our discussions. One that wasn't mentioned was the question of RP versus oxygen, which I am quite interested in; this question has many facets. We have a lot of experience with RP; on the other hand it would be very desirable if we could use hydrogen. There are the questions of single pump versus multiple types of thrust chamber and nozzle, the configuration, including air augmentation.

SESSION II

ENGINE SYSTEMS

Chairman: Hermann K. Weidner,
Marshall Space Flight Center, NASA

6. INTRODUCTION

By Hermann K. Weidner

Marshall Space Flight Center, NASA

The vehicles considered by Martin and GD/A were discussed in session I. They all look different from what we have been used to in the past, and I think some comments as to this classification, class I, II, and III, shed some light on that. We are addressing ourselves more toward the class II and class III, namely, for the purpose of searching our minds - what we should do today in preparation for things to come.

Nova certainly will be a vehicle much more powerful and larger than the present Saturn, and also, I think, should be more advanced in its technology. It should be more sophisticated and make use of things which we don't dare to use today, where we don't have all the knowledge.

One area apparently becomes very obvious, and that is that, as you look at the layouts, you do not necessarily recognize the engine any more in the classical sense as some package which you can treat separately; it has to become an integral part more and more of the overall vehicle.

This is more true, I think, with the more advanced sophisticated concepts where we try to apply all sorts of tricks to increase our efficiency, whether this is air augmentation or whether this is compensating nozzles. All these things require a close-knit working team between the stage or the vehicular people and the propulsion or engine people.

In keeping with this, we have asked all the contractors to be present to give them a chance to be exposed as much and often as possible to our common problems and thoughts, to teach them, and to show them that we somehow must work much closer together in the future than we have in the past.

We have asked the three engine contractors who have spent considerable time in studying advanced engine systems of the kind we are interested in and have had large groups of people working on these areas to give their present thinking and their present status of thoughts on how they think the engine system should look in the environment of these vehicles.

We have basically asked them two questions.

(1) Which propulsion concepts would they recommend to be most applicable to our purpose from the vehicular view point and from the engine system view point?

(2) How would we go about developing these? As these engines become larger and larger, the economics of development is something which is of paramount importance. From past experience, we know that hundreds and hundreds of engine development tests will have to be performed during the development phase of the program. As we come to these larger vehicles, possibly 20 million pounds, the

propellant cost alone, to run a minute, is staggering. It is something of the order of half a million dollars.

We have to find approaches to build these units, possibly by sectionalizing or a modular approach, or whatever the answer, to get to know our hardware so that as many of the developmental problems can be solved on as small units as possible.

7. INTEGRATION OF ENGINE AND VEHICLE

By Don M. DeMars

Marshall Space Flight Center, NASA

I would in this presentation like to emphasize the importance of recognizing the environment around an engine system. In doing this, I feel we can better come up with an optimum system.

In the past, the question was asked, "Can we build a specific engine?" We met the challenge and we then built it. Later on we would worry about the vehicle meeting its payload requirements. We progressed a little further and investigated the impact the vehicle has on the engine system. Now, I feel that we should progress still further and consider a few other areas.

Figure 1 shows the specific areas which I would like to discuss. They have been categorized into eight areas: Engine systems is the core, and branching out from it are the other influential areas: Facility Considerations, Ground Support Equipment, Vehicle Requirements, Development and Test Operations, Funding Restrictions, Launch Operations, Mission Requirements, and Technology Potentials.

First, let us discuss Facility Considerations and a few points which I feel should be used as "food for thought" for the following engine presentations. The points listed in figure 2 are by no means all the points which can be made under facility considerations. However, these are some of the main ones.

(1) Hardware. From the facilities standpoint, small, easily transported hardware is desirable. Dollars per development test will be much less on a smaller system. Hardware size also affects the engine system and has an impact on the vehicle requirements. The vehicle has to have a high reliability. Many small propulsion modules supposedly degrade the vehicle reliability. This can be prevented by accomplishing a sufficient amount of testing to increase the module reliability significantly or have provisions for engine out in the propulsion system.

(2) Vehicle Simulation. We should build an engine system which would be capable of being tested on a facility that would very nearly simulate the vehicle so that we can eliminate a lot of our vehicle-engine problems early or during the engine system development program. This will reduce the problems which might appear later on during vehicle system testing.

(3) Acoustics. The acoustics area which is a minor point, basically influences land acquisition and test stand orientation.

(4) Methods of Testing. On these new systems being proposed, the methods of testing may not be similar to the previous testing method of engine systems. I believe that Rocketdyne, in one of the following presentations, will indicate that they have done some research in this area. They show how they can use a

segmented ED nozzle as a deflector and how they can best eliminate facility hardware costs and put their money into engine hardware, thus realizing benefits in two areas.

Aspects of ground support equipment are listed in figure 3.

(1) Transporation and Handling. Engine size also has an impact on the ground support equipment as well as on facilities. We will have to handle these systems and transport them. Should they be large and require special conveyance; or of a size readily transportable on highways? These points must be considered.

(2) Engine System and GSE Program Phasing. An engine system can be built and the GSE can then be added. Maybe we should consider and investigate the impact of building the GSE in a parallel program or effort, so that we can determine the GSE requirements and attempt to phase them in with the engine system.

(3) Maintainability and Accessibility. The mission requirements would probably emphasize reuse. In emphasizing reuse we will have to maintain these engine systems and we will have to have accessibility to seals and the short-life items of the systems. When we consider development of a new engine system we will have to consider this point, and should also investigate the one of standardization. Why can't we build engine systems today on which we could replace a gas generator or turbine and not have to recalibrate the engine? - similar to a carburetor replacement on an automobile.

Figure 4 details vehicle requirements. These requirements were discussed fully in the previous papers on Nova; therefore I won't elaborate too much on these.

(1) Performance. Performance can essentially be brought about by high chamber pressure. You then return to the engine system and determine the effect high P_c has on the technology potential from the standpoint of whether this P_c level is realistic. Can we achieve altitude compensation to give the required performance? All I am emphasizing here is that every point which may be favorable from one standpoint has an impact on the engine system which, in turn, has an impact on the other areas shown. To determine the impact of one system on anything there must be an optimization study or study effort.

(2) Engine Installation. Here again the desire is to reduce the hardware weight, such as in our thrust distribution structure between interface and engine. Many modules appear to be most favorable in distributing the thrust to the skin of the vehicle where we want it - whereas in one single module propulsion system a heavy conical thrust structure will be required.

(3) Reliability. I believe this point was sufficiently emphasized in the vehicle study presentations.

Some of the points to consider in development and test operations are given in figure 5.

(1) Contracting Methods. In the past when we released a contract for an engine system it basically went to one contractor. Is this the right way to go? Should we go out with different contracting methods. Since this will probably be one of the most expensive engine system programs undertaken thus far, maybe we should break the system into subsystems and have one contractor develop the hydrogen pump, another contractor develop the oxidizer pump, thereby making these systems independent of each other, and yet coming up with an optimum propulsion system utilizing the capability in each of the contractors to its fullest extent. Possibly another contractor would develop the nozzle and the associated combustion devices. This is an attempt to make the engine subsystems independent of each other, and yet maintain an influence on the system design.

(2) Component/System Testing. With the new high pressure systems under consideration the development of a high P_c thrust chamber may present a problem. By this I mean we are going to have to supply propellants under pressure to these chambers and nozzles for development testing and evaluation. To pressurize the propellants in the tanks and pressurized systems which feed the thrust chambers will be very expensive. Why not put this money into engine hardware and develop a flight configuration, not necessarily flight weight, iron horse pumping system and utilize this for your source of feed for thrust chamber testing? This approach may result in a lower overall cost and will develop thrust chamber performance data more applicable to the final design, along with supplying lucrative pumping data.

(3) Producible Hardware. This may also be a minor point, but worthwhile to mention. We would like to avoid long lead items during the predevelopment of an engine system so that if a large injector is damaged, it could be replaced without a significant delay. The materials for this hardware must also be available to prevent any program delays. Hardware would have an impact on size optimization.

Funding restrictions are shown in figure 6.

(1) Realistic Costing. Here we enter into the subject of overrun. This is always a touchy subject. I don't think it is the fault of anybody. The problem is insufficient research at the time of contract initiation. If we could have these engine contractors or NASA thoroughly evaluate this area from the various standpoints, we could come up with a more realistic figure on these programs and be able to quote these figures with a higher confidence level. This would benefit the overall program.

(2) Cost Schedule. This is an important area of some engine systems. You might have a peak requirement for money during the first phase of the program or you might have it at the latter phase. We should look at this more and attempt to obtain a mean curve so that we could grow with the economic curve of the NASA budget. This does not necessarily apply to engine systems only, but to the overall vehicle program. Maybe we should maintain a nice slope on the funding curve and program this into our overall mission.

Points to consider as far as launch operations are concerned are listed in figure 7. Martin-Denver is doing a study on launch and facility operations.

They have some input with reference to the vehicle and engine system. But a more detailed applicable analysis should be done on the engine system.

(1) Automatic Checkout. On the larger systems we would like to have remote automatic checkout. Work in this area would give us experience and background which could possibly help us to achieve orbital rendezvous checkout. If we can demonstrate methods of automatic checkout on the launch pad we can also apply these techniques to lunar bases and future space areas.

(2) Transient Characteristics. From an operations standpoint the launch personnel will have to push the button and hopefully the system will ignite, reach mainstage successfully, complete the holddown requirements and lift off with no more than 3 to 4 seconds of main stage propellants consumed. If we could have transient conditions, such as throttling, incorporated into the system, the engines could idle at 5 to 10 percent thrust for a few seconds. This would be time to assure that all operations are going fine; throttle could then be pushed and lift off would occur with possibly only 1 or 2 seconds of main stage propellants burned. This is a new approach. Maybe we are saving some propellants and maybe we are not, but from a safety standpoint it might be easier to shut down an engine at 10 percent thrust than full thrust. That still has to be investigated.

(3) Fail-Safe Conditions. Here again some effort has been put into this area but I think there should be more. When your automobile engine fails, you don't worry about getting you and your family out of the car safely. Why can't we feel the same way with large rocket engine systems? The point is this: when a failure occurs have a system that will shut itself down and eliminate the possibility of a catastrophic failure. This point is especially important when an astronaut is aboard.

Mission requirements, figure 8, were covered fairly well in the previous presentation.

(1) Schedule. We would like to assure ourselves of a realistic availability date for the engine system. If we are talking about 1975 to 1980, should we not start now to schedule programs and do work and technology to give us background prior to going into a development program? Most of the previous programs have been pressed by schedules. Maybe we should take a better look at today's proposed scheduling. Dollars spent today may yield more information than the same dollars spent under a tight schedule.

(2) Reusability. I mentioned before that we should have long engine life. Maybe we should concentrate more on eliminating failures during transition rather than during main stage. The firing or running time of the present systems appears to be about 120 seconds. Most failures in the past have occurred during ignition, transition, and shutdown. Maybe we should do a lot more work in the area of refining the reliability during these phases of operation.

(3) Logistics. We should know when we need the engines and how many we need. If recovery doesn't seem feasible, we would probably need many engines so we would want our production costs to be low relative to our development cost

and vice versa if recovery is practical. This has an effect on an engine system in its entirety.

Some of the points under technology potentials are given in figure 9. We want to know how far out we can go from the technology standpoint before we even begin an engine system program.

(1) Chamber Pressure. Chamber pressure puts quite a strain on facility considerations, GSE, launch operations, and so forth. We would like to know if we can meet these demands.

(2) Cycles. Have we really evaluated the cycles thoroughly enough to know what is the optimum cycle? There are various types of cycles being proposed. They are gas generator, topping, lower topping, stage combustion, and various others. These cycles have quite an influence on the final system.

(3) Altitude Compensation Methods. This is another technology area which has an impact on the vehicle system. Any further comments would be reemphasizing comments already made.

(4) Materials. We would like to utilize the full potential of the materials available now. Should the state of the art of materials increase, we would like the system to incorporate these materials and use them to the best advantage, thus resulting in a performance or functional improvement.

All of the previous points should be considered in looking at the systems that will be proposed and presented. There are two questions that must be answered prior to starting an engine development program: the first being can we build the system and the second, how should it be built?

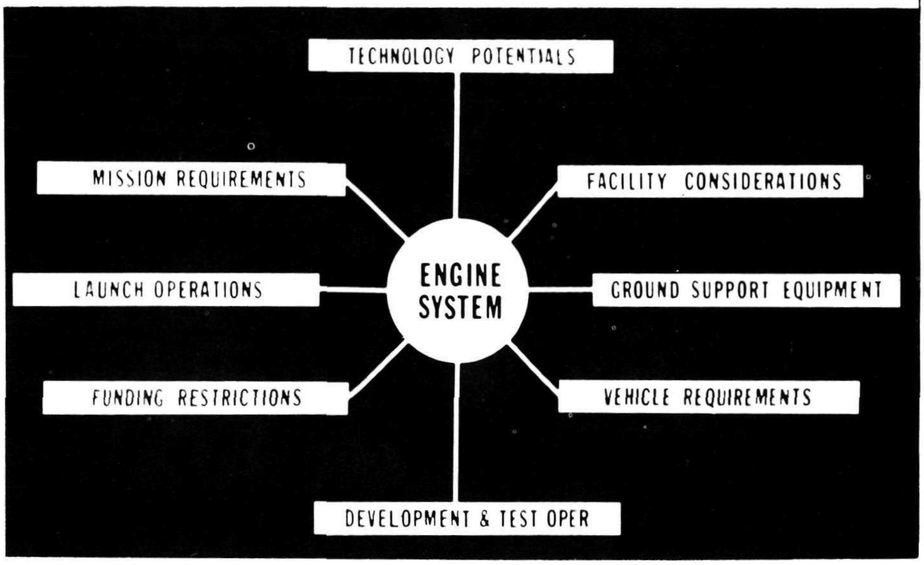

Figure 1

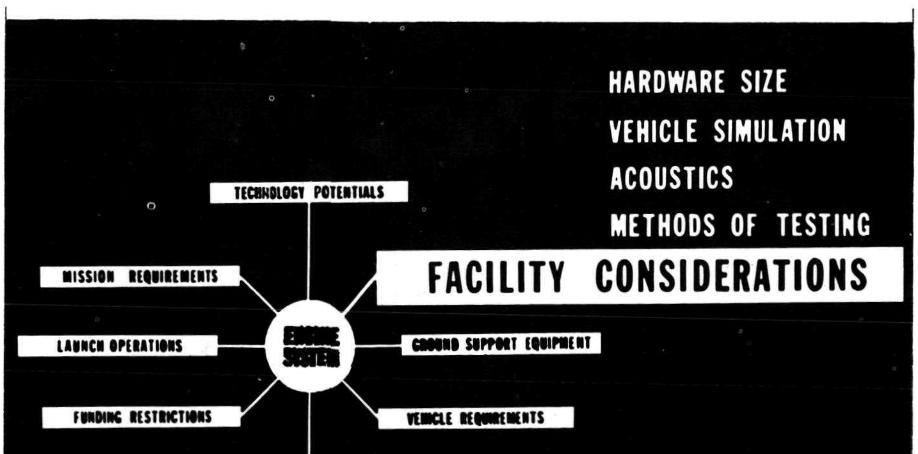

Figure 2

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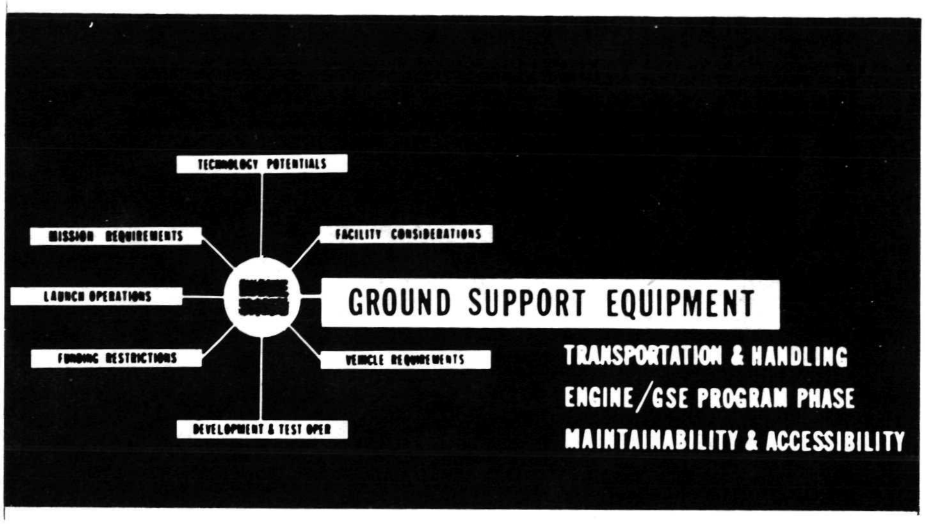

Figure 3

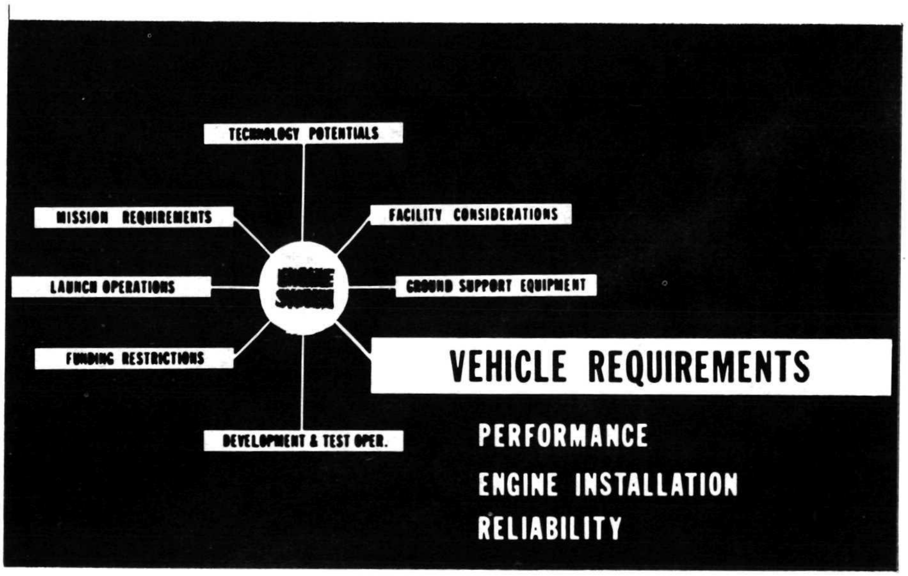

Figure 4

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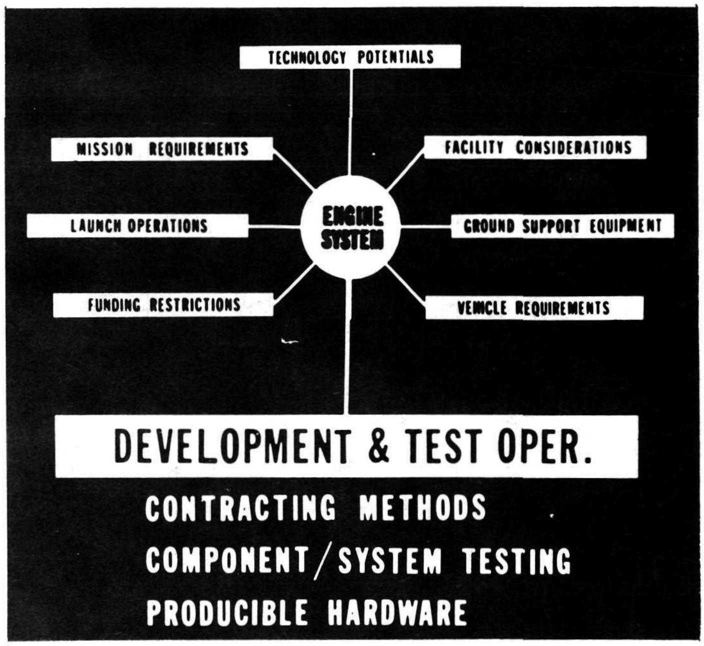

Figure 5

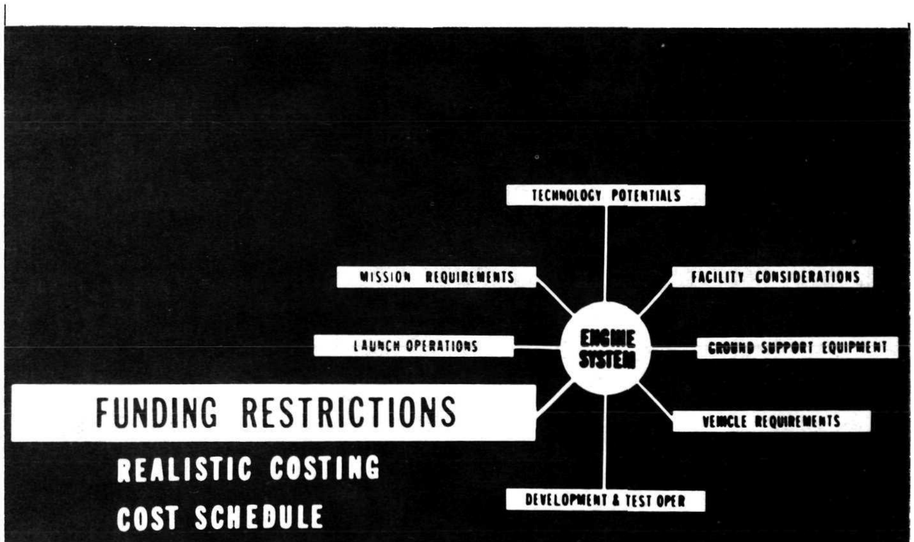

Figure 6

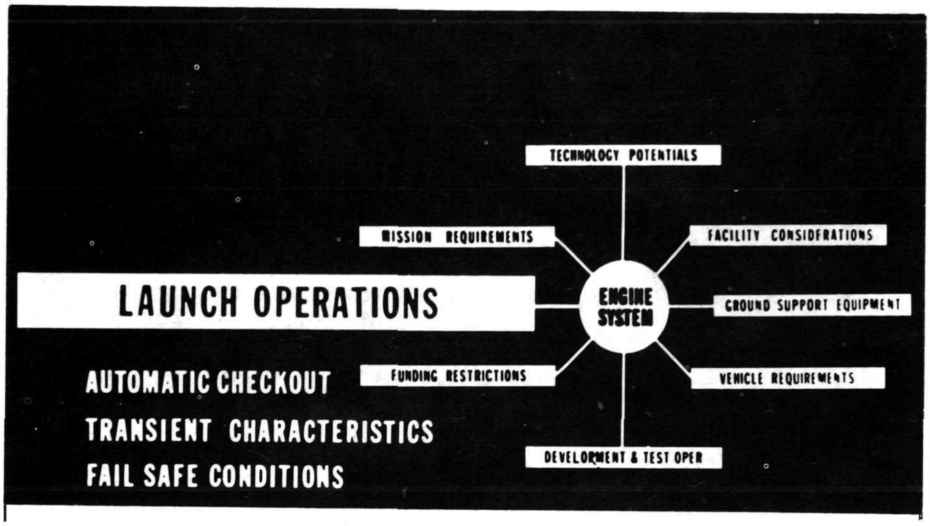

Figure 7

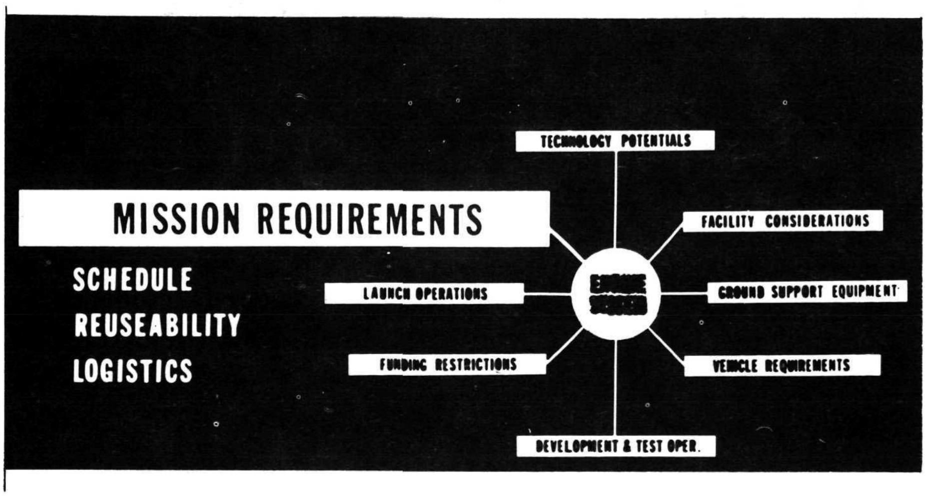

Figure 8

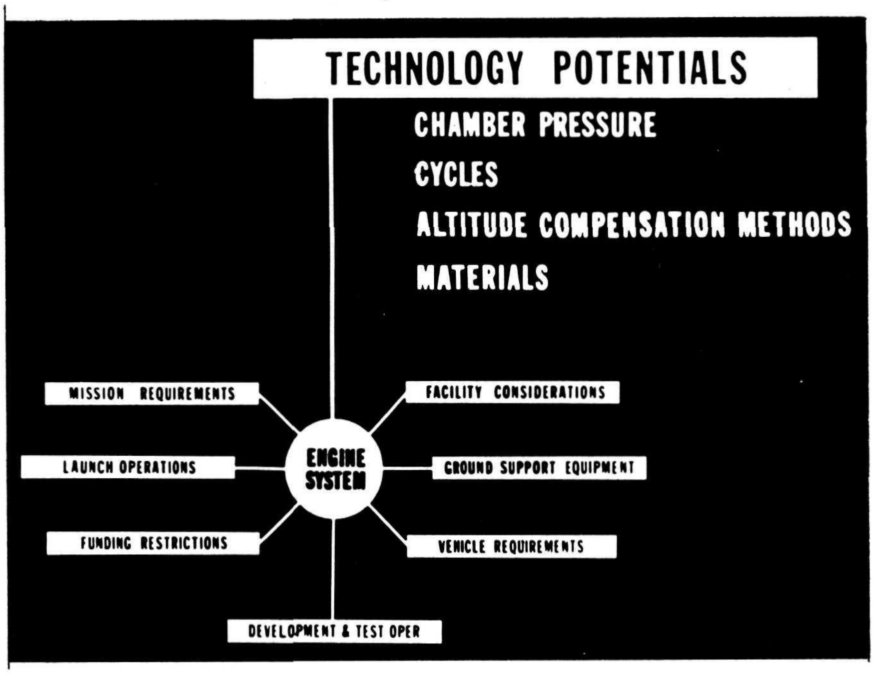
TECHNOLOGY POTENTIALS
CHAMBER PRESSURE
CYCLES
ALTITUDE COMPENSATION METHODS
MATERIALS
MISSION REQUIREMENTS
FACILITY CONSIDERATIONS
LAUNCH OPERATIONS
ENGINE SYSTEM
GROUND SUPPORT EQUIPMENT
FUNDING RESTRICTIONS
VEHICLE REQUIREMENTS
DEVELOPMENT & TEST OPER

Figure 9

8. ADVANCED ENGINE SYSTEMS

By R. C. Stiff

Aerojet-General Corporation
The General Tire & Rubber Company

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.

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.

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?

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.

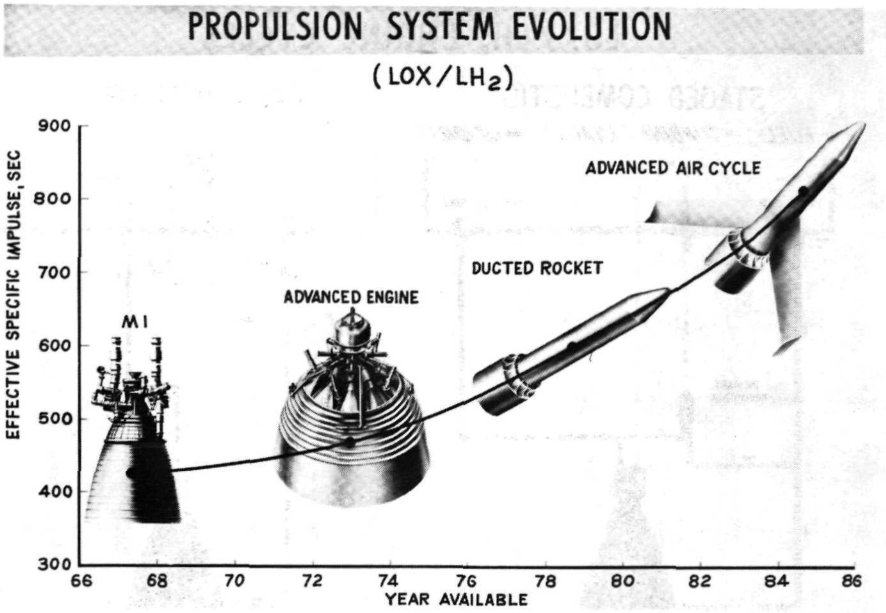

Figure 1

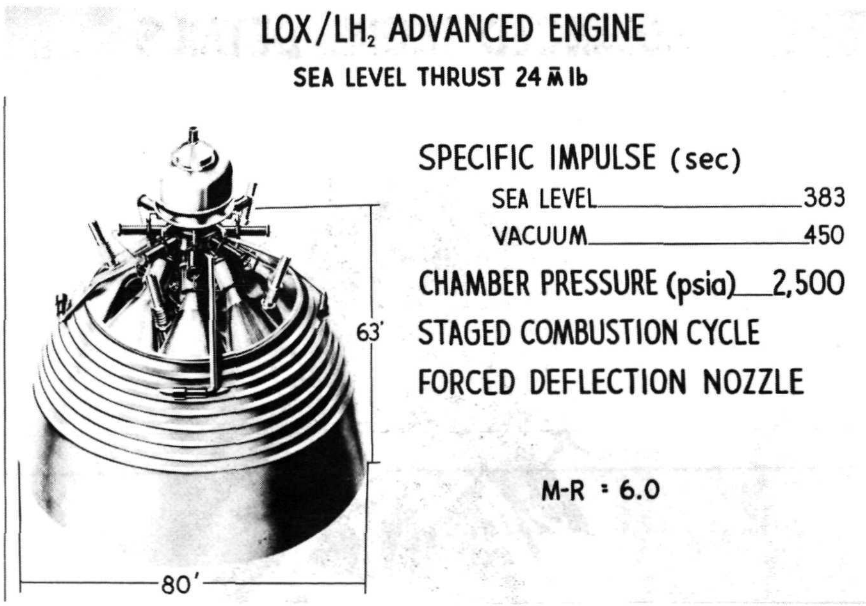

Figure 2

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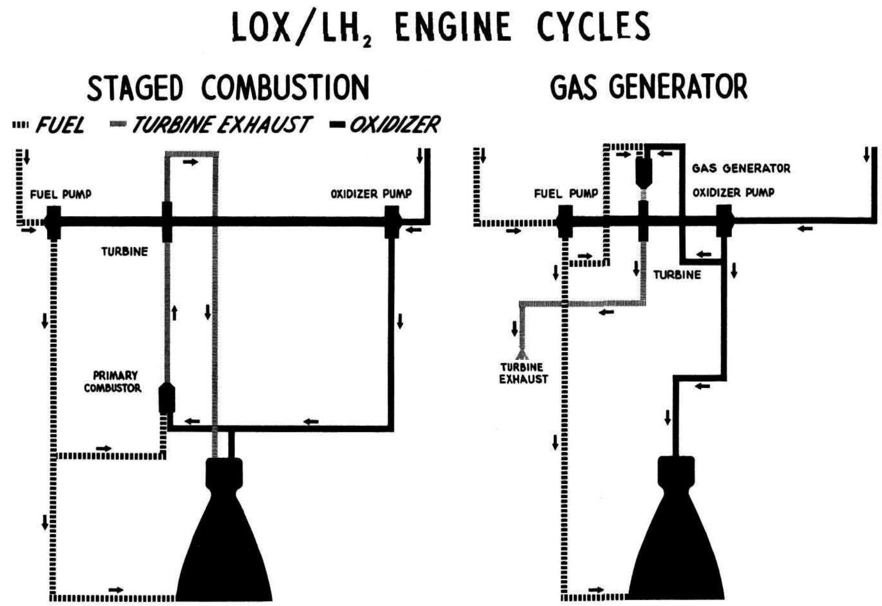

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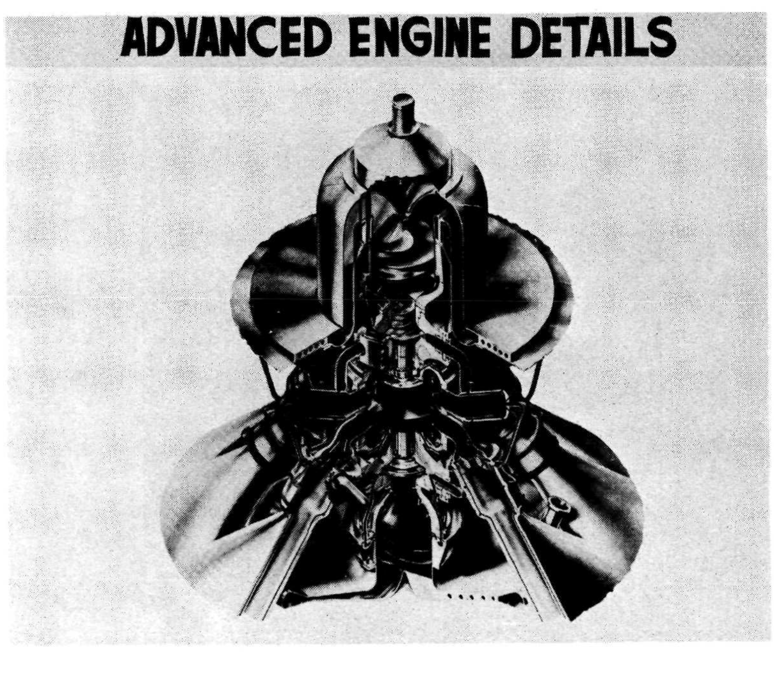

Figure 4

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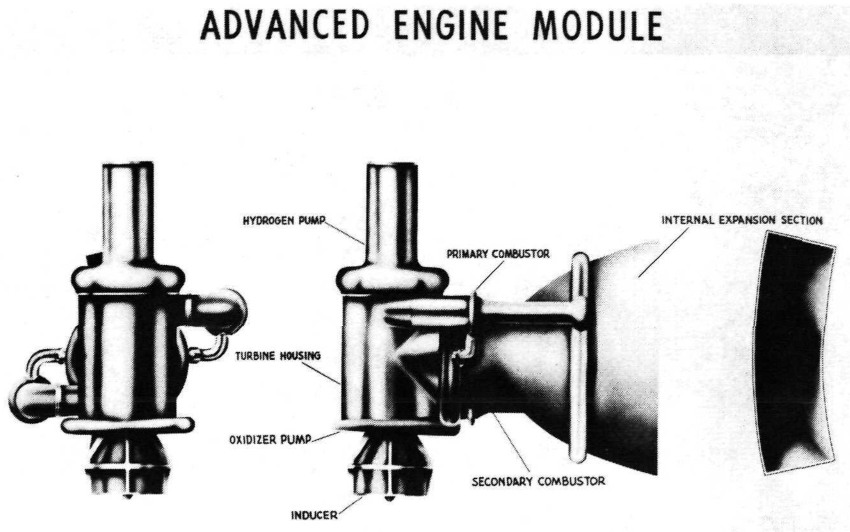

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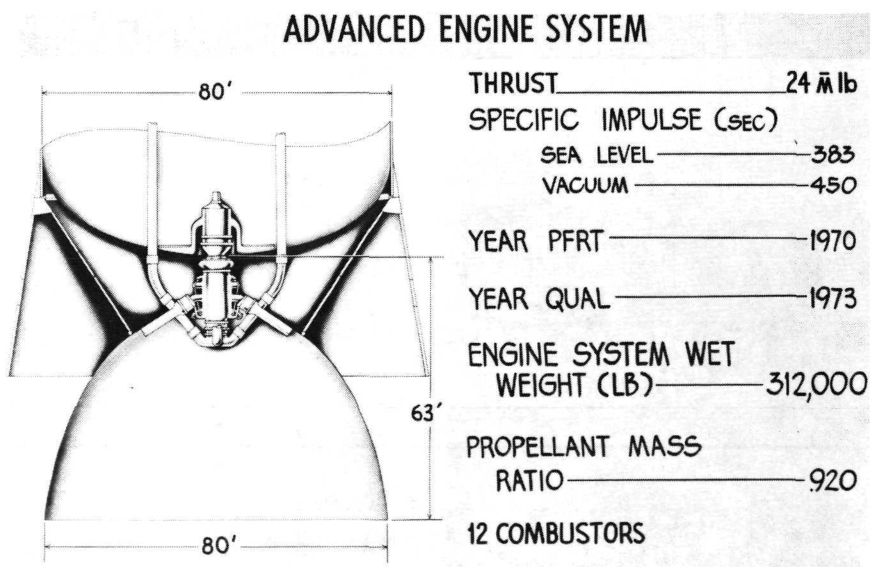

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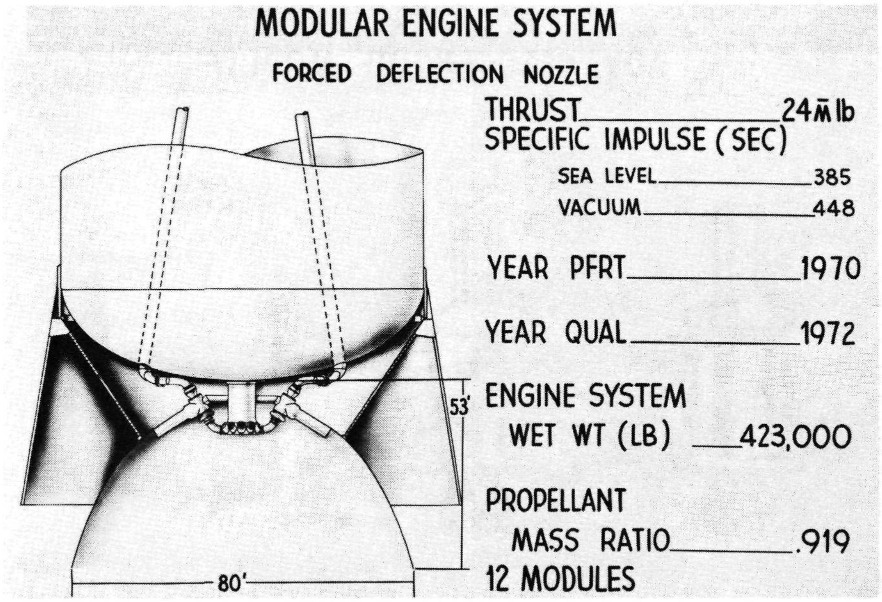

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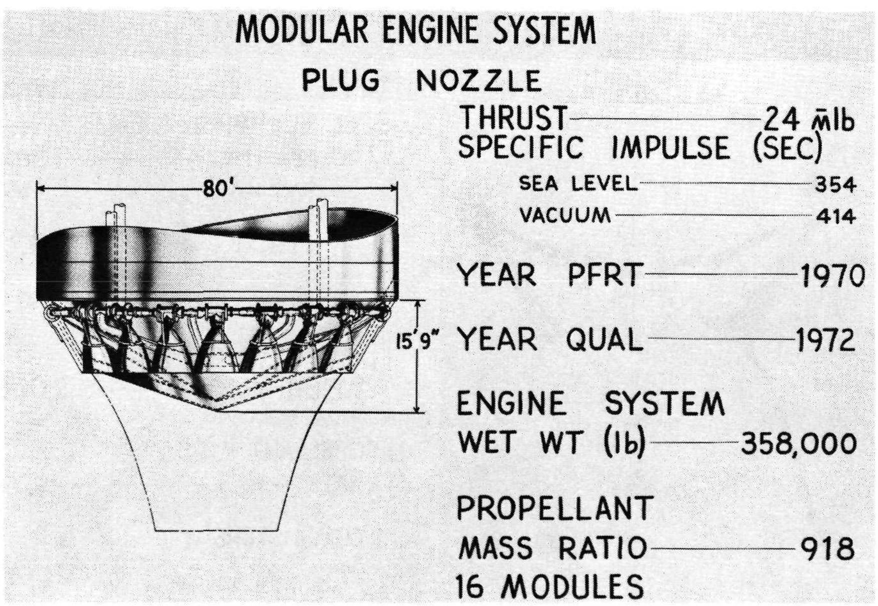

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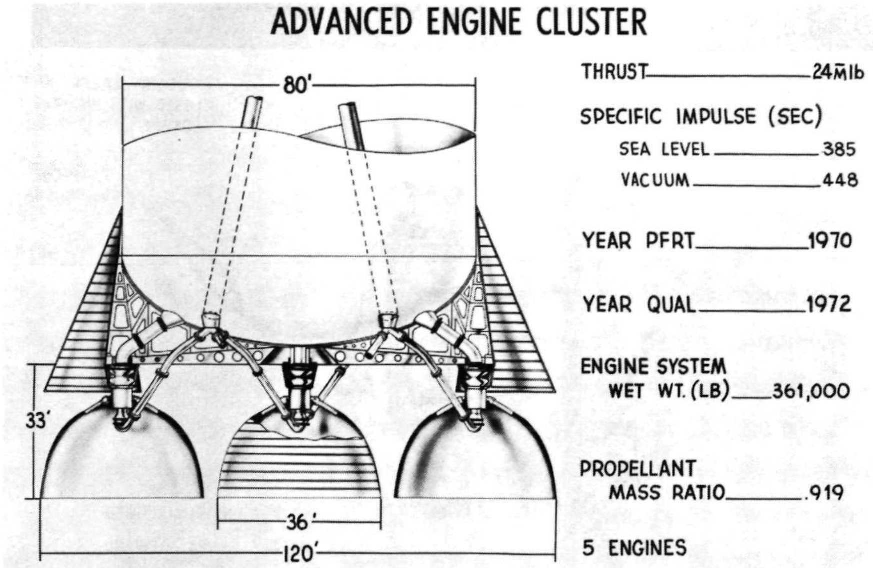

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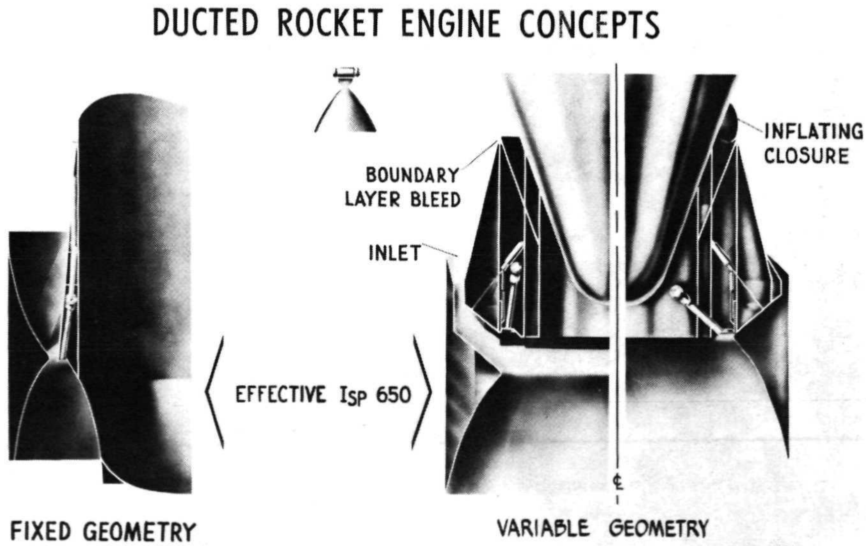

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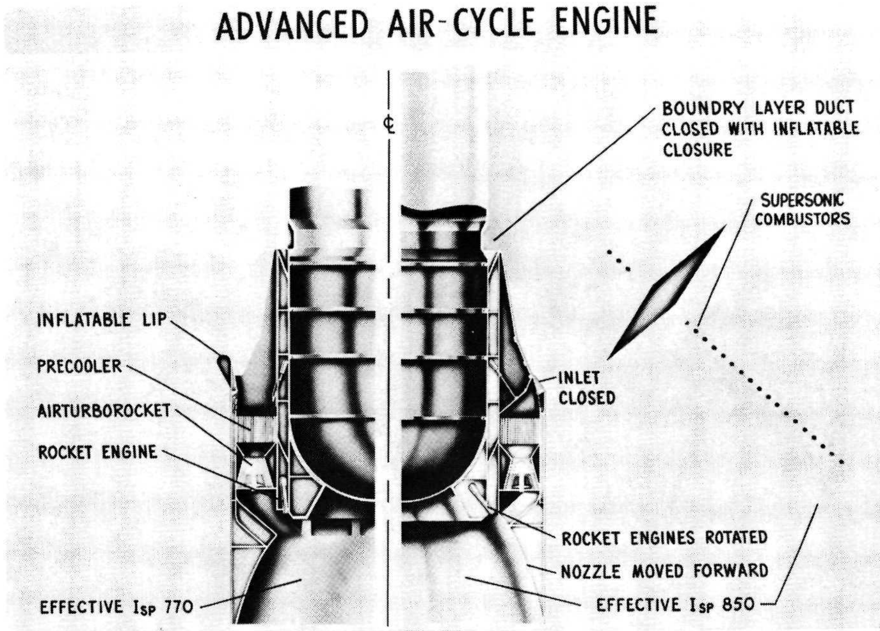

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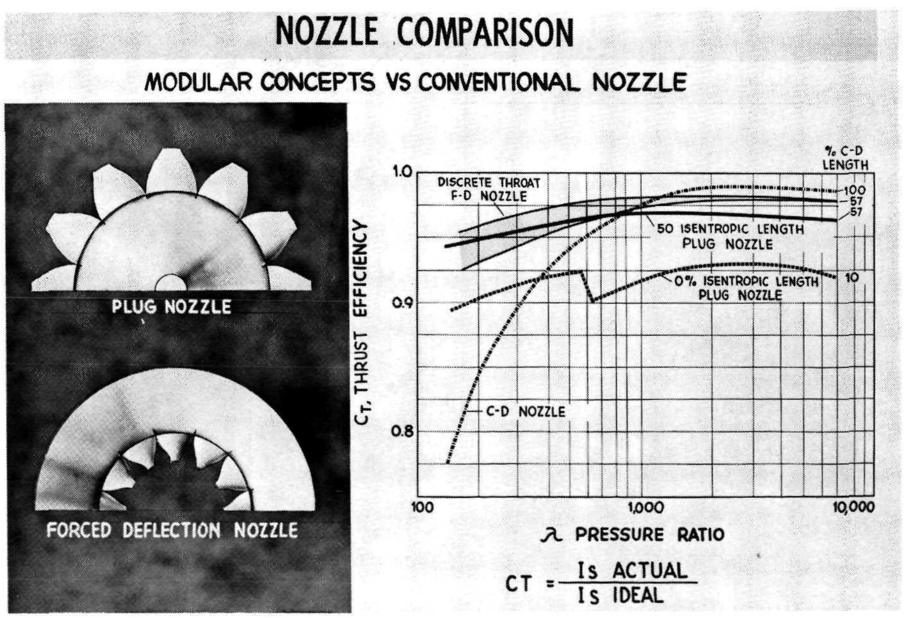

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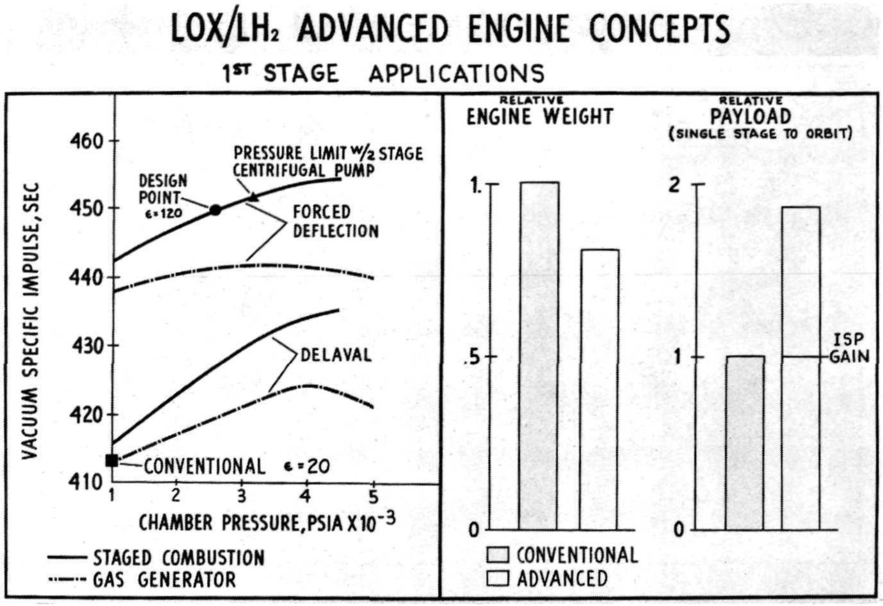

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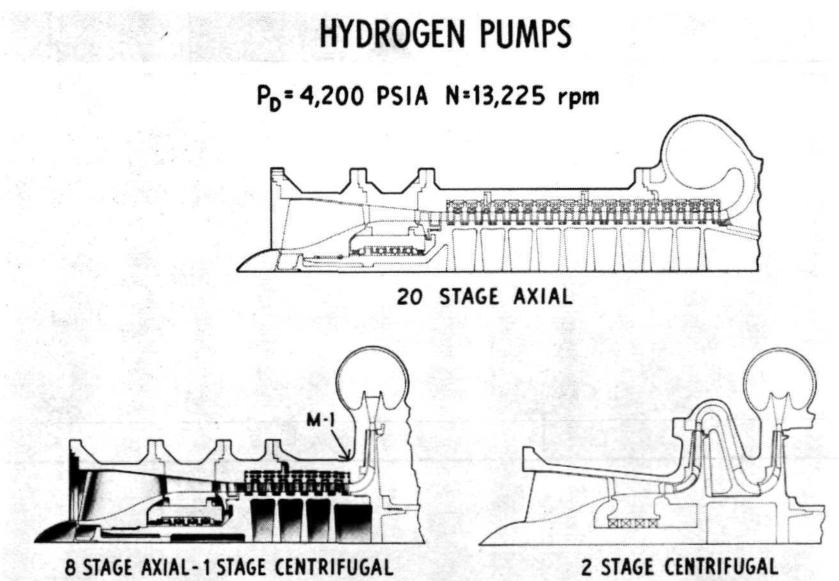


Figure 14

ADVANCED ENGINE INSTALLATIONS

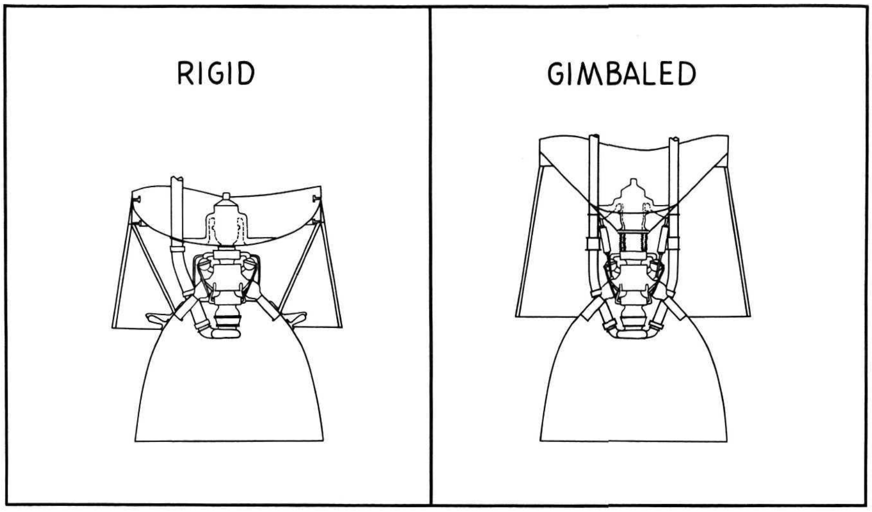


Figure 15

MULTIPLE ENGINE INSTALLATIONS

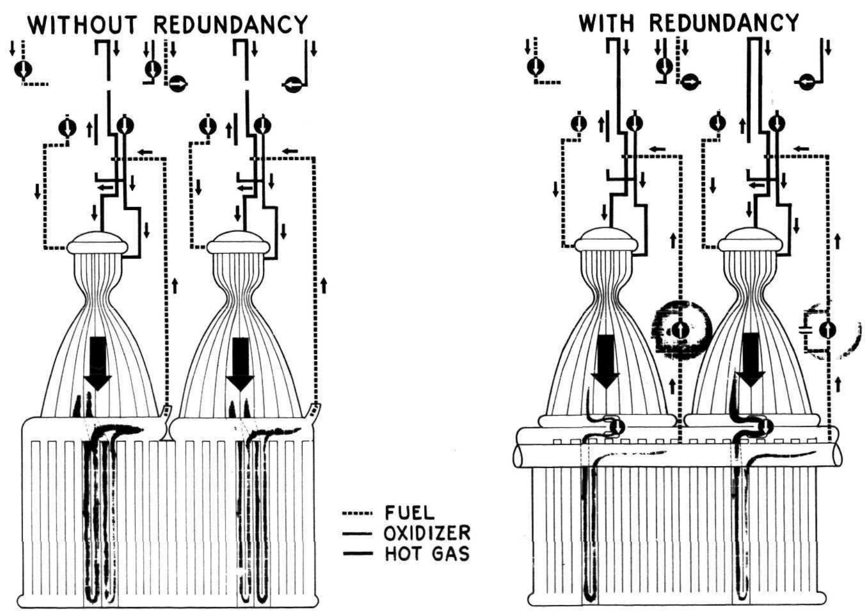
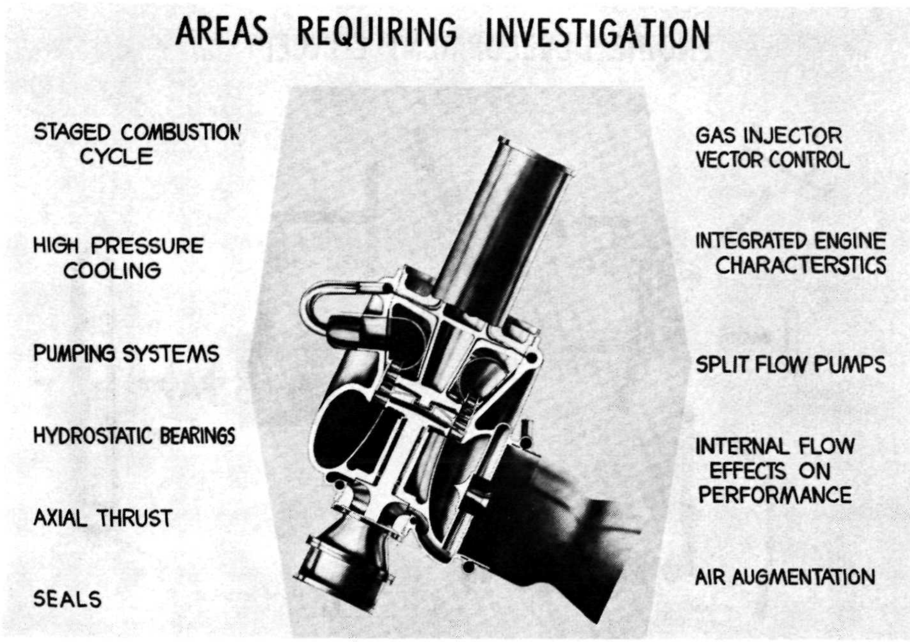


Figure 16



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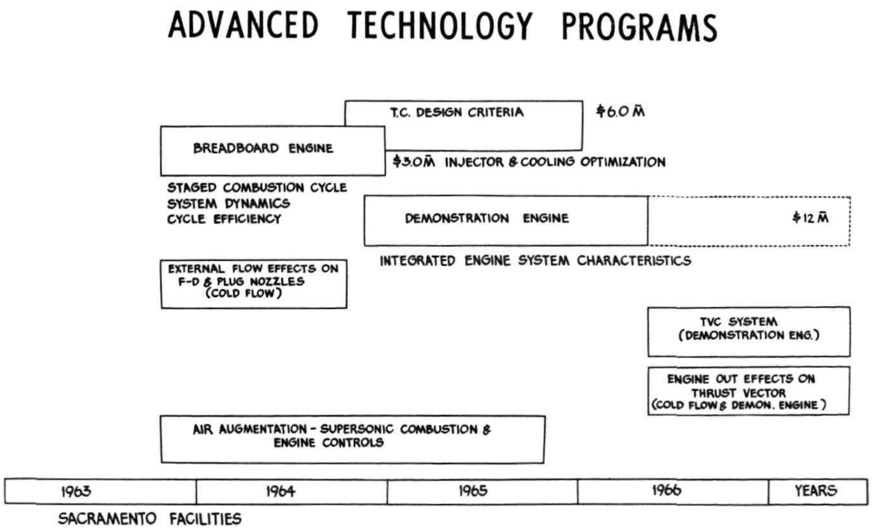

Figure 18

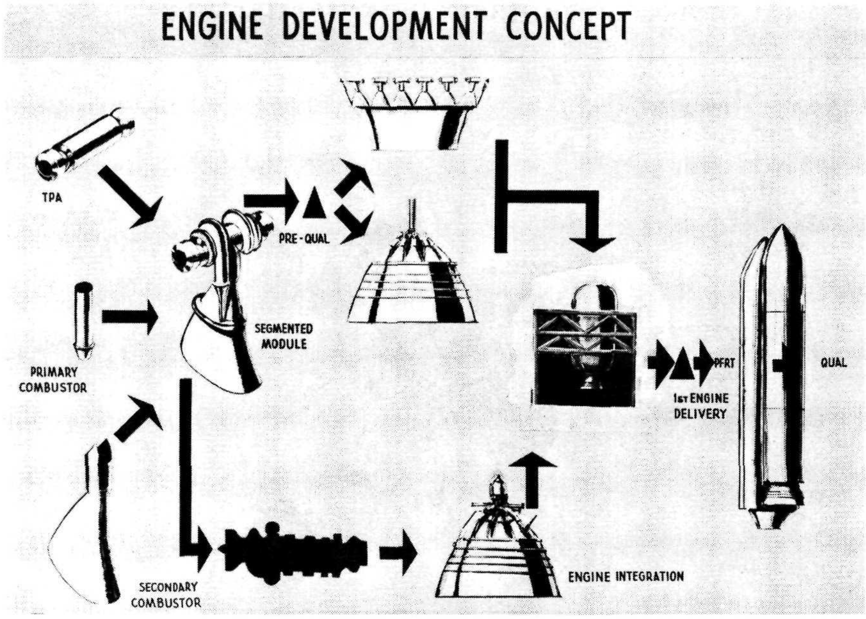

Figure 19

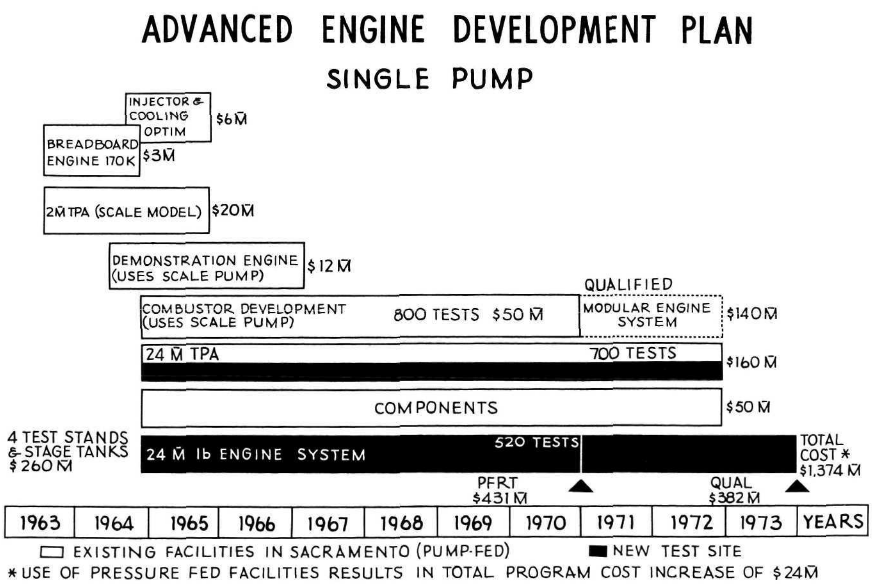

Figure 20

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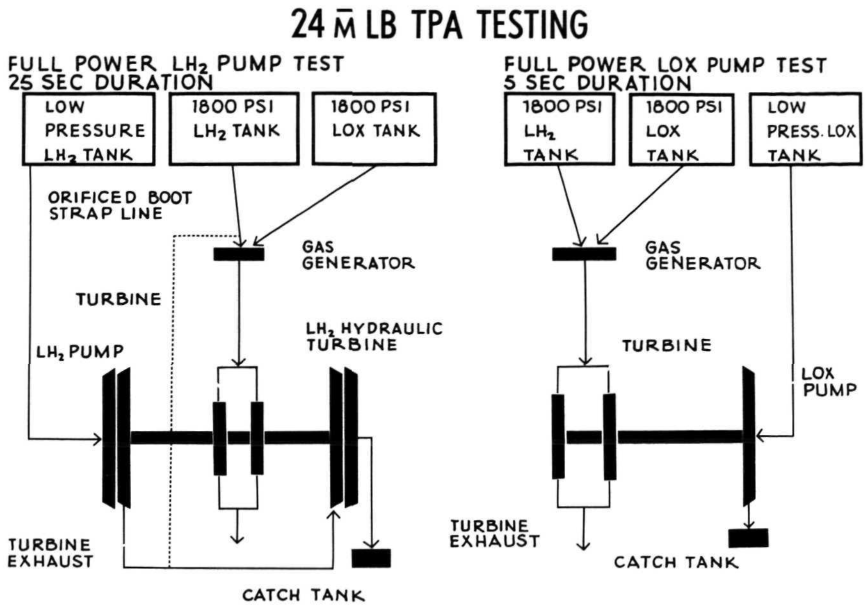
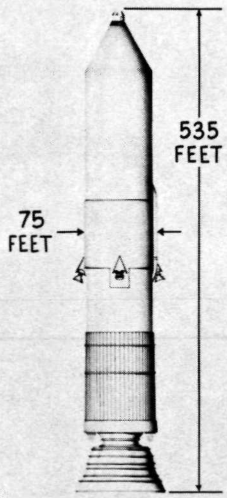


Figure 21

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

[redacted]

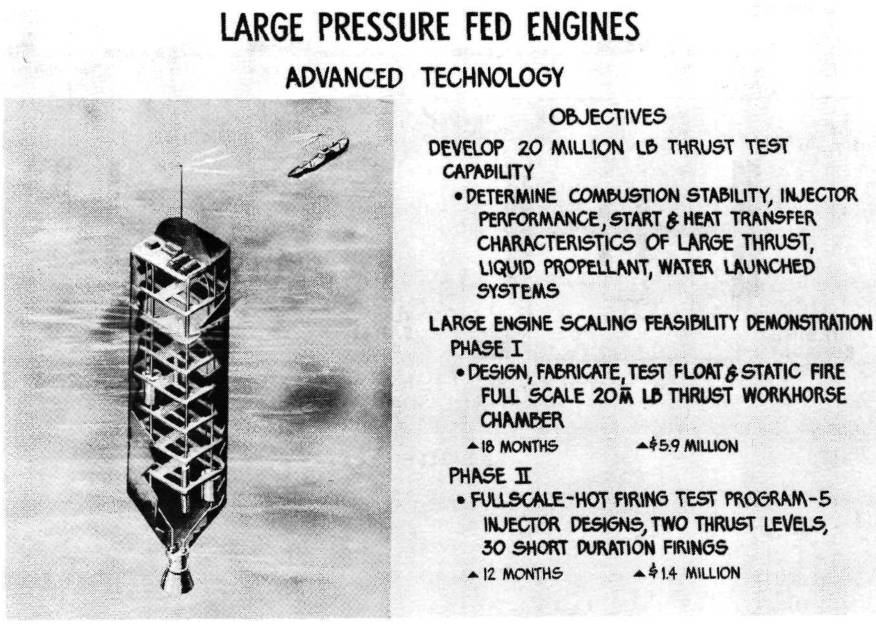

LARGE PRESSURE FED ENGINES
ADVANCED TECHNOLOGY
OBJECTIVES
DEVELOP 20 MILLION LB THRUST TEST CAPABILITY
• DETERMINE COMBUSTION STABILITY, INJECTOR PERFORMANCE, START & HEAT TRANSFER CHARACTERISTICS OF LARGE THRUST, LIQUID PROPELLANT, WATER LAUNCHED SYSTEMS
LARGE ENGINE SCALING FEASIBILITY DEMONSTRATION
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

9. VEHICLE REQUIREMENTS FOR PROPULSION

By Richard Mulready

Pratt & Whitney Aircraft Division
United Aircraft Corporation

I have chosen to narrow down this presentation to some thoughts about one of the systems that we think has some particular advantages. The things we will cover are, first, the vehicle requirements for propulsion, and we will discuss a system that we think fits pretty well. We will talk about the background that led us to these conclusions, some of the current status of technology, and some of the recommended areas where we think more work should be done.

We will also present a preliminary development program plan for such an engine system, and our guess as to what the cost would be.

A "typical" Nova vehicle is shown in figure 1. We have had feelings that have been getting stronger that single stage to orbit, or partial staging (with droppable tanks), would be one of the final answers, particularly if we were talking about a reusable Nova which we were going to recover. We would like to have a one-engine system that we could start up on the ground and bring back. This is the configuration that we have been spending most of our time on.

Some of the installation objectives that are desirable for the advanced booster propulsion system are listed in figure 1.

Direct thrust structure turned out to be a significant performance factor as far as the propulsion system configuration was concerned. It is important that the thrust loads be put directly into the outer shell if that is possible, rather than through a bridge structure across from the center to the outside. It is desirable to have a high effective area in the vacuum condition in a relatively short length. Altitude compensation, although we must admit that the importance of altitude compensation, per se, we think is less important than we thought originally, is desirable because in this kind of vehicle you are pushing hard for performance.

Variable thrust is a desirable thing to provide, in that it will permit, (1) a start up and idle checkout on the ground, (2) a reduction of accelerations in the latter part of the flight, (3) for reentry, a possible use of the engine system throttled for part of the retrothrust, and (4) it provides a much easier development path. The system can be started and run at successively more severe environments by increasing thrust level.

The system should have high mission success probability. The system should be reliable; we will discuss this further. It should be reliable in individual engines and it should have, we think, redundancy to provide an even higher propulsion system reliability.

The installation flexibility includes items such as application of the same propulsion packages to more than one stage of a vehicle if it has two stages, or, conceivably, a very similar propulsion package or module to more than one class of vehicle. This affects the reliability because it allows you to spread out and amortize the development cost over a larger use rate.

We believe that the engine should be accessible in order to change components, to do maintenance, and to inspect.

We like to think of reusability as time between overhauls. We don't see any reason why rocket engines can't be built to have long times between overhauls.

The type of installation shown in figure 2 is called the truncated plug cluster, using high-pressure engine modules. It is a hydrogen-oxygen engine system. In this design, some 91 percent of the thrust is carried directly from the modules to the outer shell of the vehicle. It has a high area ratio. In this particular configuration it has an area ratio of approximately 100, and it has a relatively short length for this area ratio. The length is about 40 percent of the diameter of the effective base area. So its L/D is rather short.

It has altitude compensation. This is by external flow adjustment and requires no internal bleed on the plug. We have test data which show this. We also have test data for one we call a secondary expansion nozzle. It is an ED, FD, or any other deflection engine. This one we found does require ventilation to have effective altitude compensation.

The engine system can be designed to provide variable thrust. We think that you can provide, in the cycle that we are using, a range of 10 to 1. It is possible to consider a 10 percent idle node.

This installation has enough engine modules to provide engine-out capability, which we will show a little later; we think it increases the system reliability appreciably.

Each unit in the 24-unit installation shown in figure 2 would yield on the order of a million pounds of thrust to provide 24 to 30 million pounds of thrust for the 24-unit device. The 1 or 1.2 million size is one that could conceivably be applied to other stages or other vehicles. The engines are all on the outside so they are accessible. The bell nozzles are for the most part regeneratively cooled and we believe that the plug from the exit of the individual modules to the aft end of the plug and across the face of the plug can be regeneratively cooled using state-of-the-art technology for building the heat transfer surface. There may be some fabrication problems, but this is designed with regenerative cooling rather than ablation cooling as we originally thought.

The thrust vector control would be provided by secondary injection, differential throttling or by cooled flaps. We are looking at these approaches as part of our work on a recent contract with NASA. Some other work has already been done on this. The installation has a multiple exhaust source so that it is readily adaptable to air augmentation should this technique prove attractive.

The hydrogen pump is located adjacent to the tank, which may simplify the cool-down and start-transient problems associated with some of the other engine systems.

One of the questions immediately raised when a large number of engines is used in a propulsion system is system reliability. It is important that our definition of the module is clearly understood. The module, to us, is the part of the system that is self-contained and can provide thrust by itself. It receives the vehicle propellant supplies and ancillary supplies, and then discharges the expellant, at relatively low static pressure and temperature. It is the element of the system upon which the major part of the development effort must be expended in order to insure that the system interactions, which always cause most of the trouble, are thoroughly examined and that the development effort is accomplished on the part of the system that is significant.

Depending upon the specific design requirements, it is conceivable that a module could have one pump, and multiple thrust chambers, or a single pump or two pumps and one chamber. For the booster systems in figure 3, the center module with the one pump and the one chamber seems to fit best, although there are some places where type A might fit. We think that that is more a matter of the individual design requirement than it is specifically a matter of reliability, provided that the system gets developed as an engine, including everything that is inside the box.

Figure 4 shows the results of a study that we accomplished last year to take a look at the effect of module size on system reliability. We can provide copies of this study to whoever would like to have it.

Our conclusions from the study were that redundancy, which is the difference between the lower curve with no redundancy, and the upper one with, for instance, 10 percent redundancy, can add significantly to the reliability of the overall propulsion system and that for a typical investment in the propulsion system, including both production and development, that the best module reliability is obtained with a modest unit size because more development testing can be accomplished on the smaller size. We must grant that there are many assumptions here but we have examined them and we think they are right; the number of development tests for the various module sizes is also shown.

The final conclusion is that it looks to us that somewhere around a million pounds of thrust would be about right for a typical investment in production and development for a module for the Nova vehicle as we see it defined.

Figure 5 shows the cycle that we are talking about. The combustion is accomplished in tandem, with most of the hydrogen being consumed in the preburner with a modest amount of oxygen added to provide the working fluid to drive the turbines, the balance of the oxygen being added in the main chamber, and part of the hydrogen being used to provide regenerative cooling for the aft portion of the nozzle and film transpiration cooling in the section from the injector to the area ratio of 4 to 1.

Figure 6 is a photograph of a model of this engine. It is a cutaway with a quarter section cut out of it. What we have found is that it is possible with this cycle to arrange the two turbopumps and the preburner system into a plug-in design which collects all of these hot pipe elements into one transition case, and that there is enough hydrogen available from the thrust bearing system in the hydrogen pump to cool effectively the inner part of that case. With this design we think we can eliminate all hot surfaces on the outside of the engine so you will either have room temperature or cold cases on the engine.

For the plug cooling (fig. 7), our studies indicated that regenerative hydrogen cooling is well within the state of the art for doing this job, that the heat fluxes here are well below those of the RL-10 combustion chamber, and the same kind of tubular structure could be used. It is interesting that if you can run the tubes hot enough - and this would require better materials than we are presently using, or higher temperature materials - so that you can get an exhaust temperature of the gas that you are using for this cooling up to something on the order of 1,400° F, you can get good impulse out of the gas with a relatively modest area ratio. The figure shows the nozzle that is exhausting the coolant overboard after it has followed a path indicated by the arrows through the plug. The impulse of that hydrogen is almost as good as the basic engine system.

For reentry, the heat fluxes for typical trajectories can be a good deal lower than they are on the way going out. For reentry, the outer edge of that chamber would probably have to be cooled or it would burn off. It doesn't take too much to do that. We calculated for a 20-million-pound-thrust booster that it would require a little over 6,000 pounds of hydrogen to do the reentry cooling. This is a great deal better both with regard to the weight of the structure and the weight of the propellant, which we are getting a pretty high return on, than we could do with the ablative system that we were originally thinking of.

Figure 8 shows the results of our latest plug test performance. This is for a 24-module plug cluster truncated to various lengths. The lengths are given in a percentage of the isentropic spike length: 0, 2.7, 9.4, and 16.0 percent. The nozzle was designed for a vacuum cluster area ratio of 14.7 and I think the pressure ratio is 244 for that. At that point, which is the vacuum performance point for this system, the 9.4 percent plug length has a velocity coefficient C_V of 0.966. In our published performance estimates that we expect to be able to get, we are expecting 0.975. So we are rather close, and we hope that in the program we are currently embarked on we will be able to achieve 0.975. Velocity coefficient is the thrust produced by the nozzle divided by the thrust that would be produced by an ideal frictionless nozzle operating at the same pressure ratio.

We have done studies for booster engines, comparing various cycles and various propellants. The present studies are not yet completed. Figure 9 shows one of our cycle comparisons; it is one of the plus-and-minus type of comparisons. We are comparing a parallel combustion or gas generator system with a tandem combustion system in terms of performance, weight, size, bulk density, and so forth.

For performance, it can be seen that the preburner cycle would have an 8-second impulse advantage over the gas generator cycle, both operating at 3,000 psia with an area ratio of 98.

In terms of weight, the gas generator cycle has about a 10 percent weight advantage over the preburner cycle. (The chart should read minus 10 percent rather than just 10 percent.) The reason for this is for the most part that the pump pressure required to work the gas generator cycle is lower than it is for the tandem combustion cycle.

The physical sizes of the two engines are much the same for equal chamber pressures but they will have slightly different shapes.

One other advantage to the preburner cycle is vehicle bulk density. The difference in overall propellant mixture ratio for the same main thrust chamber ratio between the two results is a difference in vehicle bulk density of approximately 8 percent. This difference in bulk density reflects back in vehicle structure and more than offsets the 10 percent weight difference.

From a combustion stability point of view, our feeling is that the tandem combustion cycle offers a decided advantage because, first of all, the combustion at high mixture ratio in the main chamber is with high temperature gas coming into the chamber, and from some Lewis data that we saw sometime ago, this seemed to have a considerable effect: the higher the temperature was, the higher the oxygen/fuel ratio could be without encountering instabilities.

Further, the low temperature injection of the propellants required in the gas generator cycle because there is no heat source in that system to warm up the propellants is one that is never far away from combustion instability, and it takes a lot of fine whittling to make it stable.

The low temperature injection of propellants for combustion in the tandem combustion system occurs in the preburner where the mixture ratio is relatively low, about 1. Temperature is of the order of 1,300° F. The system is thus less likely to be unstable; also it is at a lower temperature, so that you can cope with damping devices that we are familiar with if you do encounter stability problems. We think that, fundamentally, we would just put a plus sign on the preburner side; we think that that cycle is apt to have real advantages in combustion stability.

On the pump ΔP comparison there is no question. In order to provide the power to drive the tandem combustion system you must have a higher pump discharge pressure. For a 3,000-pound chamber pressure in an engine size of 1 million pounds thrust, it is approximately 5,500 pounds pump discharge pressure. This we think can be accomplished.

The final item is the configuration. We think it is possible to put the preburner system together so that you have no hot cases. You reduce the amount of plumbing and we think it makes a simpler configuration for installation in the vehicle.

The effect of chamber pressure on payload is shown in figure 10. This effect is twofold: One, because for a particular vehicle as you increase chamber pressure you can afford to go to a higher mixture ratio; this is reflected by the increasing values of oxygen/fuel ratio. The curve would look this way for an operational vehicle. The filled-in point is for a current gas generator engine, using a plugged compensation technique, using the same kind of plug performance that is applied to the upper curve, but using lower pressure gas generator cycle performance. So the difference between payload capacities is almost two to one, 1,040,000 pounds to 590,000 pounds. It is about 80 percent.

I would like now to switch from what we think it ought to be for the vehicle and the reasons why the engine looks the way it does to talk something about the status of the work that is being done on Air Force contract and also work being done which is funded by the company.

Figure 11 is a photograph of the pump that was started by the Air Force in 1961 and was funded by them for the first year of that effort and was then picked up by the company. I will talk more about the details when I get to a summary chart.

Figure 12 is a photograph of an oxygen pump; the flow capacity of this pump and the pressure level match those of the hydrogen pump so that hydrogen pumps could be used, and it is planned that they will be used, in 50,000-pound-thrust combustion chamber work under that Air Force program. This pump has been entirely funded by the company.

Figure 13 is a photograph of an injector which has been developed under this Air Force program. I think it is one of the most significant things to come out of it. This is the good picture after many bad pictures. The original work done in the period June 1961 to 1962 was in the 5K size, and no injector ever made the second firing. There were all kinds of damage to those pieces.

A lot was learned, however, in those tests. We then went to a 10K size, and this injector was developed to be used in heat-transfer measurements using a probe. That is the reason for the hole in the center of the injector. This was one of the first of the 10K injectors tested; it had been fired six times for short duration tests at the time that this photograph was taken. Most of those firings have been accomplished in a short time period, because the chambers are not cooled. It has since gone on for further testing and was fired a total of nine times, I think, with no deterioration. With this system C* efficiency is of the order of 95 to 97 percent.

Now I would like to talk about some work that is being funded by the company. These are late data. They were obtained from recent test results. The approach, is proprietary, and we have filed for patent applications. It is a practical approach to a very fundamental problem of high-chamber-pressure engines, and that is the cooling technique.

It was reasoned that one of the things that we want today when we cool with transpiration or film cooling, and this is really a combination of both, is to be able to divide up the thrust chamber into actual segments because the amount of coolant that we want to inject at the throat is far different from the amount that we want along the constant-area section of the combustion chamber.

We had done design work and had looked at systems using Rigimesh with cavities behind it and then orifices into these cavities to provide the axial distribution, and we think this could have been built. As a matter of fact, we had started on our own to build some of those when John Chamberlain came up with this idea of using a series of wafers which are stacked up to produce the thrust chamber surface.

The surface is a copper disk about an eighth of an inch thick as shown in figure 14. For a distance of about half an inch from the I.D. outward, there are a series of very fine grooves cut into the surface of the disk. These grooves are laid out in an involute pattern. This allows them to be very long. There are about 4 inches from the point where they come through the inner wall here to the place where they come out into this channel. These grooves do not go all the way through the disk. They are just cut into the surface about 0.030 inch deep. They have metering orifices in four radial locations which provide from an outer manifold the distribution of the propellant flow for the coolant along the length of the chamber. So the hydrogen flows through a jacket, through the distribution slots, and then around and out through the involute slots.

On the right is the other side of the adjacent disk, and these two fit together so that there is a crossed involute pattern that goes from this manifold to the inside of the chamber.

Figure 15 is an artist's conception of the actual rig that was tested. It shows that the rig is made up of a stack of these disks. In this design we just started with the same disk and machined the shape inside. Hydrogen is brought in; part of it passes down and through these disks and is distributed along the surface according to the size of these small metering orifices.

Figure 16 is a photograph taken from the injector and looking at the throat after some four firings had been accomplished on this, a total duration of actually about 5 seconds. The last firing was a duration of 3 seconds. This is a facility limitation; it is about as long as we can hold chamber pressure. This was at a chamber pressure of 3,200 psia. We don't have all of the data from the oscillograph records yet. But the first cut looks like the C^* efficiencies are on the order of 90 percent.

In the first test we ran with this we used a different injector which caused impingement on the wall locally, and had fired and retained about 90 to 95 percent of the surface; the C^* efficiencies were 95 percent plus. We plan to do a little more work with this to see if we can push the C^* efficiency up. We think the 3,200-pound chamber pressure with that level of C^* efficiency is getting to the point where the environment is pretty severe.

Figure 17 shows an oscillograph chamber pressure trace and a throat temperature trace from this 3-second firing. The time intervals have been interrupted here so we could get it on one sheet of paper.

The erosion that occurred in the chamber was just downstream of the throat of the chamber and was very minimum in the three firings that have been made. A little bit has occurred, and we think each time it has been occurring on the starting transient. We have a lox line filling problem. The line fills up with a slug, and we are getting very high mixing ratios during the transient which is reflected in the throat temperature at the start transient.

Figure 18 is a photograph of a flight weight design employing the wafer cooling technique. It is a stack of washers, the same kind of general design as shown in the heavyweight rig, but much thinner. They are not deep radial sections. They are in the order of 3/8 to 1/2 inch deep. It looks to us after having laid this out that we can be competitive with the Rigimesh type with a more reliable chamber and one that can be developed at a much lower cost.

In table I we have attempted to take the items that we think are important for this kind of a propulsion system, to list the major objective for each of these items, a brief statement on the status and the current effort, and some of the things that we will recommend be done.

The first one is the plug cluster nozzle. The objective was to obtain at least 0.975 C_V for a 10 percent plug length. We have gotten 0.965. A program has recently been started under NASA which will continue this plug testing. We think that this performance improvement will certainly come out of that, probably some more. Following that program we think hot firing tests in scale size will be a desirable thing to do.

Thrust vector control is the next item. It is hard to put any numbers on it. The secondary injection work is being done by our research department under NASA contract and looks very good. Under that program a demonstration will be performed on the RL-10 to push the scale of the work up a little bit, and secondary injection also will be investigated under the plug cluster work.

For differential throttling, preliminary test and analysis indicate that it will be quite satisfactory both from an engine response point of view and from the ability to provide the kind of thrust vector angle that is needed. These angles have gone down some since the initial part of the Nova studies. This also will be tested under the plug cluster test program, and we are going to attempt to try some flap configurations on the base of the plug which might be used, probably not as a fine adjustment of the thrust vector control, but as an emergency system which would provide the capability of putting in a very high angle if it were necessary. All of those will be covered in this program. We think a hot firing program should be defined following that effort.

Coupled with it, we think that an analytical study from a propulsion man's point of view ought to be accomplished. We would like to look at the things we would like to change in the vehicle design and the engine systems to make thrust vectoring an easier job.

In table II the combustion effort is broken down into a number of items. First is the preburner which has injection at the low mixture ratio, but high chamber pressure. The major objective is the flat temperature profile just as it is in the turbojet engine.

There has been no test work at Pratt & Whitney on this. We have done some design studies that indicate that there are practical designs possible. We think there should be a test program of the same size as the main chamber work program but at low mixture ratios.

In the mainburner work, using hot-gas fuel injection, at an oxygen/fuel ratio of 7, at a chamber pressure of 3,000 psi, we would like to have a C^* efficiency (and we believe it is obtainable) of 98 percent and better. So far we have obtained 95 plus, and some of the numbers go to 98. I don't know that our accuracy is good enough to say it any better than 95.

We have covered the range. So we think that this part of it is pretty well under control and that further work on this should go along with the preburner work and that they should be operated in tandem because there are some differences in the mainburner injector. Again we think this should be accomplished in the 10,000-pound size.

On combustion stability, there are two contracts, just starting in Florida. One is a test to establish, first, some of the parameters in our small scale test that adjust the stability margins, and an analysis to determine what could be accomplished with damping techniques. We think something that might be added to this, following this work, would be damping demonstration tests, with some hardware designed to do oscillation damping.

The last item is ignition. Hypergolic ignition with oxygen and hydrogen is a desirable thing to be accomplished. There is effort on these items; O_3F_2 addition to the oxygen, metal powder (submicron metal powder added to the hydrogen we found makes the combination hypergolic), and catalysts. I think there is a proposal in now to continue some of the work on the metal powder additives.

The next item is cooling. The objective is to achieve one-half of 1 percent loss in I_{SP} for a 1-million-pound-thrust engine.

This next item is a pretty strong statement. With respect to cooling effectively the 10K size, we think we have a solution and we will shortly have numbers to prove that we have a solution that backs up this kind of an estimate. It is a big extrapolation but we think it can be done effectively. We think that this particular kind of cooling design deserves effort examining not just the copper materials but other materials, and configurations of the particular wafer design going beyond the present one, including some louver work to provide additional film coverage.

The last item is regenerative cooling. We think that the heat fluxes are well in hand, but that some effort on fabrication of large cooled plug sectors would be worthwhile.

This last summary is concerned with the turbomachinery work. (See table III.) The fuel pump is the first item, and the objective shown is for this particular piece of hardware that we are now testing. The objective for a larger pump would be a much larger efficiency than 58 percent. We have already achieved 54 percent. The pump has reached 4,000 psi in a transient and has run steady state for some 20 seconds at about 3,500 psi. So it has not yet reached its full design output which was 5,000 psi, but we believe it will. Thrust balance, off-design performance, and improved performance are the areas we think need effort.

A lox pump was designed to fit with that same size for 1,000 gal/min and 5,000 psi rise. It achieved, not at the same time but during the same run, 800 gpm and 4,200 psi. It is being supported by the company and we hope shortly to make runs that will get us to the design point. We think that effort is deserved there in the performance and mechanical evaluation of the pump and its seals.

Low-speed inducer work: the RL-10 is supporting some work in this area rather than no direct support as indicated. We think further scale tests on the hydrogen and the oxygen inducer would be a desirable thing to accomplish for inducers. We think that the net positive suction head should be pushed down to well under 1 foot at 1 g condition.

With respect to both rolling element and hydrostatic bearings, there is some work going on in-house. We have gotten 2.4 million DN with the rolling element bearing and some very short time tests at higher speeds. Hydrostatic bearing DN really doesn't have any meaning, except from the critical speed point of view, and equivalent DN's of 4 million have been reached with in-house bearings we have been running. We would like to go further there in investigation of life and load at high-speed conditions.

Rotating seals: high-performance engines will require high rubbing velocities. Seals can operate and have been operating in hydrogen with these velocities, but more work needs to be done on the materials and seal configuration for high rubbing velocities with oxygen.

The last item is turbine cooling. In the cryogenic engines it is required to cool bearings with hydrogen, and this results in a disk and blade attachment gradient of something on the order of 1,700°. If you are using hydrogen at -400°, and you are using turbine inlet temperature at 1,300° F, this gives you a difference on the order of 1,700°. We think that material spin tests under simulated gradient conditions would be very worthwhile for providing turbine design data.

Figure 19 presents a quick summary of what we think could be accomplished for a million-pound-thrust engine or an engine in that size class. For development time, PFRT, the research and advanced technology would overlap the engine development. Component development would proceed first with tests of turbopumps plus preburner which could be operated as a bootstrap cycle. The first engine test would occur in something of the order of 18 months, PFRT in $3\frac{1}{2}$ to 4 years, and qualification tests in 5 to 6 years. As the engine development proceeds, the engine vehicle integration test would involve three engine segment tests at the engine contractor's plant, and shortly following PFRT engines would be delivered for tests on the vehicle for the complete system.

The test facilities required to accomplish this are: First, the module in which the major development effort must be accomplished. We think it will take a four-stand complex to get a satisfactorily high test rate, and most of the problems will be solved in this size. Following that, a single stand testing three modules and a plug segment would give most of the integration answers of the two systems together. The complete cluster test could be accomplished finally on the vehicle facility, either at the vehicle contractor's plant, at the government plant, or even conceivably at a launch facility for this very large testing, using vehicle hardware to avoid costly duplication of test facilities.

Figure 20 shows our estimate of what it would cost to develop an engine of 1-million-pound thrust operating at 3,000 pounds chamber pressure. The cost at the time of PFRT (average cost) is taken from a compilation of data that we have taken from the Congressional Record for what engines have cost at PFRT; we have compared the function of their size. That was one of the key points. Then we looked at what we thought the cost would be following PFRT; that gets rather divergent, depending on what you do with the engine. This is the kind of accuracy we can put on cost. There are uncertainties in cost and time of accomplishment, and the facilities are shown as a separate line at the bottom.

QUESTION PERIOD

MR. GOMMERSOL, Lewis: Did you say that this engine would reach PFRT in $3\frac{1}{2}$ to 4 years at a cost of $200 million?

MR. MULREADY: I think the midpoint is of the order of $160 million.

MR. GOMMERSOL: Don't you think that is rather low?

MR. MULREADY: This is labor, material, and overhead, G and A. It does not include facilities and propellants. It is the number that we have been able to get out of the Congressional Record to fit the M-1 funding as of October 1962. What we have tried to do, instead of playing the game of adding up all the cards, was to go back and take the M-1, F-1, and J-2 data, put them on a curve and watch the curve. The curve seems to be going up a little bit.

MR. WEIDNER: What you are really saying is that this is not your own estimate but the estimate of others.

MR. MULREADY: It is history. This is what NASA is saying to Congress, as of October 1962, as to what it was going to cost to PFRT these engines.

QUESTION: What do you really think the cost will be?

MR. MULREADY: Perhaps figure 21 would help to explain.

MR. TISCHLER: It seems to me that you are saying to us that you are anticipating or we have anticipated the history and you have recorded it. I am not quite sure what we have got.

MR. MULREADY: It depends on how far down the road you are on development. These numbers are not hard to come by. We went into the Congressional Record and took NASA testimony for what it was going to cost to do the J-2, and you usually think when you do this that you can pull the part out. The reason we picked PFRT is that that seemed to be the easiest milestone to identify. It is hard to identify any beyond that, and facilities usually get separated out. We don't think that propellants are included in most of those numbers. So we have taken most of those numbers and put them on a chart. As of last October we had this center black point, which we knew was history, the estimate of the J-2, and the points for the M-1, and the F-1.

We have since put the two stars on which we think reflects a change since then. But we are not sure that our intelligence is good enough to make it accurate. One of the things we think is significant is that a lot of bids for these engines are a factor of 2 off what they cost now.

What we said is let's start with this as a basis and draw a line through it. If we are talking about this size engine, there seems to be enough good data here that we are not smart enough to second guess what has happened, but this has happened in a number of cases so we think in this business it is a reasonably good estimate at this stage of the game.

COMMENT: This is not really based on any plan, program, or anything of this nature. It is just based on past history as you could derive it.

MR. MULREADY: Right. It appears to us to be reasonable for program estimates we have made. Our estimate for a 200K development turned out to be right on the one that was the number for the J-2 as of October 1962. This is also the case for our estimate of what the M-1 would cost. We did that cost estimate in great detail. This is a long expensive process. We said, "Let's not do it again, let's take a curve that goes through a size class and probably no one can really defend one way or the other if it is going to be far off, and most of the estimates have been off a factor of 2 for the winners." That is the basis.

MR. THOMPSON: It looks like you might be defeating your own purposes here. It looks like if we leave you alone and don't fund you at all you will have the 3,000 P_c engine working soon.

MR. MULREADY: I wouldn't bank on that. We have been doing some Indian wrestling with the management and we say help is right around the corner. The first two firings we made on the cooled chamber are a good example. For the first one, we had an injector failure that caused damage to the wall. Most of it was satisfactory. We were satisfied that was good, but we didn't want to talk about it then. On the second one we had a test stand malfunction that caused the system to be damaged, and we had just about run out of money at that stage and had to go back and get some more, which fortunately they gave us. I don't think we can count on that forever.

MR. PAUL, Marshall: Can you comment on what you intended to do in order to provide heat for your pressurization system?

MR. MULREADY: We haven't approached it strictly from that direction. Our major effort was looking at providing boost pumps driven by the engine system which would require practically no pressurization from the engine for net positive section head, so we would be able to take essentially saturated liquid on board. We have not yet done this. We will do this.

There are places where we can provide pressurization directly off the preburner side to pressurize the hydrogen part of the system. We have to take some water out of it, or maybe a heat exchange loop using hydrogen. We have not done that yet.

MR. BARTZ, JPL: I would like to ask the same question I asked of Aerojet this morning, and that is the strength of your conviction about the propellant combination oxygen and hydrogen which seems to be contrary to what we heard from at least one of the contractors on the vehicle.

MR. MULREADY: We have looked at the economics. The way we looked at it, if you look at large vehicles carrying a fair amount of payload and being reusable, if we want to get to the position of doing this, the economics of the situation will be a serious concern. We have examined to the best of our ability, going as far as we can in the design of the systems; we can compare propellant combinations and engine design pretty well. For the kind of engine performance that we think is going to be required I don't think we are going to be satisifed in this time period with moderate, mediocre performance. You are going to want pretty good combinations. This O_2-H_2 combination to us looks like the best economic bet. Fluorine and hydrogen and mixtures of oxygen and fluorine look good. But the economics kind of argue against it.

From the engine-man point of view hydrogen as a coolant for a high performance engine does things that we haven't been able to accomplish with a lot of other propellant combinations.

MR. BARTZ: Are you looking at the economics from the propulsion man's viewpoint?

MR. MULREADY: From an operational viewpoint.

MR. BARTZ: Perhaps Congress is looking at it from the vehicle standpoint? Might not this cause the difference in the conclusions? We are both talking economics. But are we talking about the same economics?

MR. MULREADY: I am not sure that we are. I am not sure what there is yet.

QUESTION: I think the difference in the two results is basically a difference in ground rules. Pratt & Whitney was using a one-stage-to-orbit module and GD/A was using a two-stage vehicle. Mr. Mulready, in your performance estimates or performance tests of plug cluster, have you ever tested this thing with external flow simulating flow around the vehicle?

[redacted]

MR. MULREADY: No. I think that is planned as part of the current program.

QUESTION: You have done some preliminary tests?

MR. MULREADY: Yes.

MR. THOMPSON: Some preliminary tests indicated, I think, a half-percent loss. At that time the aspect of shrouds and other things had not been considered in the vehicle configuration. I think the present study will give us the answers.

MR. MULREADY: With regard to the secondary flow on the outside, looking at how the vehicle would fly, it seems that by the time the velocities are high enough for us to be concerned seriously about this, the pressure ratio is already high enough so that it is not a problem any more. It looks like you skirt the problem.

MR. ATHERTON, Pratt & Whitney: Maybe I can shed a little light on this external flow. I think the Lewis Center has done considerable work in this area, which is worth mentioning, testing both completely internal expansion and combined internal-external expansion plus clusters. They found that the amount of internal expansion was significant in the effect that it had on performance reduction with external flow. For a modular type construction, with perhaps 20-to-1 area ratio internal to 100-to-1 overall, the more the internal expansion provided the more the effect of external flow is reduced, chiefly because the boattail angle is a lot lower in the back.

MR. BARTZ: I didn't quite follow the configuration of the spiral cooling disks. If I understood correctly, the spiral must intersect the plane of the wall at one point - the spiral has to run out at one point.

MR. MULREADY: Yes.

MR. BARTZ: Does that not mean that you are essentially injecting your coolant at one point and not circumferentially uniform?

MR. MULREADY: Those involute grooves cut into the surface were cut all the way around, very close together. It is a way of making up the porous material that you divide up axially.

MR. WEIDNER: Do you have a sample?

MR. MULREADY: Yes.

COMMENT: In this particular case there are 35 involutes and they are 0.020 of an inch apart, very little difference.

MR. SLOOP: I realize that your technique is experimental at this point. When you said it was competitive to other methods, what did you mean?

MR. MULREADY: We did two designs all the way through. We did them at a 250,000-pound-thrust size, and the transpiration cooled, using Rigimesh, all

[redacted]

stainless steel, was on the order of 170 to 180 pounds. This design was on the order of 60 to 70 pounds heavier. So it is somewhat ~~heaver~~, as well as we know how to do it now. This could change depending on the temperature limit of the material that we used.

MR. SLOOP: To what area ratio did you go for this?

MR. MULREADY: 3 to 1. We would carry it to 4 to 1 on the flight-weight design because that would drop the heat-transfer rate to a value that can be handled by regenerative cooling.

MR. ATHERTON: For the benefit of John Sloop, the weight of that chamber compared to the Rigimesh type construction makes less than a 4-percent change in the engine weight. It is a little heavier but it is less than a 4-percent change.

MR. MULREADY: In the plug-in cycle the preburner, which is the low-mixture-ratio combustion section, is located inside the transition case. The discharge of the 1,300° gas from the preburner system provides the gas flow which flows down through the two turbines which are plugged in; the turbine packages are plugged in so there are no hot lines between the preburner and main chamber. All the lines are either ambient or cold, the cryogenic lines, and the engine surfaces are cold.

MR. THOMPSON, Marshall: I believe the longest duration I have heard before on high-pressure combustion hardware was approximately 0.9 second.

MR. MULREADY: That is correct. Three seconds is an eternity. The erosion that we suffered is a very, very small quantity of copper that we lose in a fraction of a second.

[redacted]

TABLE I

ADVANCED BOOSTER TECHNOLOGY

ITEM	OBJECTIVE	STATUS & CURRENT EFFORT	RECOMMENDED EFFORT
PLUG CLUSTER NOZZLE	Cv = 0.975 FOR 10% LENGTH	Cv = 0.965 NAS8 - 11023 WILL CONTINUE PROGRAM	HOT FIRING TESTS
TVC			COMPREHENSIVE ANALYTICAL STUDY-NOVA CLASS VEHICLES
A) SECONDARY INJECTION	ADEQUATE RESPONSE AND EFFECTIVE ANGLE MINIMUM PERFORMANCE LOSS	A) SATISFACTORY SCALE MODEL PERFORMANCE - DEMONSTRATION ON RL10 - NAS8 - 5070 PLUG CLUSTER TEST - NAS8 - 11023	HOT FIRING MODEL TEST PROGRAM TO BE DEFINED FOLLOWING CURRENT EFFORT
B) DIFFERENTIAL THROTTLING		B) PRELIMINARY TEST AND ANALYSIS SATISFACTORY - NAS8 - 11023 WILL CONTINUE PROGRAM	
C) COOLED FLAPS		C) NAS8 - 11023 WILL CONTINUE PROGRAM	

TABLE II

ADVANCED BOOSTER TECHNOLOGY

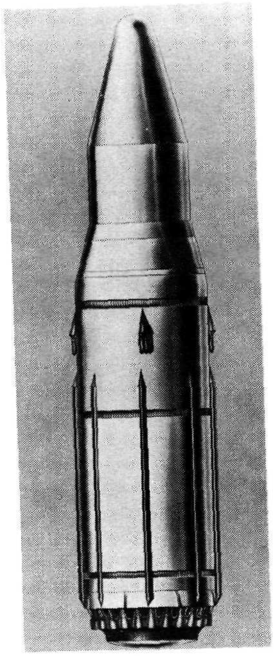
ITEM	OBJECTIVE	STATUS & CURRENT EFFORT	RECOMMENDED EFFORT
COMBUSTION A) PREBURNER	LIQUID FUEL INJECTION O/F = 1.0 $P_c \cong 5000$ PSIA FLAT TEMP PROFILE	NO TEST WORK- DESIGN LAYOUT & ANALYSIS INDICATE PRACTICAL DESIGNS POSSIBLE	A) TEST PROGRAM AT 10,000# THRUST SCALE
B) MAINBURNER	HOT GAS FUEL INJECTION O/F = 7.0 P_c = 3000 PSIA η_{c*} = 98% +	TEST η_{c*} = 95% + AT P_c FROM 2500 TO 3500 PSIA	B) TO BE COVERED UNDER C)
C) TANDEM	A) + B)	NO TEST EFFORT- COMPLETE ENGINE SYSTEMS HAVE BEEN DESIGNED	C) 10,000# THRUST TANDEM COMBUSTION TESTS
D) STABILITY	ADEQUATE DAMPING	NAS8-11024 STABILITY TESTS NAS8-11038 DAMPING ANALYSIS	D) DAMPING DEMONSTRATION TESTS
E) IGNITION	HYPERGOLIC IGNITION WITH O_2 - H_2	HYPERGOLIC IGNITION DEMONSTRATED WITH 1) 0.05% O_3F_2 IN LO_2 (NAS8-2690) 2) METAL POWDER IN LH_2 3) CATALYSTS (NAS8-2690)	E) TEST OF METAL POWDER IN LH_2
COOLING A) FILM-TRANSPIRATION	½ OF 1% I_{sp} LOSS FOR 1 MILLION POUND THRUST ENGINE AT P_c = 3000	DEMONSTRATED RELATIVE COOLING EFFECTIVENESS AT 10 K NO DIRECT SUPPORT	CONTINUED COOLING TEST IMPROVED MATERIALS
B) REGENERATIVE	12 BTU/INCH²/SEC	RL10 EXPERIENCE	FABRICATION FEASIBILITY PLUG SECTOR

[redacted]

TABLE III

ADVANCED BOOSTER TECHNOLOGY

ITEM	OBJECTIVE	STATUS & CURRENT EFFORT	RECOMMENDED EFFORT
TURBO—MACHINERY			
A) FUEL PUMP	50K SIZE η = 58%	η=54% AT 4000 PSI NO CURRENT DIRECT SUPPORT	PERFORMANCE OFF DESIGN THRUST BALANCE
B) LOX PUMP	50K SIZE 1000 GPM, 5000 PSI	800 GPM 4200 PSI NO CURRENT DIRECT SUPPORT	PERFORMANCE AND MECHANICAL EVALUATION
C) LOW SPEED INDUCER	MINIMUM NPSH (UNDER 1 FT AT 1G)	WATER FLOW TEST NO DIRECT SUPPORT	SCALE TEST O_2 & H_2
D) BEARINGS	ROLLING ELEMENT 2×10^6 DN HYDROSTATIC - ?	2.4×10^6 DN ROLLING ELEMENT 4×10^6 DN HYDROSTATIC NO CURRENT DIRECT SUPPORT	INVESTIGATION OF LIFE/LOAD AT HIGH DN
E) SEALS	17000 FPM RUBBING VELOCITY	SEALS OPERATE IN H_2 AT THESE VELOCITIES NO CURRENT DIRECT SUPPORT	MATERIAL & CONFIGURATION TESTS
F) TURBINE COOLING	1700° DISC AND ATTACHMENT THERMAL GRADIENT	NO CURRENT DIRECT SUPPORT	SPIN TESTS UNDER SIMULATED ENVIRONMENT



INSTALLATION OBJECTIVES

- DIRECT THRUST STRUCTURE
- HIGH AREA RATIO IN SHORT LENGTH
- ALTITUDE COMPENSATION
- VARIABLE THRUST
- HIGH MISSION SUCCESS PROBABILITY
- INSTALLATION FLEXIBILITY
- ACCESSABILITY
- REUSABILITY

Figure 1

TRUNCATED PLUG INSTALLATION

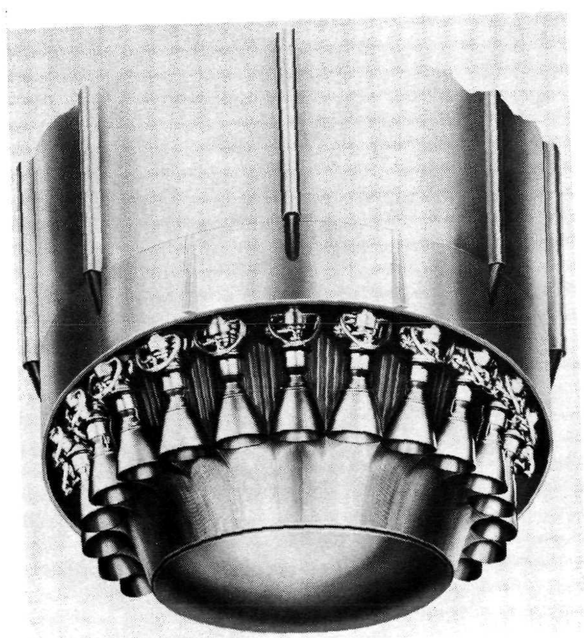


Figure 2

MODULE DEFINITION

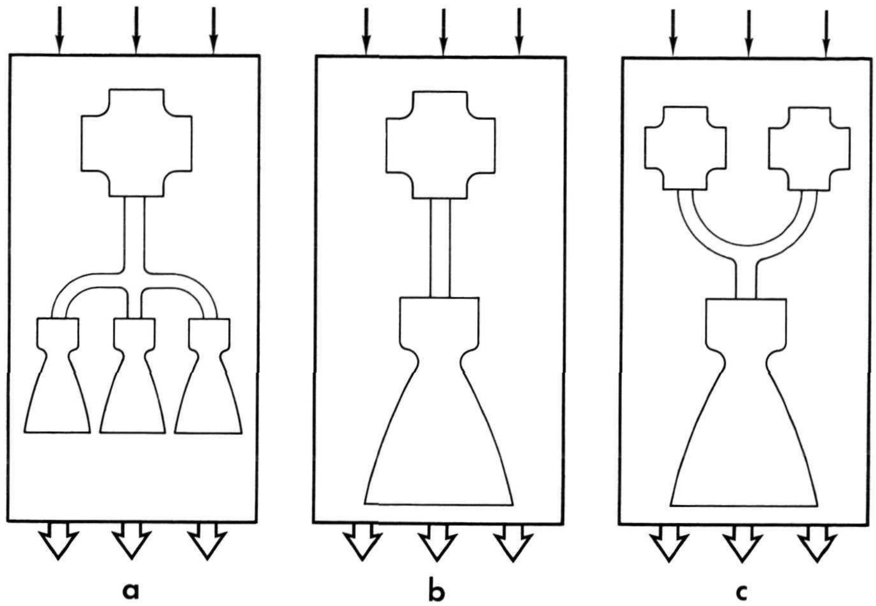


Figure 3

ESTIMATED SYSTEM RELIABILITY VS MODULE THRUST SIZE

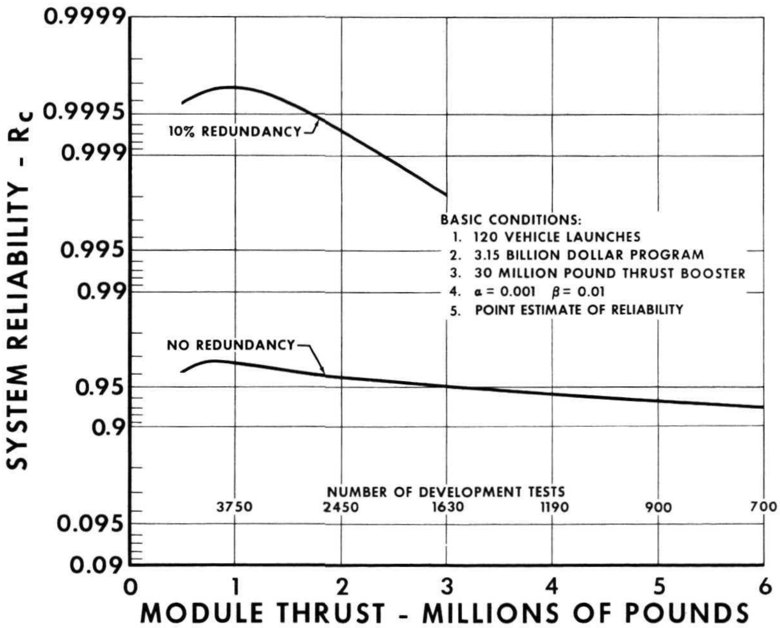


Figure 4

PROPELLANT FLOW SCHEMATIC - TANDEM COMBUSTION

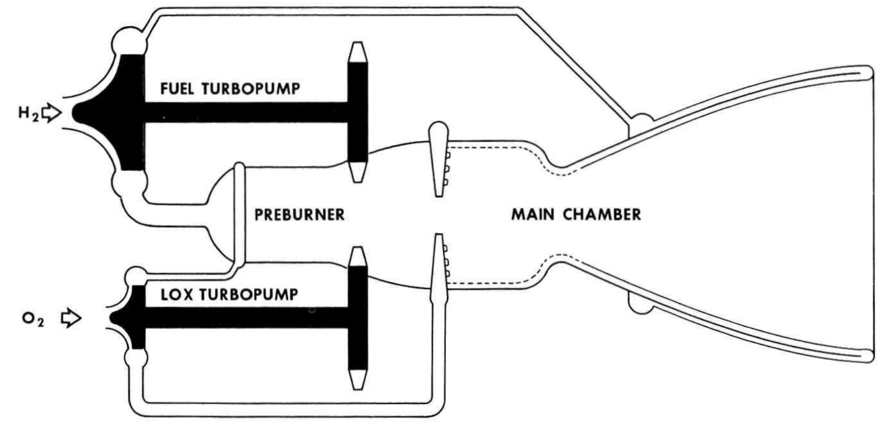


Figure 5

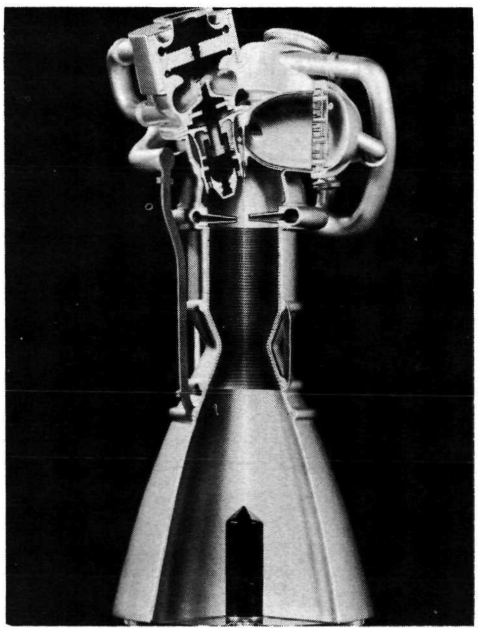


Figure 6.- Engine model.

COOLED PLUG SCHEMATIC

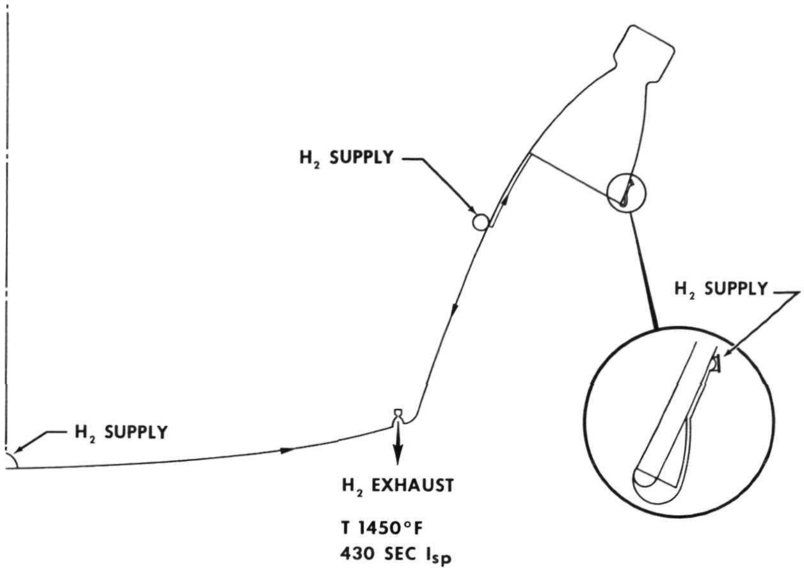


Figure 7

PERFORMANCE OF 24 MODULE PLUG CLUSTER NOZZLE

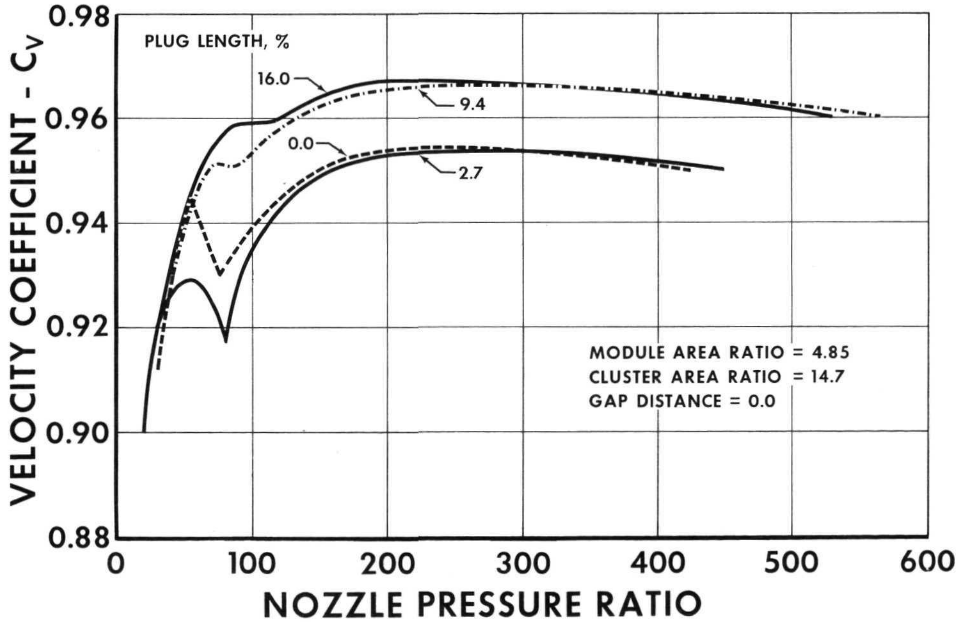
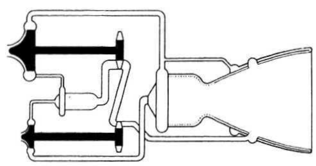


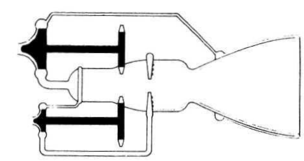
Figure 8

CYCLE COMPARISON

3000 Pc ϵ =98



PARALLEL COMBUSTION
(GAS GENERATOR)



TANDEM COMBUSTION
(PREBURNER)

	GAS GENERATOR	PREBURNER
PERFORMANCE		+ 8 SEC
WEIGHT	10%	
SIZE	SAME	SAME
BULK DENSITY		+ 8%
COMBUSTION STABILITY		BETTER
PUMP ΔP	LOWER	
CONFIGURATION		SIMPLER

Figure 9

EFFECT OF CHAMBER PRESSURE ON PAYLOAD

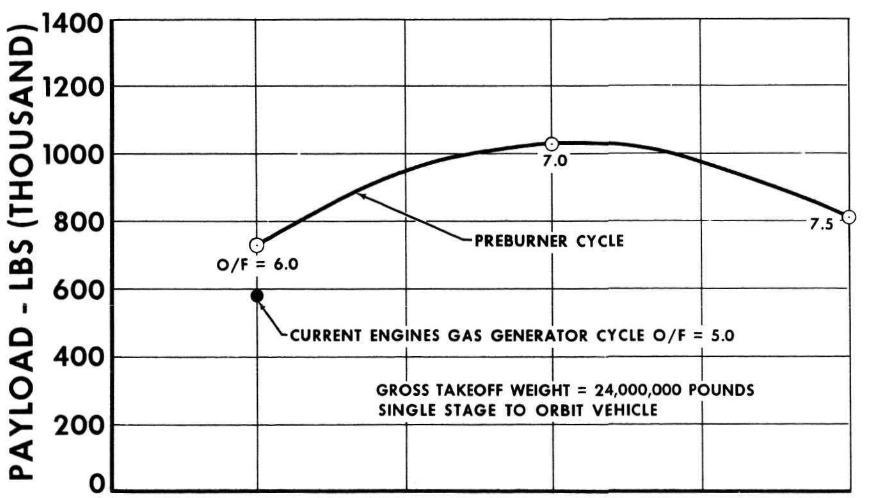


Figure 10

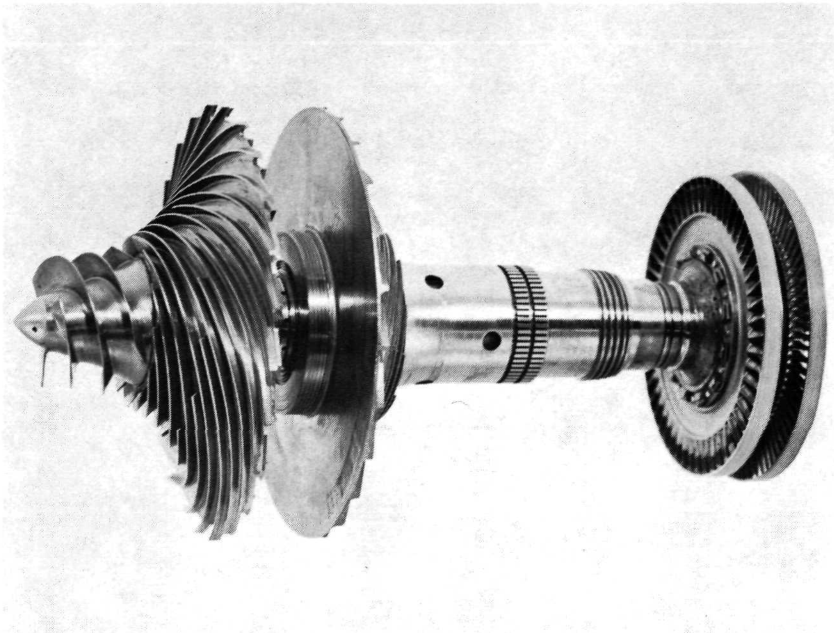


Figure 11.- Pump developed by Air Force.

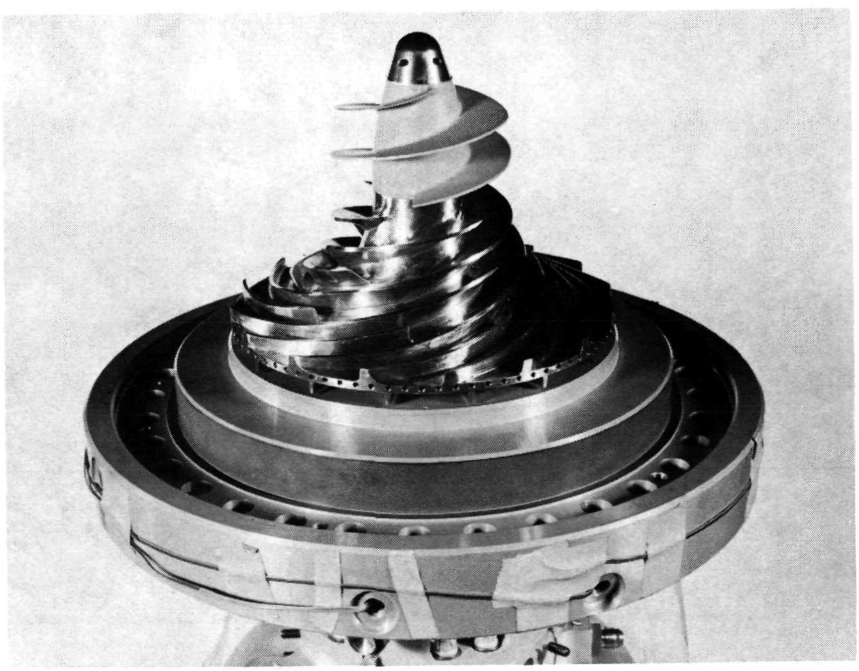


Figure 12.- Oxygen pump.

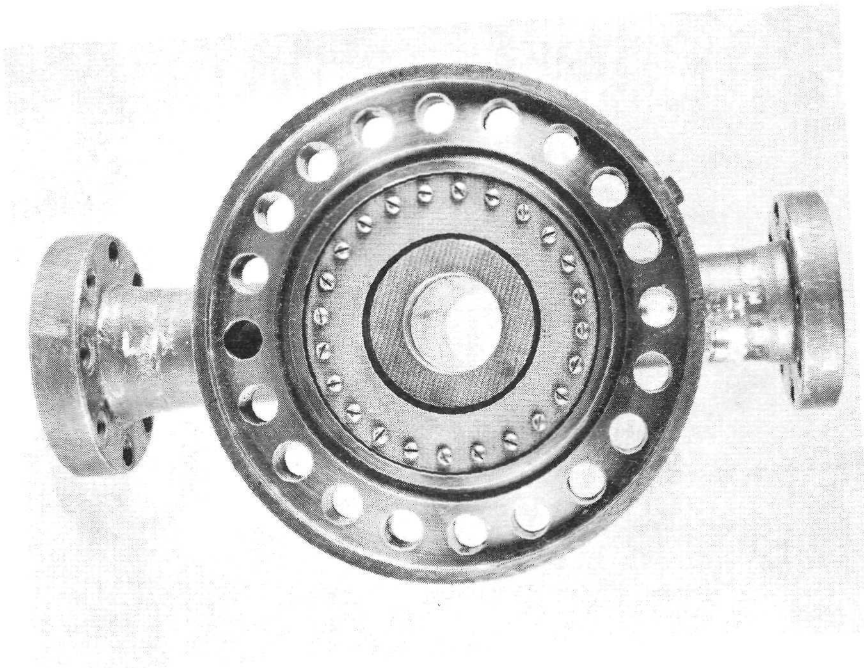


Figure 13.- Injector developed by Air Force.

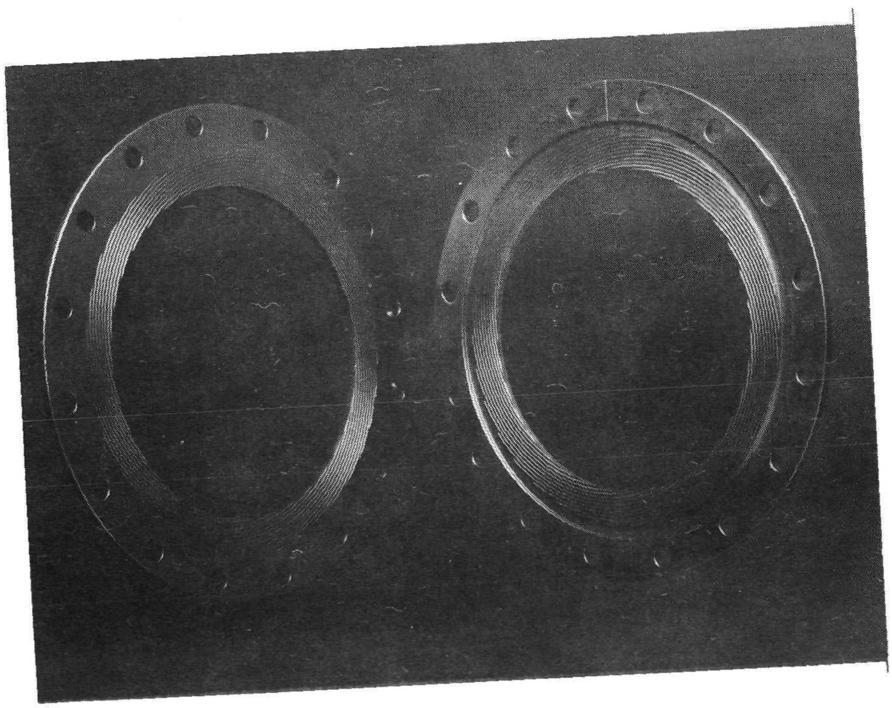


Figure 14.- Copper disk.

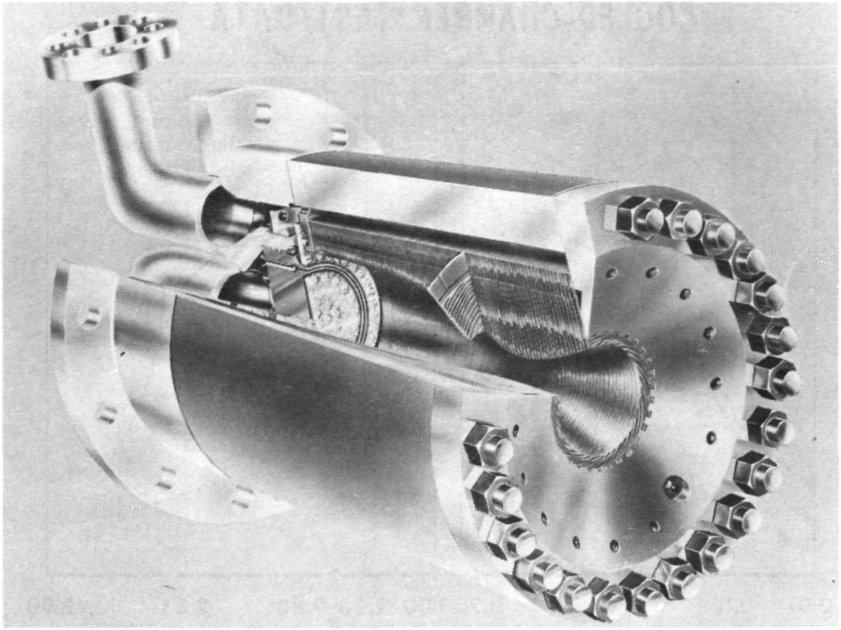


Figure 15.- Artist's concept of test rig.

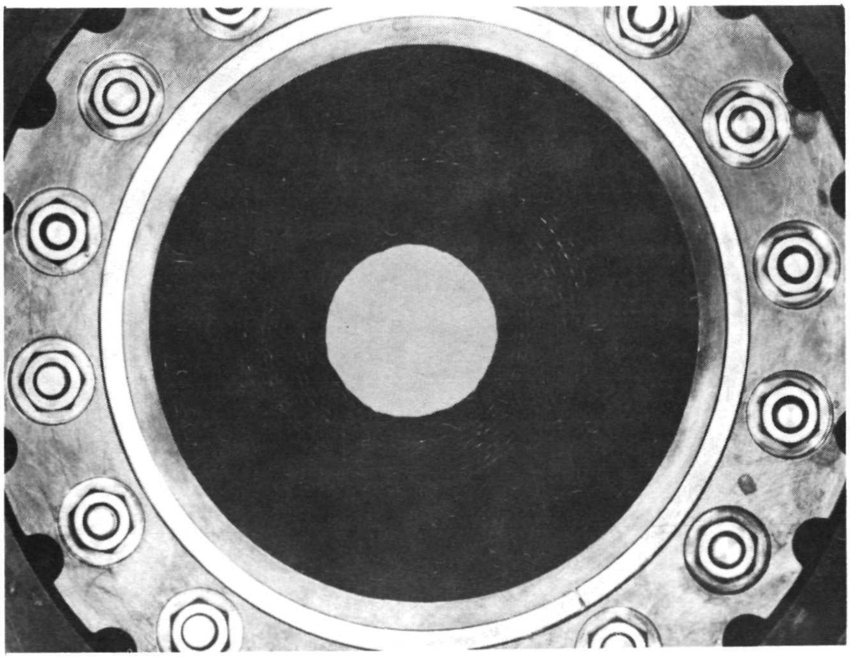


Figure 16.- Photograph of throat.

[redacted]

COOLED-CHAMBER TEST DATA

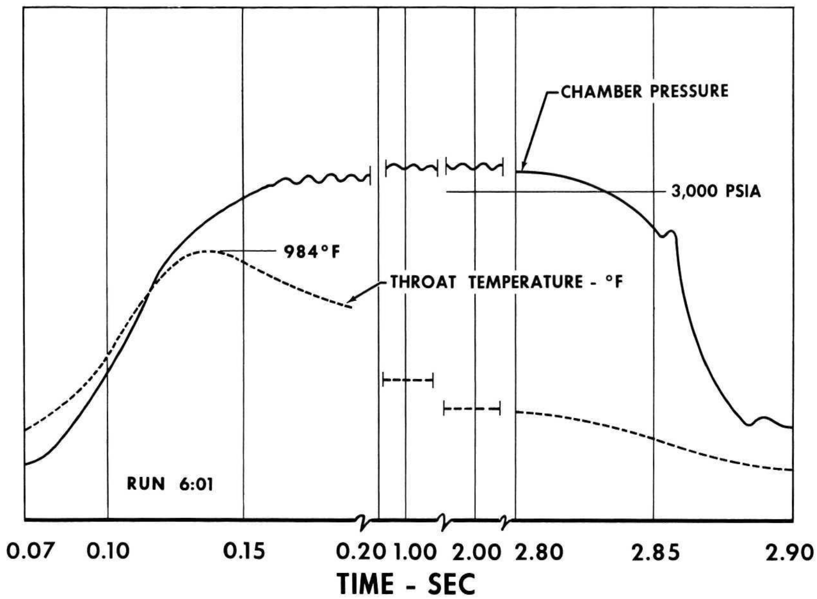


Figure 17

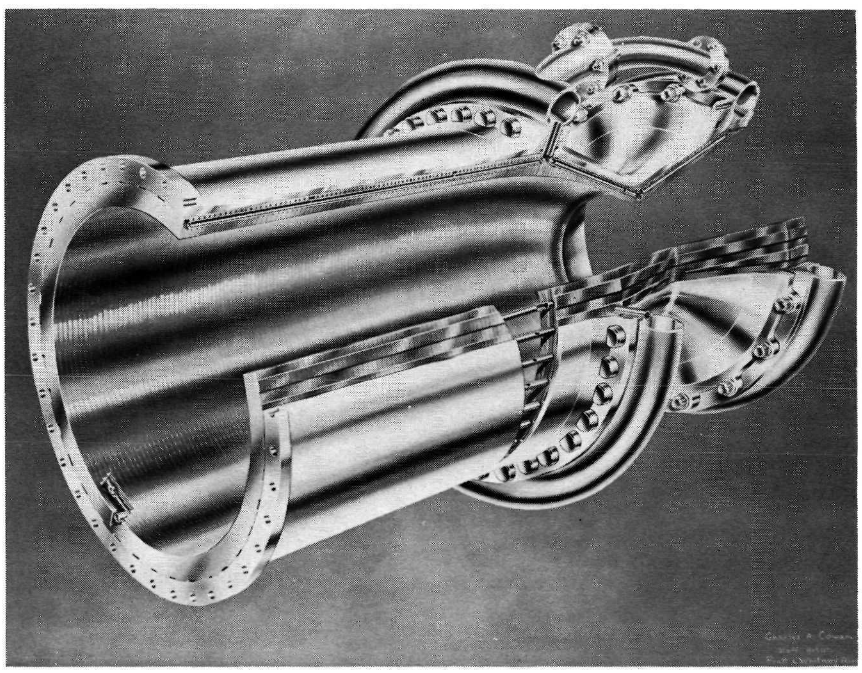


Figure 18.- Flight weight design - wafer cooling technique.

[redacted]

HIGH PRESSURE ENGINE DEVELOPMENT PLAN

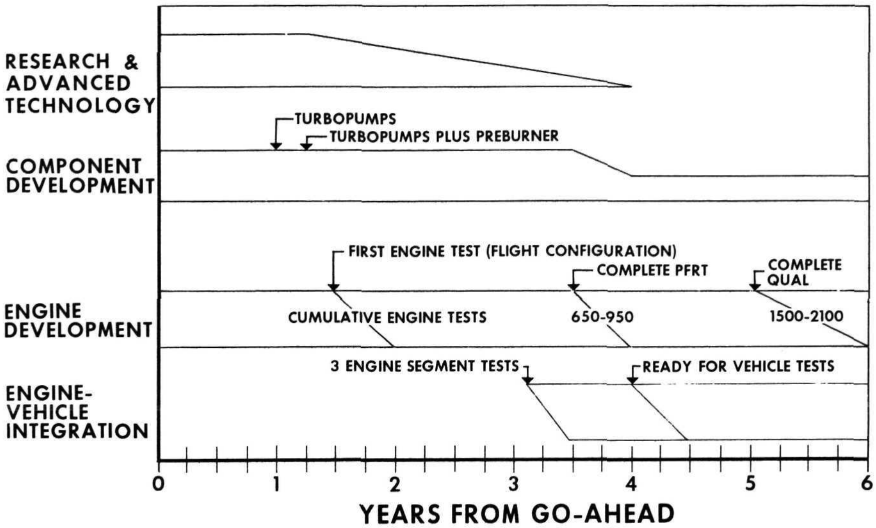


Figure 19

DEVELOPMENT PROGRAM FUNDS REQUIRED

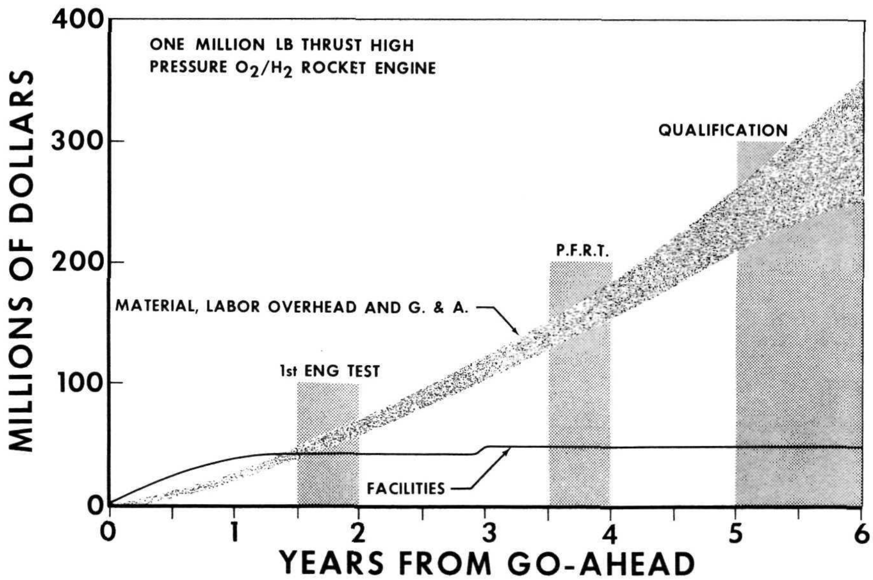


Figure 20

[redacted]

DEVELOPMENT COST THROUGH PFRT

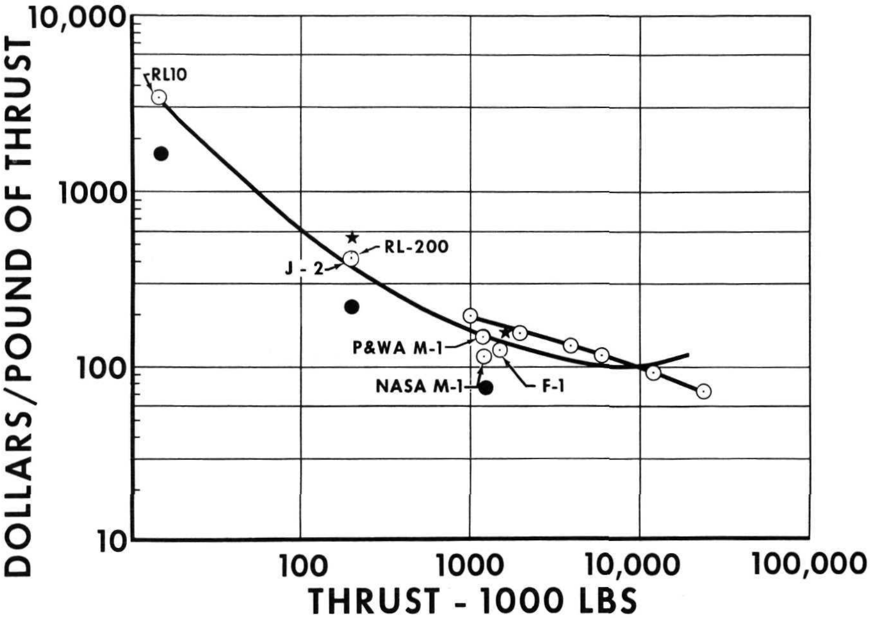


Figure 21

10. ENGINE SYSTEMS

Part I

By Sam F. Iacobellis

Rocketdyne Division
North American Aviation, Incorporated

Before showing our recommended engine configurations it might be well to discuss some of the pertinent technology areas.

Our first area is advanced nozzles. During the recent years there has been considerable work addressed to the application of annular nozzles to rocket engines. The peculiar expansion process of the annular nozzles allows a high efficiency for relatively short length expansion surfaces. Furthermore, the annular nozzles lend themselves quite well to segmentation. Segmentation in some form we feel will be necessary for the thrust levels that we are speaking of now.

Figure 1 shows two wind-tunnel schlieren photographs of an RF nozzle. The photograph on the right was taken at a high pressure ratio. Note that the gas is fully expanded and flowing in the axial direction. The nozzle efficiency at high pressure ratios is something like 99 percent. The schlieren photograph on the left was taken at a low pressure ratio. In this case the gas is prevented from fully expanding by the base pressure and there is a degree of altitude compensation.

In addition to numerous wind-tunnel tests, we have hot fired two annular thrust chambers. We have fired a 50K storable propellant E-D chamber and a 10K lox/hydrogen R-F chamber under Air Force support. The 10K chamber is shown in figure 2 contrasted to an 80-percent-length bell chamber of the same area ratio. This model was fired at Rocketdyne at low pressure ratios and is presently in the Lewis Research Laboratory tunnel for altitude data points. I believe more on this work will be discussed in Mr. Beheim's session; however, I would like to bring out a few points at this time. In this figure we have C_T, defined as the ratio of C_F actual to C_F ideal, plotted against pressure ratio. At the high pressure ratios we see that the performance of the short length RF nozzle is equal to or about equal to the 80-percent bell nozzle, and at design pressure ratio and below it is superior to the 80-percent bell.

The point that I think is significant is that this 10,000-pound-thrust engine is no longer than $8\frac{1}{2}$ inches from the throat to the nozzle exit. It is only some 20 percent of the overall length of the equivalent bell thrust chamber. This could mean length reductions of 30 to 40 feet when you speak of the Nova size engine.

I would also like to point out that we feel that the annular nozzle altitude compensation, while providing performance higher than the bell nozzle, is not high enough. The work done at Lewis a couple of years ago by Connors and others showed something like 98 percent of ideal nozzle performance. That was when the whole center body was open to ambient air. I am sure some of that work will also be discussed in subsequent papers. We are of the opinion that the degree of altitude compensation shown in this figure can be improved, perhaps with center body bleed or by other devices.

In our work with annular nozzles we have experienced some structural problems. We have found that it is very difficult to hold the throat gap in the annular nozzles. When the throat area is distributed around the periphery of an annular nozzle the linear throat gap dimension is relatively small. Consequently, a throat tolerance problem is created. Furthermore, it is difficult to design a simple cooling system to cool the throat lip and a simple structural member to resist the bending moment created in the throat region. The same problems are magnified in a spike nozzle configuration because the throat is spread out at a larger radius.

A solution to the structural problems as well as a way to achieve segmentation is the multichamber concept. This is a concept in which the annular throat is divided among discrete ports.

At Rocketdyne we have cold-flow tested spike models with a continuous annular throat, 8 discrete ports, and 12 discrete ports. Figure 3 shows one of the models and the wind-tunnel installation.

We also have tested an E-D type nozzle with a continuous throat configuration, 8-port, and 12-port throat configurations. The E-D wind-tunnel model is shown in figure 4.

The performance of these multichamber nozzles is shown in figure 5. We have normalized the performance and compared it to that for the annular nozzle, so we have C_F multichamber divided by C_F annular for the spike and E-D configurations. The dashed line is for the 8-hole cluster, and the solid line is for the 12-hole cluster. With the spike model we found some 10 to 12 percent loss in C_F. With the E-D configuration, we found 4 percent and 2 percent loss in performance for the 8- and 12-port nozzles, respectively. It is intuitive that more and more throats would improve the performance and cut this deficit down. However, there is a limit to how many nozzles are practical. We believe the practical number is in the region of 8 to 12.

The other concept that we are studying at Rocketdyne is the toroidal thrust chamber, which also solves the structural problems mentioned earlier. In this concept the nozzle is segmented down to its most basic element, the coolant tube. The coolant tubes are formed in a hoop configuration, and assembled as shown in figure 6 to form the combustion chamber. The injector plate is shown. Propellants are mixed and burned in this region and egress through the throat region which was formed by swaging of the individual tubes. A complete toroid is shown in the upper left-hand side of the figure. The torus could be broken up into linear segments as we will show later.

When the concept was first conceived we were concerned about the shock losses, or loss of total pressure, caused by the tubes in the throat region. We studied this analytically and experimentally and found that for the ratios of throat to chamber tube diameters of interest, we were experiencing a 5 percent loss in total pressure. This converts to a 0.4 to 0.5 loss in C_F.

On the benefit side, admittedly somewhat of a serendipity, we found a degree of controlled expansion in the throat region of the toroid. Figure 7 is a cold-flow schlieren photograph showing three circular tubes. The flow accelerated to Mach 1 at the aerodynamic throat which is located near the geometrical throat. The flow continues to accelerate under the influence of the circular-tube surface until a Mach number of approximately 2.2 is reached.

We analyzed this controlled expansion by the method of characteristics. The calculated flow fields for a continuous throat and the toroidal throat nozzle are plotted in figure 8 for a pressure ratio of 20. The continuous throat flow field is shown on the left. The flow is Mach 1 in the throat region and expands until it reaches the base pressure.

With the toroidal configuration, the gas leaves the influence of the throat, or the control surface, at a Mach number of 2.2. Therefore, there is a reduced expansion angle at the throat although the gas senses the same base pressure. The resulting flow field is shown on the right. The net result is that there is a greater degree of recompression on the nozzle wall for a toroidal throat than for a continuous throat configuration at the same pressure ratio and base pressure.

These analytical results were verified by an experimental test program. Our wind-tunnel model is shown in figure 9. It is a two-dimensional slice of an annular nozzle. The model was so designed in order that various throat configurations could be tested with the same nozzle shroud. The model in this photograph has a toroidal throat section with 9 circular tubes installed. Also tested was a rectangular continuous throat section having equivalent throat area allowing us to compare the toroidal throat with the continuous throat configuration.

The model was equipped with plexiglass side plates so we could take schlieren photographs as well as thrust and pressure readings. Notice that this two-dimensional model simulates a slice from an R-F or E-D type nozzle, as well as a spike type nozzle. We think the data obtained are valid for both configurations.

A schlieren photograph of the flow field for the continuous throat configuratio is shown in figure 10. While the pressure is slightly higher, 40 not 20, the point can be made. The flow is at Mach 1 at the throat and expands to angle described by the last left running characteristic line. There is a strong compression near the region that was predicted analytically. Note that the compression and expansion pattern is very similar to the analytical flow field shown in figure 8.

A schlieren photograph of a toroidal throat configuration at a pressure ratio of 20 is shown in figure 11. In this case the flow has reached Mach 2.2 before it leaves the expansion surfaces of the toroidal throat tubes. Consequently, the Prandtl-Meyer turning angle is considerably less and the last left running characteristic line is displaced considerably from that in figure 10. Note that the flow field is almost identical to that predicted analytically. The strong recompression on the nozzle wall is in the region nearer the throat, with compression and expansion regions varying along the wall.

Figure 12 shows some results of the cold flow test program. The experimental C_F values of the two models are plotted as a function of pressure ratio. The solid line is drawn through the continuous throat data points and the dashed line is drawn through the circular pin toroidal throat configuration data points. At the high pressure ratios, there is a loss of about one-half percent in C_F for the toroidal throat configuration, because of the 5 percent loss in total pressure.

In the lower pressure rate region, the beneficial controlled expansion effect resulted in a higher C_F which was significantly higher for the toroidal model.

Figure 13 shows the performance of the multichamber spike, multichamber E-D, and toroidal models plotted on the same normalized basis, the C_F of the multichamber or toroidal model divided by the C_F of the respective continuous throat configurations. For the 8- and 12-port spike configurations, the performance was down approximately 12 and 10 percentage points, respectively, at design pressure ratios and above. In the case of the 8- and 12-port E-D, the performance was approximately 4 and 2 points down, respectively, at design pressure ratios and above. In all spike and E-D multichamber configurations the relative performance was somewhat lower at pressure ratios below design.

In the case of the toroidal model, the performance was down approximately 1 percent at the high pressure ratios, one-half percent due to the total pressure loss mentioned earlier and the other half percent being due to the slight adverse effects of controlled expansion at altitude conditions. At the lower pressure ratios the performance of the toroidal model is significantly better than that of the continuous throat configuration. It can be seen that the toroidal throat configuration provided better relative performance than the multichamber models over the entire range of pressure ratios tested.

So based on performance alone we are quite enthusiastic about the possibilities of the toroidal combustor.

We have hot fired two toroidal models. (See fig. 13.) The model on the left utilized storable propellants and the model on the right utilized lox/hydrogen. The thrust level of both was approximately 6,000 pounds. The chamber pressure of the storable model was approximately 300 psi, and the lox/hydrogen model approximately 450 psi. The models ran stably in both programs and we achieved what we think is pretty high efficiency, although the main test objectives were feasibility demonstrations. The C^* efficiency was 94 percent in the storable case and near 100 percent in the lox/hydrogen program. Both efficiencies were based on frozen equilibrium. The 100-percent C* is quite reasonable when you consider it was based on frozen equilibrium. At the mixture ratio tested, 100-percent frozen equilibrium corresponds to approximately 98 percent of shifting equilibrium.

The main thing about the annular nozzles and the toroidal in particular is their capability of being segmented. (See fig. 15.) We foresee a development program that would lead toward a 24-million-pound engine which could conveniently be broken down into four phases. A considerable amount of development work could be conducted in Phase I with a 100K thrust test segment. Most of the injector, cooling, ignition, and stability development testing would be conducted during Phase I. During Phase II enough tubes would be put together to investigate the coupling effects. This phase of the development could be at the 1 million to 1.5 million thrust level; however, we believe that you could adequately conduct Phase II at the 400K thrust level.

In Phase III a deliverable segment would be developed. It would be this sector that we would develop to PFRT and Qualification. The engine manufacturer would stop his development work at Phase III and deliver the sector to the vehicle people or to NASA, much like we are doing in the case of the H-1.

The main point here is that we are proposing the development of a 24-million-pound thrust engine, or even a 30-million-pound engine, which would have most of the development work conducted at the 100K thrust level. This is more than two orders of magnitude lower in thrust, yet we would be doing development work directly applicable to the end item. There would be no scaling problems or factors to consider as the thrust chamber tubes and injector developed in Phase I would be the actual size used in the final engine.

The parameters of a toroidal combustor for a 30-million-pound engine that has a nominal chamber pressure of 2,500 psia are indicated in figure 16. We have found that because of the swaged portion of the tubes in the throat region, the best pressure vessel configuration is not a circular hoop but an oval one. The 15-inch dimension is the average diameter of the oval. For the 30-million-pound engine the major diameter of the oval would be 18 inches and the minor diameter 12 inches. The other dimensions are shown in the figure. Based on the available data, we believe that 1 percent of hydrogen film cooling will be adequate (1 percent of the total propellant flow).

Figure 17 shows a hard mockup of the Phase I toroidal segment. The segment and tube dimensions are identical to the segment that would be used in the development program for the 30-million-pound Nova engine. This flight-weight model measures only approximately 5 inches × 18 inches × 18 inches but will develop 80,000 pounds of thrust, about 100,000 pounds with a nozzle shroud at 2,500 psi. The hard mockup is made of aluminum and weighs approximately 20 pounds. We recommend Inconel 718, which would increase the weight to some 60 pounds. Still we are talking about 60 pounds, not 600 pounds as you might assume if current thrust-to-weight ratios of conventional engines were used.

The model is equipped with a flight-weight manifold and injector. The oxidizer enters the swirler injector tubes from the lower half of the manifold. The hydrogen enters the upper half of the manifold from the left, leaves through the annulus of the concentric tube injector, and mixes with the oxidizer swirling out of the center. The injector design is much like the J-2. Rigimesh is employed for cooling the injector face.

Going back to the development sequence, we think that you can do much of the development work with a model of this size and bring the reliability up to a high level with a reasonable-cost test program. Let me reiterate that while this work would be done at more than two orders of magnitude lower in thrust level, the Reynolds numbers, combustion temperatures, and so forth are the actual ones that will be experienced in the 30-million-pound engine.

Another nozzle configuration that has evolved at Rocketdyne is what we call the aerodynamic nozzle. Figure 18 shows two aerodynamic nozzle concepts. The one on the left is the concentric tube concept. Briefly, primary flow comes through the center tube, chokes at the exit, and continues to accelerate against the secondary flow. In most cases the secondary flow also goes supersonic. We have tested this concept with both cold and hot flow models and achieved overall efficiencies as high as 92 percent C_F (compared with 0.983 for a 15° half-angle cone) with approximately 10-percent secondary flow. We also achieved about 90 percent C_F efficiency with no secondary flow.

The concentric tube concept is interesting; however, the configuration that looks more attractive to us for liquid engines is the aerodynamic spike shown on the right. In this case the primary flow is directed around the periphery and expands against the secondary flow in the center to form an aerodynamic contour much like an actual spike metal contour.

To date, we have optimized two of the five parameters that appear important, the primary flow discharge angle β and the amount of secondary flow for a given area ratio. We found that 45° was the best β angle after testing 15°, 30°, 45°, and 60°. We have found that secondary flows on the order of 2 to 5 percent seem best.

Some experimental data for a β angle of 45°, and for 5 percent secondary flow, are shown in figure 19. On the left-hand side of the figure is a schlieren photograph that shows the formation of the aerodynamic spike. This model had an area ratio of 25:1 and a β angle of 45°. This photograph was taken when the secondary flow was approximately 5 percent and the pressure ratio was 491. Note the supersonic pattern of the primary flow as it leaves the throat and the converging subsonic secondary flow, which also chokes a short distance downstream.

The combustion efficiency ratio is plotted against pressure ratio. At the design point we have increased the performance from 93 percent with zero secondary flow to 96 percent with approximately 5-percent secondary flow. The performance with approximately 2-percent secondary flow also looked almost as good.

These data can be looked at in another light. Let us compare the aerodynamic spike with a typical system like the F-1. Because of the gas-generator flow, the specific impulse of the F-1 engine system is approximately 98 percent of the reference thrust chamber I_s. After the gas-generator flow leaves the turbine, it is somewhat surplus and is discharged as conveniently as possible, mainly in such a manner as to avoid a recirculation problem. Maximum performance recovery usually is not the governing factor. In the F-1, the gas-generator flow is discharged in the nozzle a considerable distance downstream.

If the gas-generator flow is available as secondary flow for the aerodynamic spike nozzle, the performance values quoted in connection with figure 19 can be increased. By discharging the gas-generator gases in the center body of the aerospike, the secondary flow can be considered somewhat gratis. On this basis the I_s of the aerospike engine would be 0.977 of the reference, as opposed to 0.98 for the conventional engine. At first it appears like good salesmanship to compare the systems this way. In many respects it looks like it holds.

In our cold-flow tests, the secondary flow was taken from the main chamber, or the plenum, then its pressure was reduced to a value on the order of 0.03 of P_c before it entered the center body cavity. This pressure is about the same as that available after the gas-generator gas goes through a conventional turbine cycle. It should be noted that in figure 19, our C_F calculations were based on the total flow rate, $\dot{W}_P$ (primary) and $\dot{W}_S$ (secondary). In this case, 5 percent was actually taken from the primary flow and diverted as secondary flow and the result was a 3 percent increase in performance. In a way we would have been pleased with just the same 93 percent because we were taking high pressure gas and diverting it to the center body as secondary flow.

In summary, the 96-percent performance looks promising, especially after examining only two of the aerodynamic spike parameters. In addition, 2 percent of the secondary flow might come somewhat as a gift if the gas-generator gases are discharged in the center body of the aerodynamic spike nozzle.

Another attractive configuration that combines the toroid and the aerodynamic spike nozzle is shown in figure 20. Additional control surface is added to the toroidal combustion chamber to provide an area ratio of approximately 16 to 1. The engine configuration is shown in the lower left-hand portion of the figure and consists of 10 linear segments of toroidal combustors. The toroidal segments are designed to rotate so that at sea level and low pressure ratios the toroidal chamber throats would discharge in the axial direction. Therefore, the C_F versus pressure ratio would approximate a 16:1-area-ratio bell nozzle. At the desired altitude the linear segments would be rotated inward so that an aerodynamic spike nozzle of higher area ratio, say 100:1, would be formed. Therefore, the preferred portions of the C_F versus altitude of both a 16:1 two-dimensional bell nozzle and a 100:1 aerodynamic spike nozzle are utilized as shown in the qualitative plot in the lower right-hand portion of the figure.

Numerous studies on chamber pressure optimization are summarized in figure 21. The results of a chamber pressure optimization for an O_2/H_2 single stage to orbit vehicle are plotted in this figure for two cases. The results on the left were calculated for engine weights based on today's technology and on the right, based on advanced technology. Curves of percent change in payload versus P_c are plotted in both cases for bell and annular nozzle configurations. Each curve actually represents the peaks of a family of curves of constant area ratio.

Based on today's weight technology, a chamber pressure of approximately 2,200 psia is near optimum for both the bell and annular nozzle configurations. Gains of 8 percent and 30 percent are shown for the bell and the annular nozzle, respectively, over a bell nozzle operating at a reference chamber pressure of 1,000 psia. Here we assume full altitude compensation - not 100 percent, but we have assumed that P_{base} equals $P_{ambient}$. If we assume engine weights based on advanced technology, the maximum payload changes occur at 2,800 to 3,000 psia. Payload gains as high as 40 percent are possible if we could achieve these lighter engine weights. It should be noted that base pressures were assumed equal to ambient pressure for the annular nozzles in these studies. We might summarize here that 2,200 to 2,500 psia appears to be a good chamber pressure goal. Certainly the optimum lies between 2,000 and 3,000 psia.

I believe that it should also be pointed out that Rocketdyne, as well as the other engine manufacturers, have been quite optimistic about the film cooling potential of hydrogen. Because we have exceeded the capability of the regenerative cooling techniques at these chamber pressures, we are depending on film cooling. These curves do not include any performance degradation because of the film cooling as hydrogen appears to be an excellent film coolant. However, if we do suffer a performance loss, the optimum chamber pressure, of course, would be lower.

Cycle analysis is another pertinent area. The question of cycle selection has been with us for the last decade at least. We have conducted extensive cycle comparison studies for the J-2, and earlier for the F-1, and I have read where similar studies were conducted for the V-2 engine.

The two main candidate cycles, the gas generator and the topping, are shown schematically in figure 22.

The secondary flow paths are represented by dashed lines in both cycles. In the gas-generator system, depending on the P_c, we take something like 1.5 to 3 percent of the main propellant flow and route it to the gas generator. The gas-generator flow drives a high-pressure turbine, either in series or in parallel. It is usually then discharged overboard via the main thrust chamber.

In the basic topping cycle, for chamber pressures above 1,100 or 1,200 psia, it is prudent to route all of the available hydrogen fuel to the gas generator or preburner. There it is mixed and burned with an amount of oxidizer dependent on the turbine design temperature, usually in the range of 1,200° to 1,700° F. The gas-generator gases drive a relatively low pressure ratio turbine and are then injected with the remaining oxidizer and burned in the main combustor. The topping cycle does offer the possibility of higher performance because all of the gases are dumped into the main combustor.

The big thing that concerns us in the topping cycle is the required pump discharge pressure to sustain a given chamber pressure. It is much higher in the case of the topping cycle than it is in the case of the gas generator. Furthermore, this high-pressure fluid must be burned in this fairly substantial topping cycle gas generator. In the gas-generator cycle on the left, something like 2 percent of the flow is burned in the gas generator, whereas in the topping cycle gas generator something like 40 percent of the total weight flow is required. Because it is mostly hydrogen, the flow ratio is higher volumewise.

One might argue in favor of this system because the hydrogen is truly in the gaseous phase, and this would help combustion stability in the main chamber. This is true. But up here in the gas generator or preburner we are still dealing with liquid propellants and you might just be transferring the problem. The temperature is down in the gas generator but the advantage might be offset by the significantly higher combustion pressure. So you might be just transferring the stability problem from the main chamber to the precombustor.

Admittedly most of the previous comments in opposition to the topping cycle have been qualitative in nature. Figure 23 contains some quantitative data. The required pump discharge pressure versus chamber pressure is plotted in this figure for both the topping cycle and the gas-generator cycle. Two sets of curves are shown for the topping cycle. The left two curves are for relatively low pump and turbine efficiencies and the right two curves are for relatively high pump and turbine efficiencies. The two curves shown dashed are for a turbine temperature of 1,200° F and the solid curves are for 1,700° F. Film cooling was assumed to minimize pressure drops.

Two curves are shown for the gas-generator cycle. The dashed curve represents the required pump discharge pressure if regenerative cooling were the only cooling technique employed, whereas the solid line represents combined film and regenerative cooling.

Assuming the high efficiencies and a relatively high turbine temperature, it can be seen that with the topping cycle a pump discharge pressure of approximately 6,000 psia is required to sustain a chamber pressure of 3,000 psia. In the case of the gas-generator cycle, on the other hand, a chamber pressure of 3,000 psi can be sustained with a discharge pressure of approximately 4,100 psia. We think that this higher pump discharge pressure is quite important. Furthermore, the sensitivity or influence coefficients of the topping cycle will be of the first order. Note the slope of these constant efficiency curves. If the efficiency is missed by a couple of points, it will switch you quite a way down in P_c. It would be quite difficult to develop a thrust chamber simultaneously with pumps because of the uncertainty in pump and turbine efficiencies.

Another way to look at that comparison is to take the heart of the rocket engine, the pump, and hold it constant. Let the pump discharge pressure be equivalent, say 5,000 psia. With a topping cycle a 5,000 psia pump discharge pressure would provide a P_c of about 2,500 psia; with the gas generator a 5,000 psia pump discharge pressure would sustain a P_c of something like 3,700 psia. With the higher area ratio that it would allow, the 3,700 psia chamber pressure and gas-generator cycle could provide a higher specific impulse than that provided by the topping cycle operating at a chamber pressure of 2,500 psia. At this time, Rocketdyne feels that the gas-generator cycle is the preferred system even for high P_c.

Air augmentation is the next area that we would like to discuss. Figure 24 perhaps best explains the interest in this area. For an ideal mass increase, the thrust augmentation ratio is plotted as a function of velocity ratio V_o/V_r for constant ratios of air to rocket gas flow rates. For ratios of air to rocket gas of 1, 10, 20, and 30, air augmentation ratios of roughly 1.4, 3, 4, and 5, respectively, are shown at low rocket velocities. As the vehicle velocity V_o approaches the rocket gas exit velocity V_r the augmentation ratio decreases. At the limiting case of vehicle velocity equal to rocket gas velocity, the augmentation ratio reduces to unity. The basic problem involved with air augmentation is whether the required mixing duct can be made light enough so as not to overcome the specific impulse gains.

The result of some work conducted at the Martin-Marietta Company 2 years ago is shown on the left in figure 25. A mixing duct with a length-to-diameter ratio of 4 was assumed for a Titan type booster with two nozzles. The object of this figure is to suggest the possibility of using advanced nozzles, the toroid shown on the right, to reduce the required mixing length for the same mixing surface. By using two concentric toroid chambers as shown, the length was reduced by roughly one-half while maintaining the same mixing surface area.

Perhaps we should approach air augmentation from the other end; it might be that a very slight amount of air is best. Earlier the comment was made that the recommended air to rocket gas ratios were decreasing from approximately 10 to approximately 4. We would like to suggest that these ratios may be even less than 1; perhaps 0.40 or 0.50 might be most attractive from an overall standpoint.

Figure 26 depicts an aerodynamic spike configuration, with inlet provisions to bring aboard a relatively small amount of air as secondary flow for the aerodynamic spike. In this manner we would be utilizing the aerospike center body cavity, so to speak, as an air-augmenter mixing duct. Recall that the 96 percent efficiency shown for the aerospike was based on total flow.

While some controversy exists as to the net amount of payload increase possible with air augmentation, I believe most investigations are in agreement that it is not prudent to have an energy exchange as well as a momentum exchange between the ducted air and rocket exhaust products. It doesn't pay to leave the rocket exhaust products in the fuel-rich condition and wait for ducted air to be brought aboard for subsequent afterburning. For example, at, say, 20,000 to 30,000 feet of altitude at Mach 2 or 3, the ducted air has only a total pressure in the neighborhood of 30 psi. So if you just delay with some of the fuel-rich high-pressure rocket exhaust products and burn at that pressure, there isn't too much performance to gain.

We would like to suggest another approach, where the air would be taken onboard, liquefied, pumped to a relatively high rocket pressure, and burned in a conventional rocket engine. Figure 27 shows what we mean by conventional. Air enters the duct and is precooled and condensed in a heat-exchanger configuration similar to the Marquardt and Air Research design, with the hydrogen being the working fluid. The condensed air is collected in a sump and pumped to high pressure. In this case we have pumped the liquefied air to 500 psia. It is then burned with approximately 10 percent of the tanked hydrogen in the rocket augmenter at a mixture ratio of liquid air to hydrogen of 33 to 1. The remaining hydrogen is burned with the oxidizer in the main aerodynamic spike toroidal engine. In this case, 7.9 million pounds of thrust was delivered by the augmenter and 30 million pounds of thrust by the conventional rocket at sea-level conditions. Our calculations showed an increase of some 90 seconds with this system at sea level over a conventional O_2/H_2 rocket engine.

We have run trajectories with this configuration as a booster of an O_2/H_2 two-stage-to-orbit vehicle and found a 20- to 25-percent payload advantage over a conventional two-stage-to-orbit rocket vehicle. A 40-percent payload advantage was calculated for a single-stage-to-orbit vehicle. We find that if you take the air inlet and the heat exchanger and stage it much like the stage-and-a-half Atlas, then a payload advantage of some 84 percent is possible over a conventional single-stage-to-orbit vehicle.

As an indication of the weights we are talking about here, all of the paraphernalia required for the augmenter, liquefier, and pumps comes out to a thrust-to-weight ratio of approximately 35:1. This is low compared with that of a rocket engine but is quite high in comparison with conventional air-breathing engines. The rocket thrust-to-weight ratio was on the order of 100:1. We believe that this system is much simpler than other LACE systems that are being considered. In this system the air will be liquefied and pumped to rocket chamber pressure and immediately burned thereafter. Some of the air brought on-board that might not be liquefied could be discharged much like secondary flow as previously described in the discussion on the aerodynamic spike nozzle.

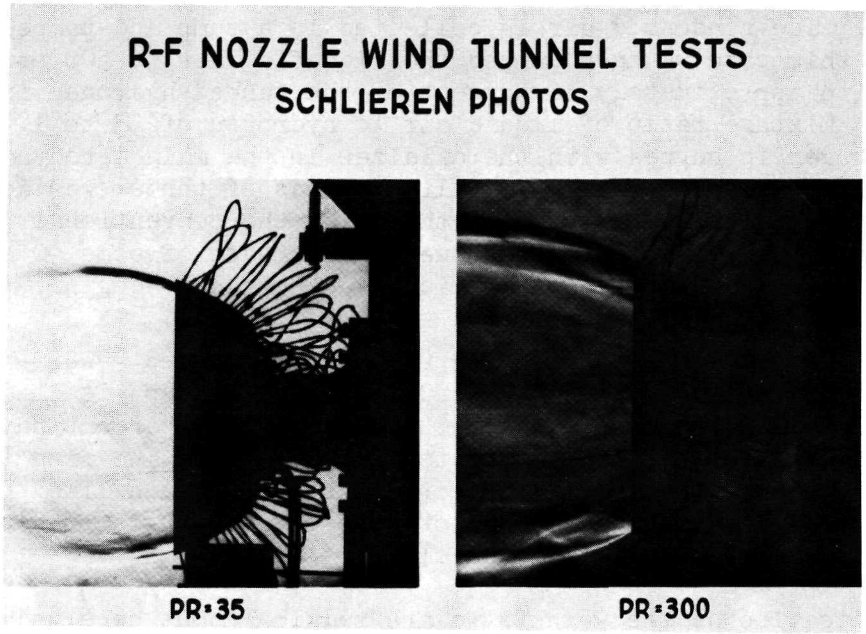


Figure 1

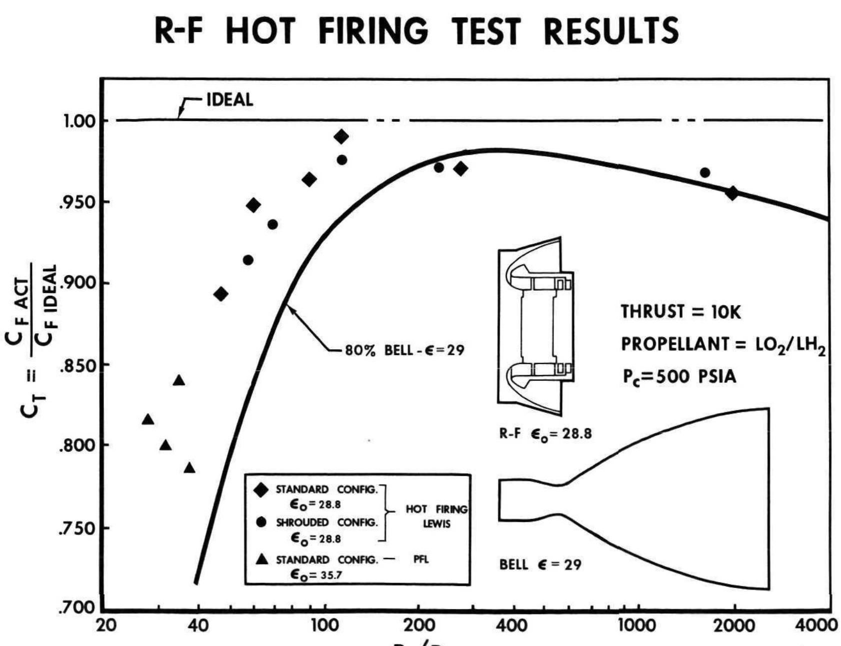


Figure 2

ANNULAR SPIKE MODEL

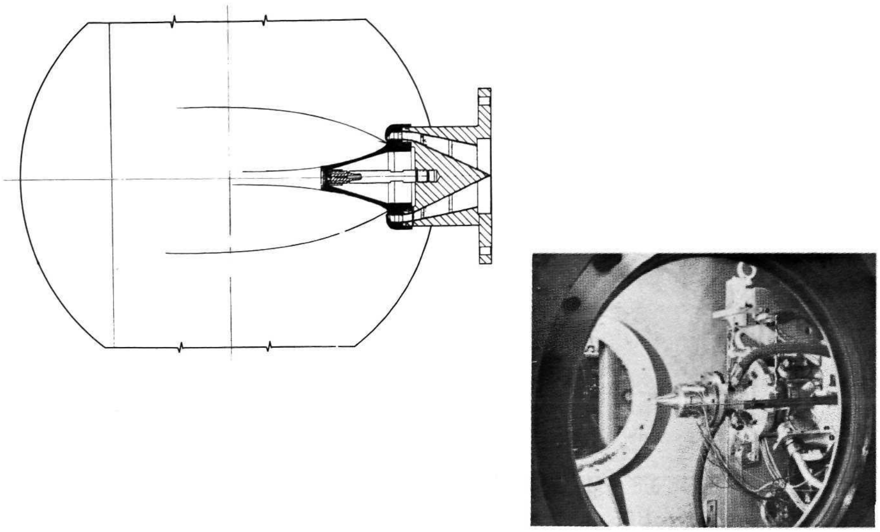


Figure 3

12 HOLE E-D MODEL

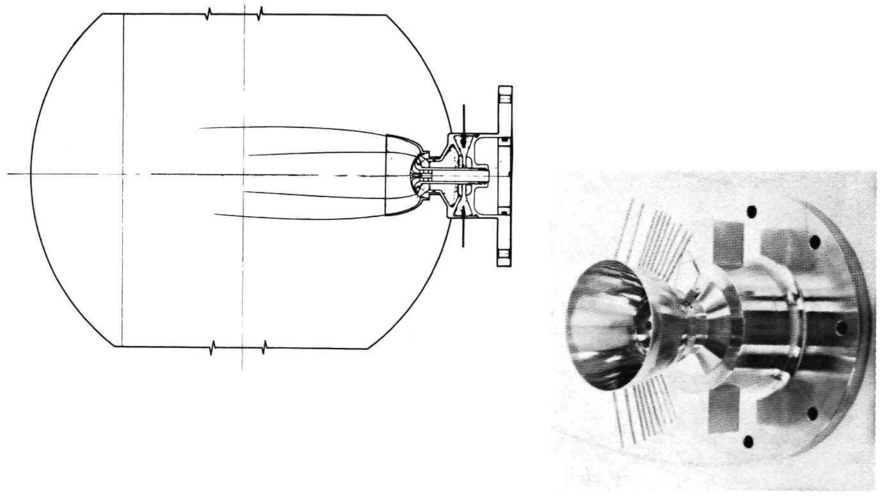


Figure 4

COMPARATIVE PERFORMANCE
RATIO OF MULTI-CHAMBER TO ANNULAR THRUST COEFFICIENT

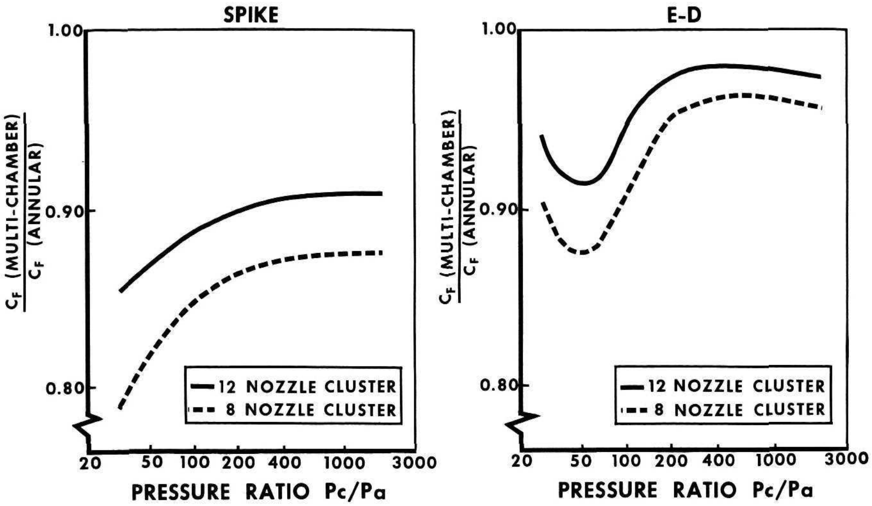


Figure 5

TOROIDAL CHAMBER

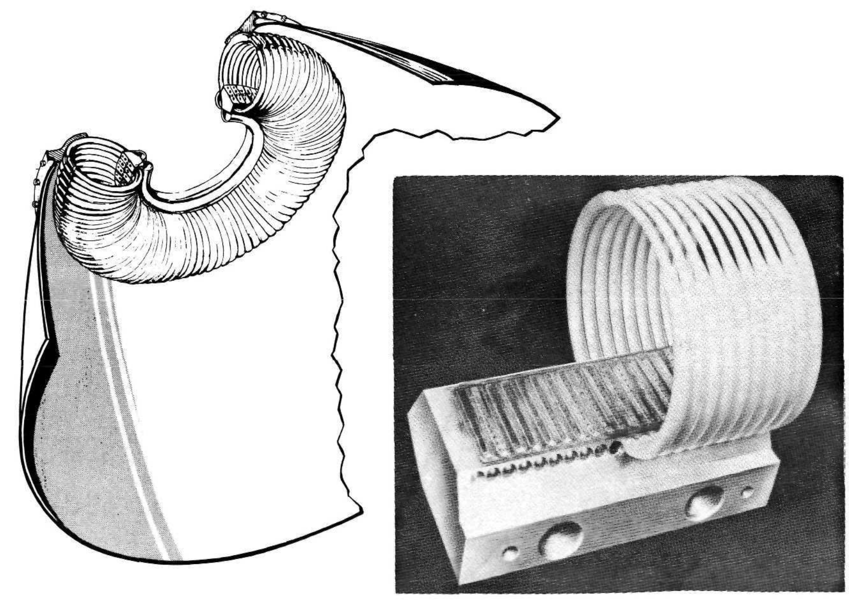


Figure 6

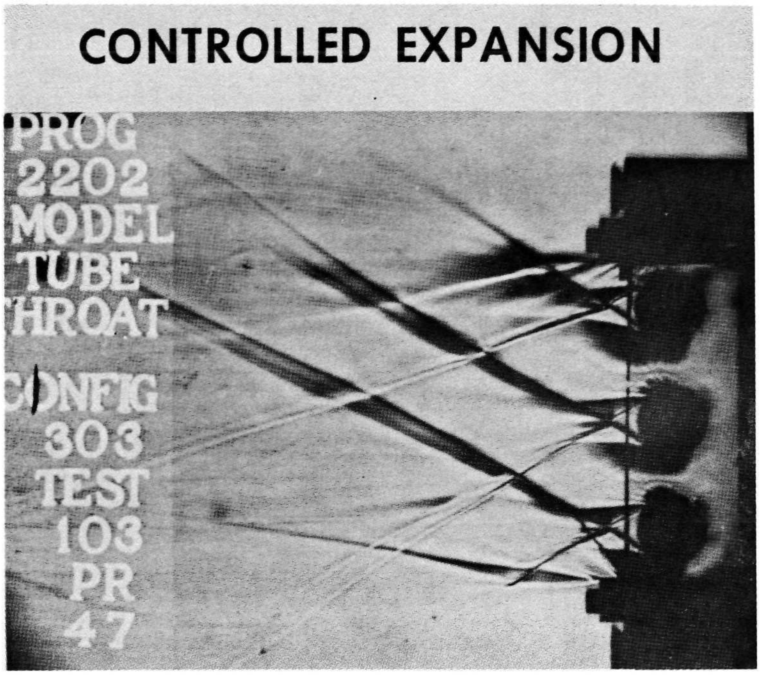


Figure 7

FLOW FIELD CALCULATED BY METHOD OF CHARACTERISTICS

(PRESSURE RATIO = 20)

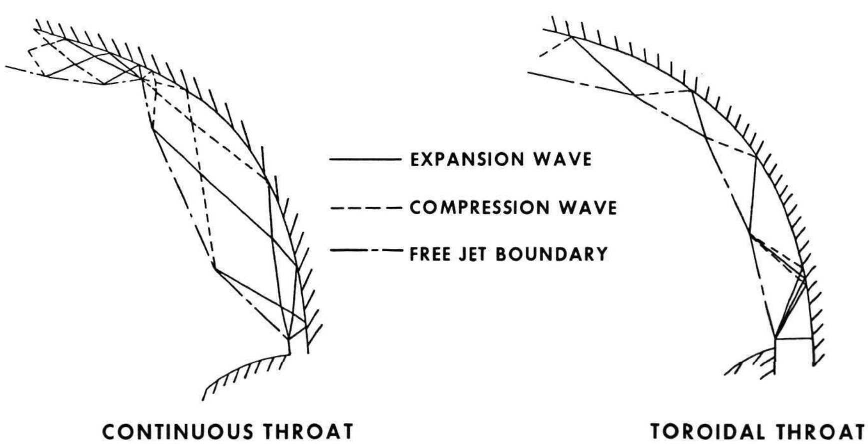


Figure 8

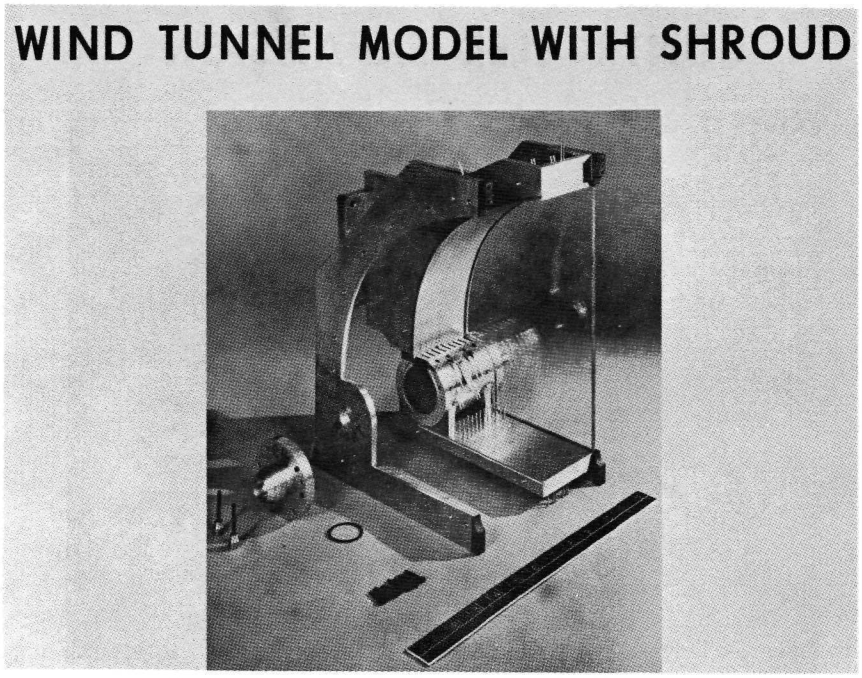


Figure 9

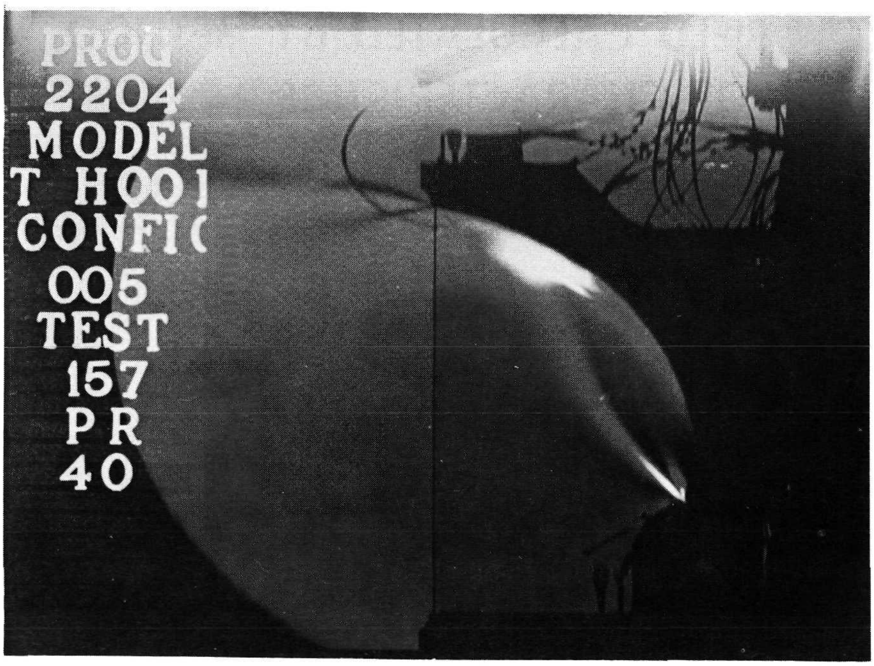


Figure 10

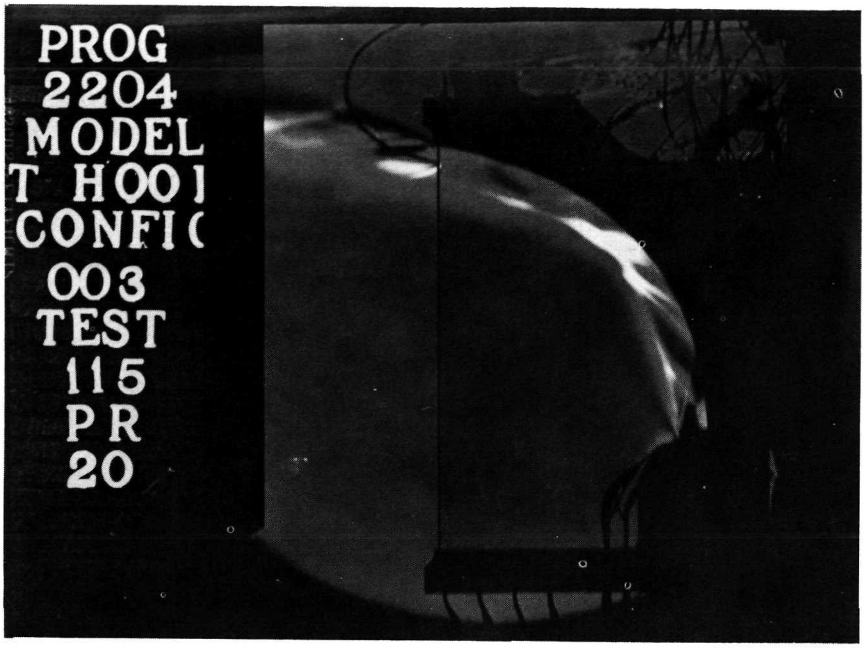


Figure 11

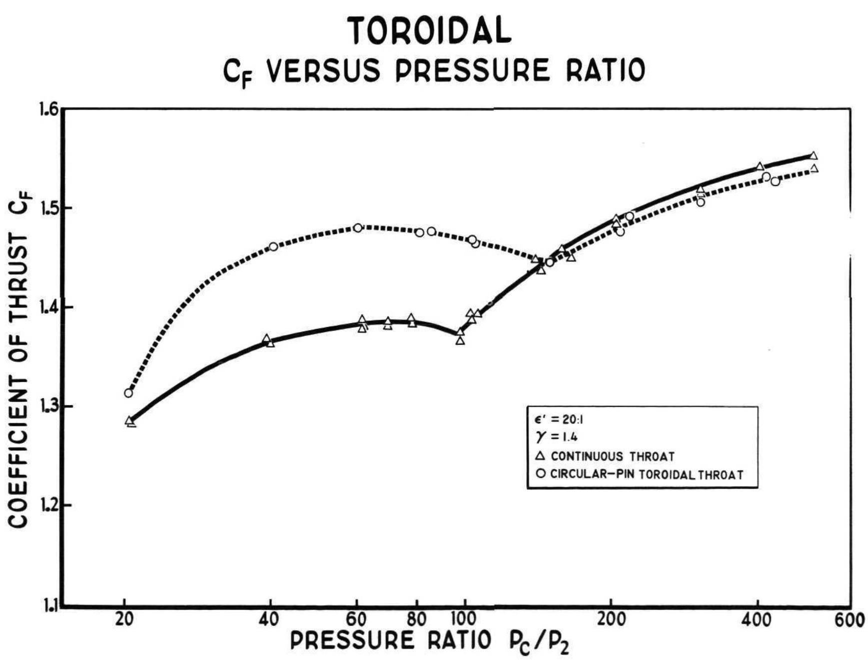


Figure 12

COMPARATIVE PERFORMANCE

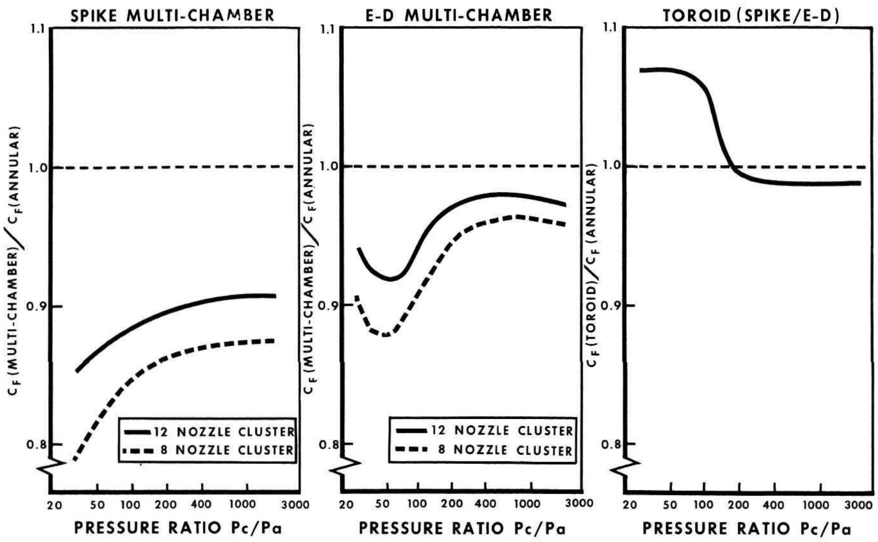
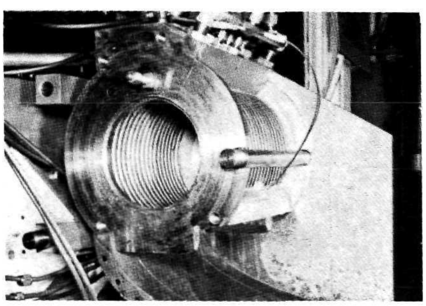


Figure 13

TOROIDAL HOT FIRING MODELS

STORABLE PROPELLANT MODEL



LOX/H_2 MODEL

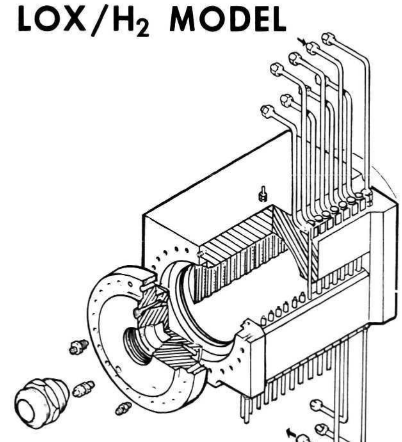
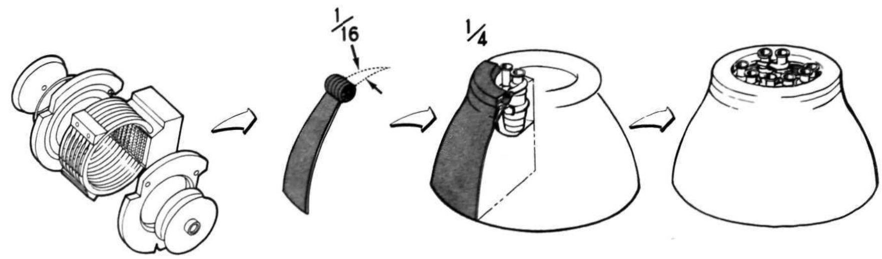


Figure 14

[redacted]

DEVELOPMENT SEQUENCE



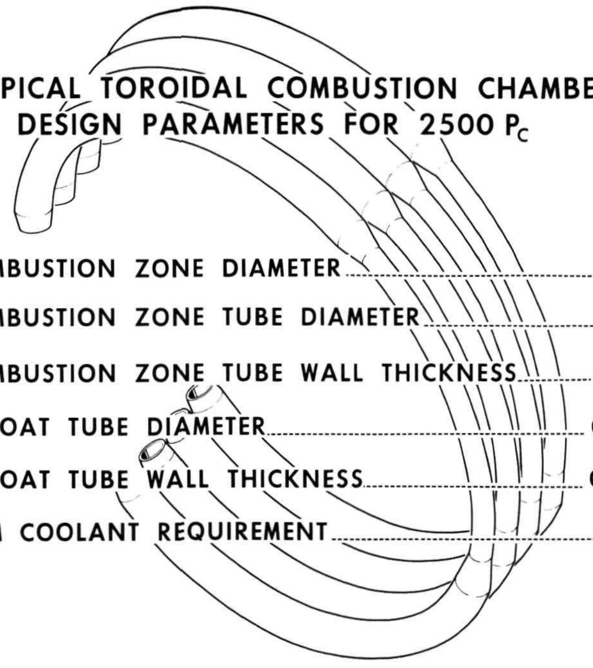
PHASE I - 100K TEST SECTION

PHASE II - 1.5M̄ THRUST CHAMBER SEGMENT

PHASE III - 6M̄ SEGMENT ENGINE

PHASE IV - 24M̄ ENGINE

Figure 15



TYPICAL TOROIDAL COMBUSTION CHAMBER
DESIGN PARAMETERS FOR 2500 P_C

- COMBUSTION ZONE DIAMETER .... 15″
- COMBUSTION ZONE TUBE DIAMETER .... 1.25″
- COMBUSTION ZONE TUBE WALL THICKNESS .... 0.105″
- THROAT TUBE DIAMETER .... 0.774″
- THROAT TUBE WALL THICKNESS .... 0.208″
- FILM COOLANT REQUIREMENT 1%

Figure 16

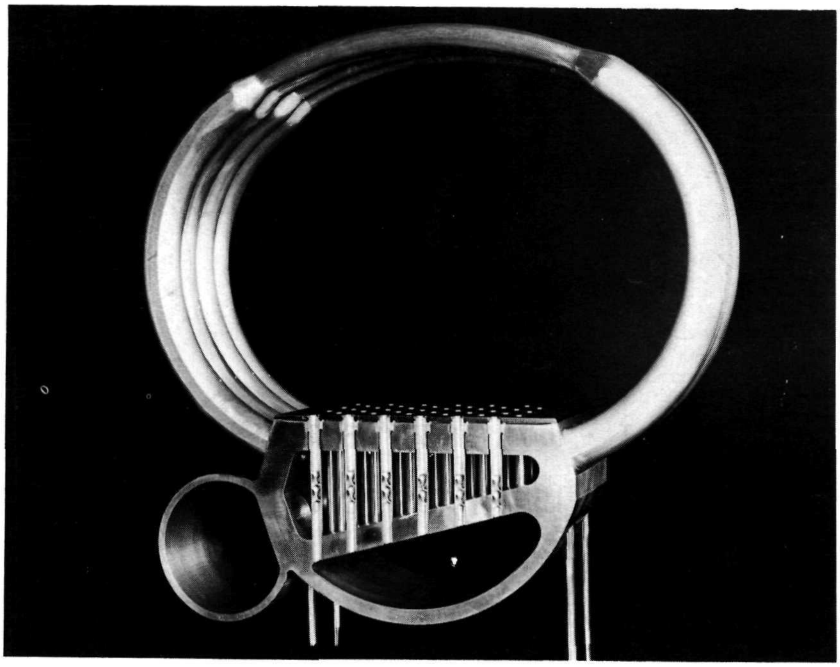


Figure 17.- Mockup of Phase I toroidal segment.

AERODYNAMIC NOZZLE CONCEPTS

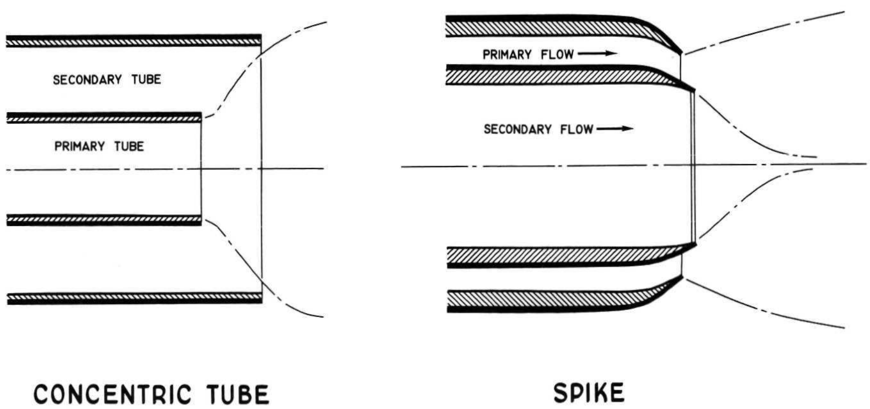


Figure 18

AERODYNAMIC SPIKE NOZZLE

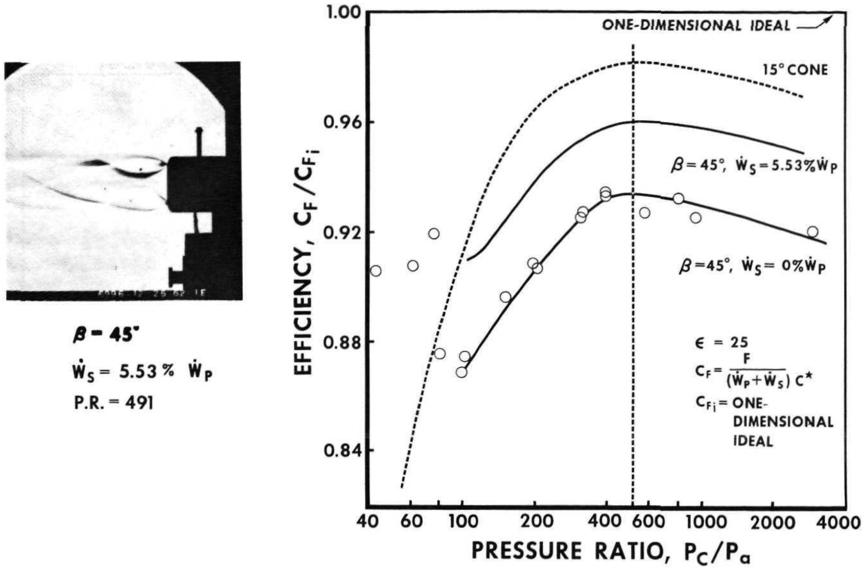


Figure 19

DUAL ε TOROID

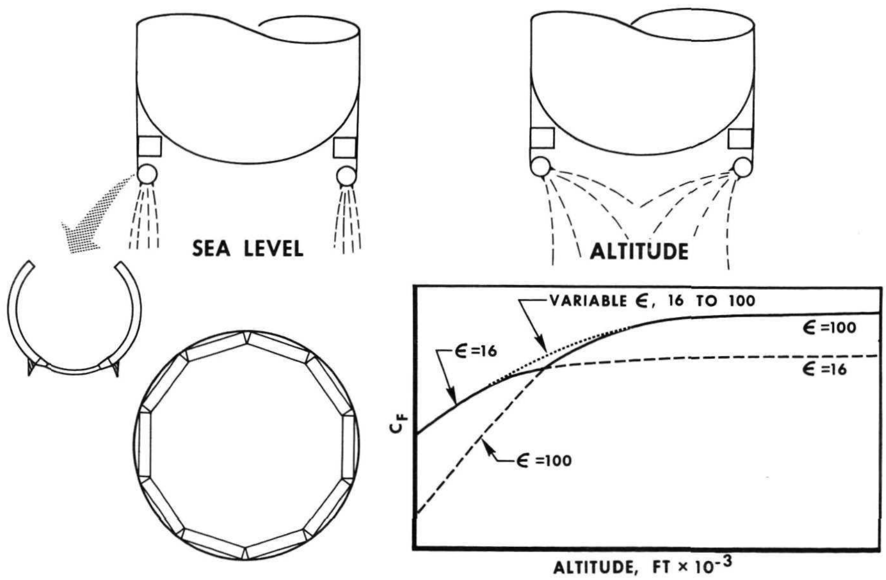


Figure 20

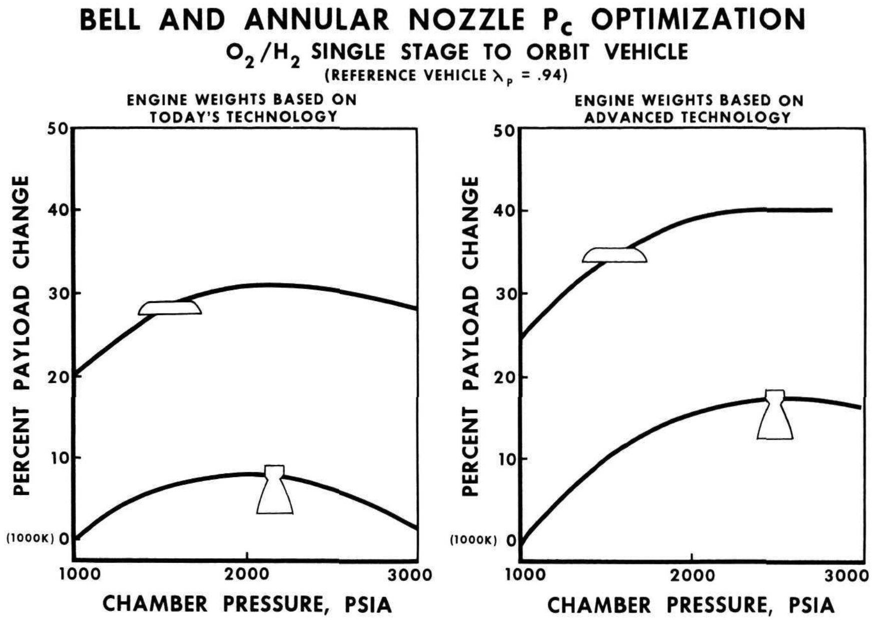


Figure 21

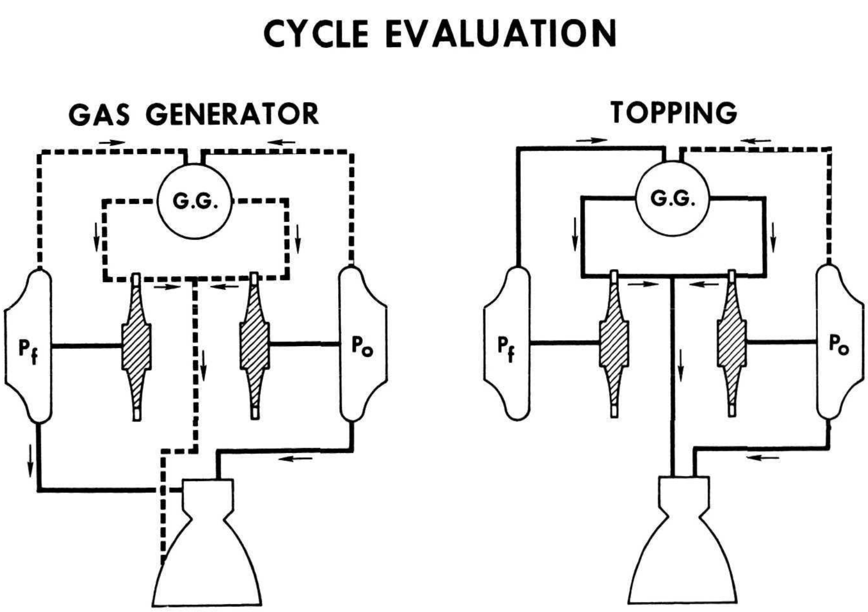


Figure 22

[redacted]

PUMP DISCHARGE VS CHAMBER PRESSURE

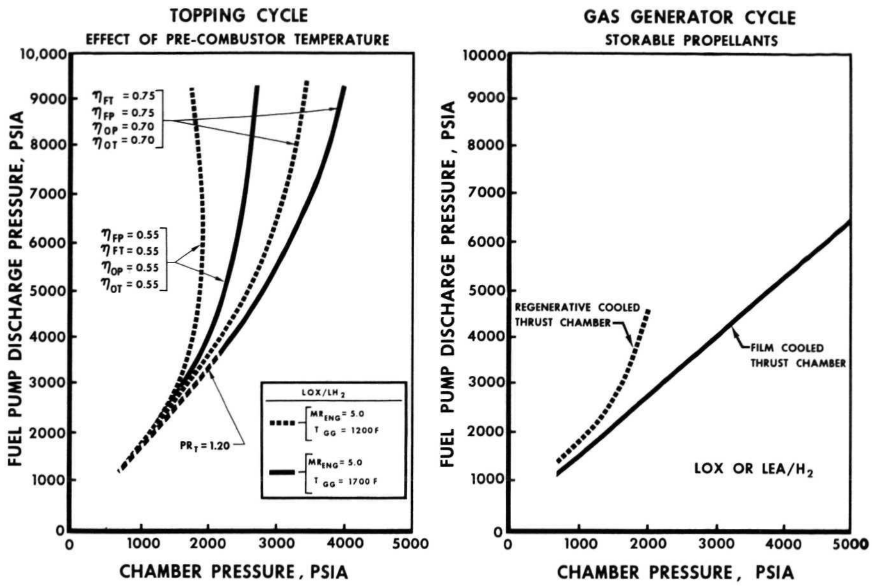


Figure 23

THRUST AUGMENTATION DUE TO IDEAL MASS INCREASE

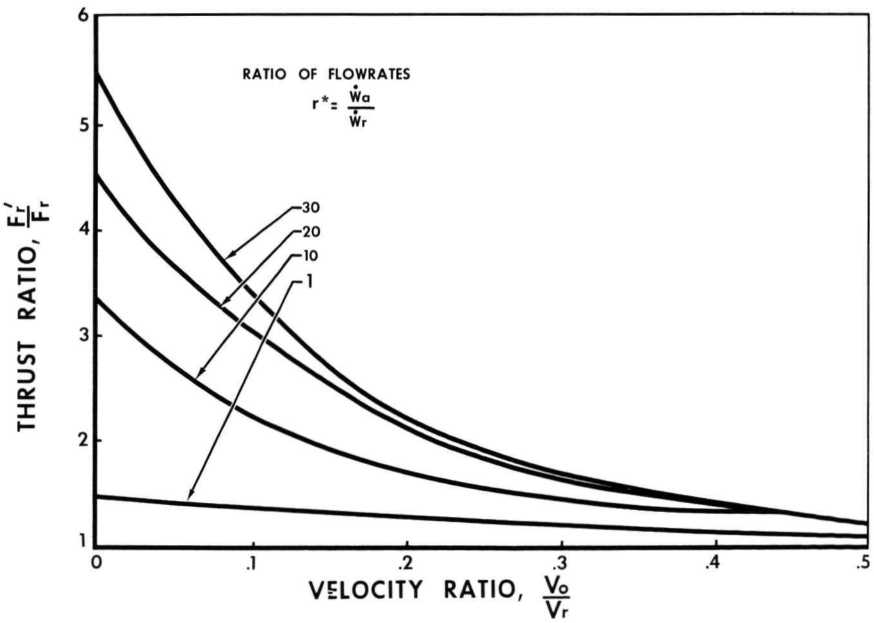


Figure 24

[redacted]

MIXING DUCT COMPARISON

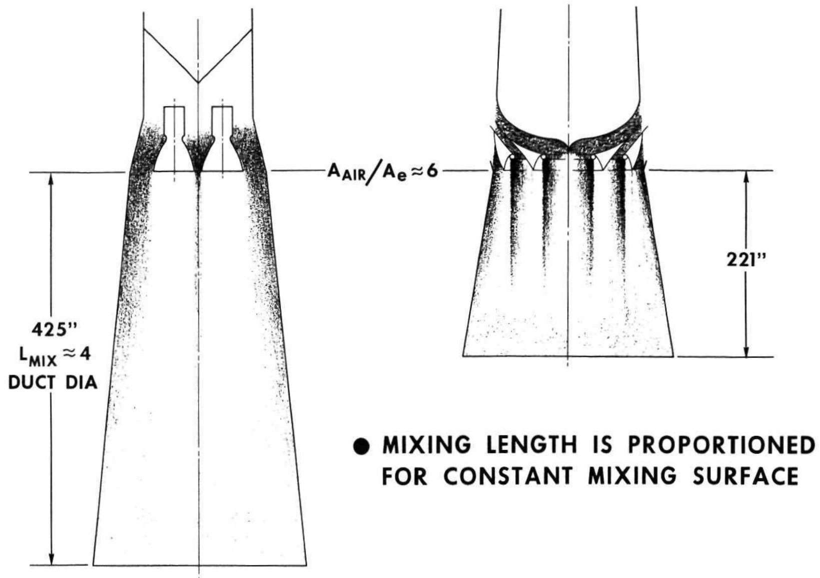


Figure 25

AERODYNAMIC SPIKE NOZZLE

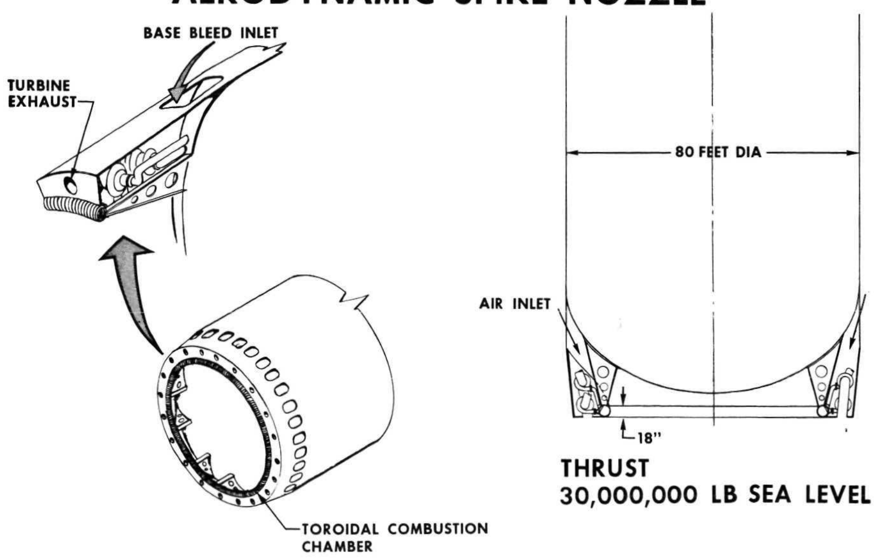


Figure 26

ADVANCED
AIR AUGMENTED ROCKET SYSTEM

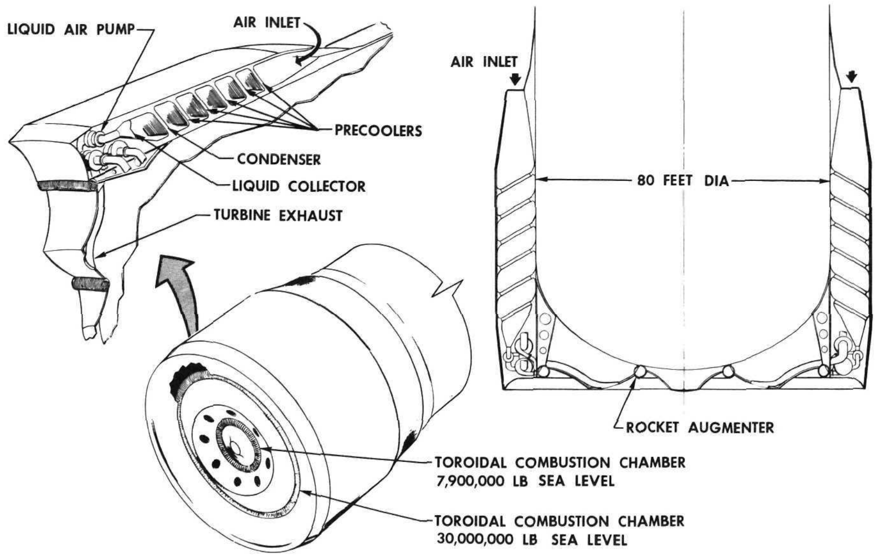


Figure 27

[redacted]

Part II

By Douglas Hege

Rocketdyne Division
North American Aviation, Incorporated

The preceding discussion covered many of the ideas that we think are probably going to be incorporated in the next generation of engines. I would like to put some of these ideas together. We do not arrive at a specific engine. At the moment it comes out a spectrum of engines: the ED, the spike, the aerospike, air augmentation, and advanced air augmentation. (See fig. 1.)

The "ideal" engine, if we had to build it now, we think would have the toroid combustion chamber. The more we work with this, the easier it gets. When every step in a project becomes more and more difficult, it is, I think, a good indication that you are on the wrong track. The farther we go with the toroid, the easier it is. We can take a single tube and run under actual heat transfer and cooling conditions. We can afford to burn out a few tubes at a cost of a few thousand dollars during film cooling development, which is something that cannot be done in a big engine for millions of dollars. Thus, we think it would have the toroid combustion chamber. Whether it has an ED or spike nozzle is immaterial to us.

The airframe people are tending to write off a certain amount of the nozzle weight as part of the tank bottom. This will have an effect on the chamber pressure.

There is one point that disturbed me a little bit in the previous discussions. Here we take the base pressure problem from the vehicle people and are being criticized for not having it equal to ambient. With bell nozzles the vehicle people have to worry about the base pressure. In an aerodynamic nozzle the propulsion people are accounting for it. We have great hopes we will get it equal to the ambient pressure. But this should be taken into account in the overall vehicle optimization.

If things go as well as we hope, the aerodynamic spike will be the answer. That is, you can eliminate the metal nozzle and the air flow will create a pressure field very similar to that for the metal contour.

Air augmentation, I think, is something we want to consider. It is going to take some time to determine whether it is the thing we are after.

There has been a lot of pressure put on us, and I am sure on others in the business, to make some significant changes. Let's just not work for that 1 or 2 percent. If this vehicle is going to be single stage to orbit and if it is going to be recoverable, there is going to be a requirement for significant improvement. Some 25 and 50 percent performance improvements should be attempted. If these improvements are real, then some new approaches could be tried. The single stage under today's technology, I think everybody realizes,

[redacted]

is marginal. This is the reason we look at things like the advanced air augmentation.

More or less specific versions of the advanced engine would probably be something like those shown in figure 2 - ED or spike, or aerodynamic spike, we hope. It will have a toroid combustor, an advanced nozzle. It will have provisions for air augmentation until proven good or bad. It will have the capability of segmentation testing. We think that you can design, develop, and get the reliability you want in a segment. We need some way of cutting down on the development cost, some way of getting a higher reliability without having to test the whole system the brute-force way.

We are looking at a thrust range of 20 million to 30 million pounds. I think the thrust level depends on the Nova vehicle configuration. The decision is not up to us. We will provide an engine whose thrust equals that required. With regard to chamber pressure, the optimum lies between 2,000 and 3,000 psia, as discussed in the preceding pages. The aerodynamic nozzle may optimize out even higher. The optimum chamber pressure also depends on how much you write off against the vehicle and nozzle. Area ratio and mixture ratio pretty well optimize out with chamber pressure. The specific impulse figures listed cover the range of chamber pressure, area ratio, and mixture ratio considered.

In the development program, we have assumed a segment of something on the order of 6 million pounds of thrust. (See fig. 3.) We say 5 years to PFRT; 7 to Qualification. This is our estimate at the moment of what the development program would be.

We think that the cost is affected slightly by the chamber pressure: 2,000P_c and 3,000P_c. Figure 4 shows the cost-thrust curves for lox/hydrogen. We have the same set of curves for lox/RP. A 6-million-pound engine will cost something like $350 million through PFRT.

With regard to reliability, we have run a series of studies in relation to number of segments. (See fig. 5.) From this you can get a whole lot of answers. We found that choice of segment size should not be based on some parameters that are very highly sensitive. If the basic segment reliability is assumed to be 0.99, the overall engine reliability curve is fairly flat in the low number-of-segment range. It drops off toward the high number of segments.

In relation to the toroid, which has only one combustion chamber, we think that we will have some high impedance fences or interconnects so the toroid will not be completely open, so that you may throttle it or shut down a section. But it will run and operate like a single combustion chamber. Thus, the main factor is really the pumps. How many pumps? We believe that it is desirable to have the biggest pump that can be easily built; we think that today this is a segment of around 6 million pounds of thrust.

We are looking at the facility requirements. Figure 6 emphasizes the fact that we think we will test the engine just as a segment. You can run it either vertically up or horizontally out. We think that the engine manufacturer can do his job with a 20 or 25 percent segment.

Experience has taught us that you are not going to get any of these engines or any new ideas unless you do go through a predevelopment program and that you will not actually incorporate advanced features into a propulsion system unless you have seen it run in large hardware. Almost everything that is in an engine today had previously been run experimentally at a significant size. To us, a minimum size is 100,000 to 400,000 pounds of thrust.

We estimate that if you assume calendar year 1967 as a go-ahead date, you have approximately 3.5 years to conduct technology work. (See fig. 3.)

Table I shows the time schedule for the predevelopment program. During that time period we must make these decisions: In the thrust chamber, what combustor? Cylindrical, multiengine, or toroidal? What is the nozzle; ED, spike, or aerospike?

There are certain key dates by which these decisions will have to be made to be able to run an experimental engine program. I believe that what isn't run in an experimental engine will not go into the engine development program. We have seen it in the J-2 and F-1. The man who has the responsibility for doing a job wants his propulsion system based on fact. He wants to see something. He doesn't want to base the system on a paper design.

We are in the process of running what we call a 3-D PERT chart. Included are thrust chambers, air augmentation, combustion, and nozzles, only the things we think need to be done. We are plotting the critical paths and the key decision dates as to when we must decide which feature we will put in an experimental engine.

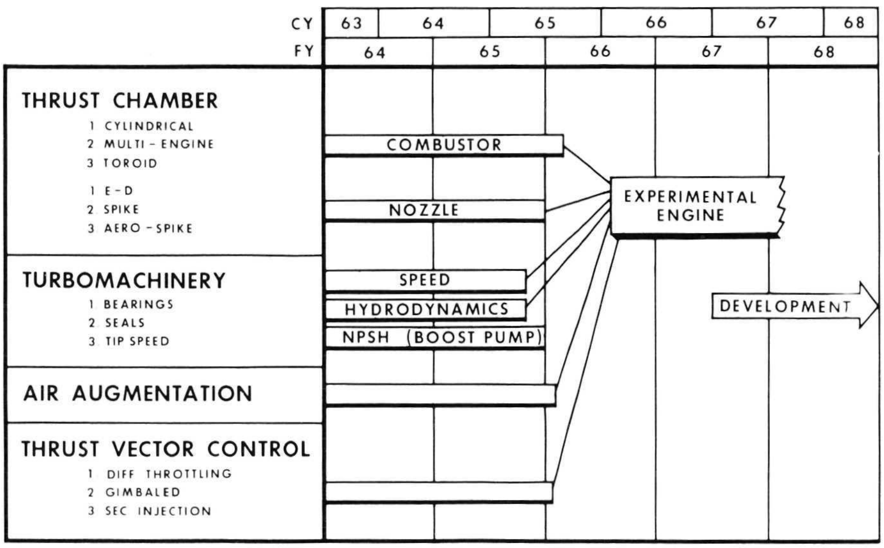
Figure 7 indicates our concept of what the thrust chamber of such an experimental engine could be like. We think that the minimum size experimental engine should be 400K; at this thrust level meaningful data could also be obtained on an experimental turbopump.

Figure 8 shows a typical example of what you can expect. Based on today's technology, this is what a 6-million-pound turbopump will look like. Note the J-2 engine turbopump. If we can increase the speed through hydrodynamic work, bearing and seal work, we can arrive at the advanced engine turbopump shown. And this is typical of the components, the advanced features, that will allow reduction in weight and size.

This then ends up with a 400K experimental engine (fig. 9) which we think should be used to test the features and count on them in a 30-million-pound engine.

TABLE I

ADVANCED TECHNOLOGY AND PRE-DEVELOPMENT PROGRAM



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ADVANCED NOVA ENGINE SPECTRUM

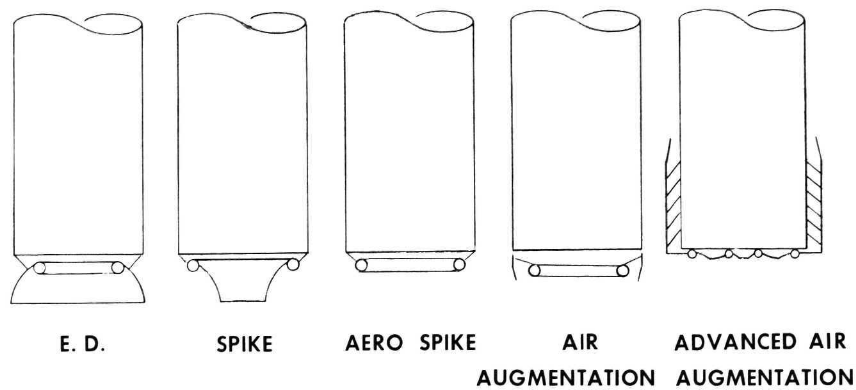
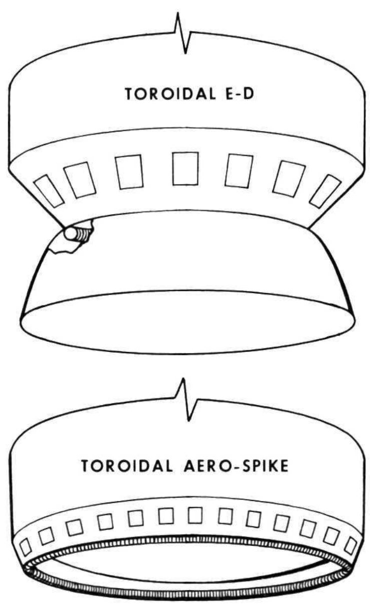


Figure 1

NOVA ENGINES



FEATURES

- TOROIDAL COMBUSTOR
- ADVANCED NOZZLE
- PROVISIONS FOR AIR AUGMENTATION
- CAPABLE OF SEGMENTATION TESTING
- PERFORMANCE

  THRUST = 20-30 MILLION LBS

  CHAMBER PRESSURE = 2000 TO 3000 PSIA

  AREA RATIO ~ 70-100:1

  MIXTURE RATIO 5:1-7:1

  SPECIFIC IMPULSE

  S. L. = 370-400 SEC

  VAC = 440-460 SEC

ASSUMPTIONS

SHIFTING EQUILIBRIUM

η_{c^*} - .96-.98

GAS GENERATOR CYCLE

BLEED GAS FROM TURBINE

AMBIENT E-D BASE PRESSURE

Figure 2

[redacted]

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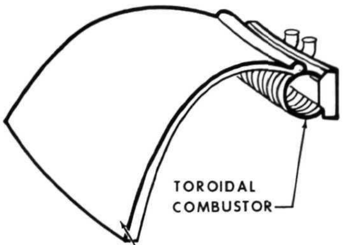
TOROIDAL SEGMENT TESTING

THRUST__________________100K TO 400K

CHAMBER PRESSURE ________2000 - 3000 psia

PROPELLANTS _____________LOX/H_2

TEST TOROIDAL COMBUSTOR SEGMENT OF FULL SCALE ENGINE

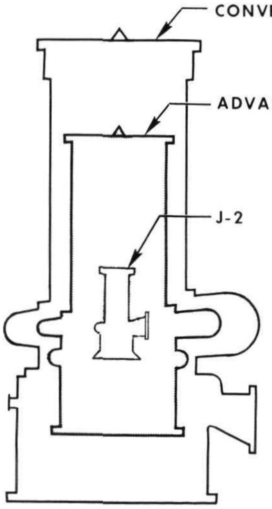


OBJECTIVES

- VERIFY SEGMENT PERFORMANCE
- ESTABLISH OPERATING CHARACTERISTICS
- EVALUATE COMBUSTION STABILITY
- ESTABLISH NOZZLE DESIGN CRITERIA
- STUDY MATERIALS & CONSTRUCTION

Figure 7

LH_2 TURBOPUMP COMPARISONS



	CONVENTIONAL	ADVANCED
STAGES	11	7
SPEED	5,400	10,500
WEIGHT	+28,000	17,300
DN VALUES	1.2×10^6	2.7×10^6 MAX.

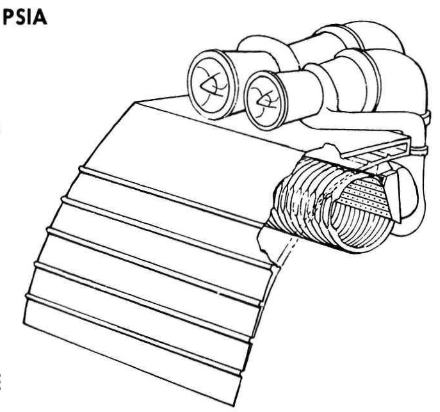
Figure 8

EXPERIMENTAL ENGINE

THRUST ______________ 400K
CHAMBER PRESSURE ____ 2000-3000 PSIA
PROPELLANTS _________ LOX/H_2

COMPONENTS

FULL SCALE TOROIDAL COMBUSTOR
EXPERIMENTAL PUMPS



OBJECTIVES:

- ESTABLISH OPERATING CHARACTERISTICS
  - START AND CUTOFF
  - SEQUENCING
  - VIBRATION
- EVALUATE TAP-OFF TURBINE DRIVE
- VERIFY COMBUSTION STABILITY
- EVALUATE CONTROL REQUIREMENTS
- ESTABLISH SERVICING REQUIREMENTS

Figure 9

11. SUMMARY DISCUSSION

MR. THOMPSON: How long will it be before you fire a toroidal engine at a chamber pressure of 2,500 to 3,000 psia?

MR. IACOBELLIS: Our program started in May 1963. It is a 9-month program. That would put it early in 1964.

MR. THOMPSON: What thrust level?

MR. IACOBELLIS: We will start out with one tube, which will deliver approximately 20,000 pounds of thrust. We are going to use Rigimesh side plates and do some film cooling work with the one tube. We hope to perfect the film cooling with this one-tube model at 20K. Then we will test a four-tube model and that will be at 80K, 2,500 psia. Our facility will sustain a P_c of only 2,100 to 2,200 psia for long duration tests. But for short duration tests, 4 to 5 seconds, we hope to bring the chamber pressure up to 2,500 psia. The model is designed for 2,500 psia.

MR. THOMPSON: What type of liquefier were you considering in your advanced air augmented rocket?

MR. IACOBELLIS: It was the Marquardt heat exchanger. Depending on the total temperature, the construction material varies from titanium to stainless steel to aluminum and is approximately 2 to 3 mils thick. We have been working with Marquardt on their LACES system for the last 2 years. It appears that a scheme much like the one they tested in 1962 would work here.

MR. CHANDLER, Marshall: Do you have a figure handy in your cycle analysis on the specific impulse difference? What is the ΔI_{SP} between the topping cycle or the preburner cycle and gas-generator cycle?

MR. IACOBELLIS: First, if you look at it for a bell gas generator and a bell topping, it gets to where it is 7 to 9 seconds lower in the gas-generator cycle. If you look at it for the case of an annular nozzle, where we might recoup more performance from the gas-generator flow, much like we showed in the aerodynamic spike, we think that the difference would be less. However, if you go along with our constant pump discharge pressure viewpoint, where you might operate a gas-generator system at 3,700 psi for the same pump discharge pressure, as opposed to 2,500 psi for a topping cycle, it is quite conceivable that you can have a higher specific impulse with the gas-generator cycle.

MR. CHANDLER: What was the mixture ratio on the P_c optimization?

MR. IACOBELLIS: We did it for both 5 to 1 and 7 to 1. I might add that for a mixture ratio of 5 to 1, we assume 100-percent frozen equilibrium efficiency, which is about 0.98 of shifting. With a 7 to 1 mixture ratio, we felt that you probably should assume a lower percentage of shifting because now the spread between frozen and shifting is greater. I don't think we are at all sure whether the process follows shifting or frozen equilibrium. So at the 7 to 1 ratio we lowered our 98 percent down to 96 percent of shifting.

I mention all of that because of our different assumptions and also that it might add caution in going to mixture ratios of 7:1 or 8:1. For example, at 2,000 psi with an 100:1 area ratio, there is a difference between 100-percent frozen and 100-percent shifting equilibrium for a mixture ratio of 7:1 such that you might be off some 30 seconds if the process was frozen rather than shifting.

I think all of us are of the opinion that lox/hydrogen and fluorine/hydrogen will shift more than lox/RP, but we are not sure to what degree.

MR. VERN JARAMILLO, Hq-MLP: What factors led you to select the elliptical shape for the toroidal tube?

MR. IACOBELLIS: The surface area isn't equal all around the tube for the forces to act on because of the throat region. The oval shape is, what we might say, the minimum energy shape. In other words, there are not any bending moments in a tube of this shape when the chamber is pressurized. If it was a circular hoop, then you would have some bending moments.

MR. ESCHER, Marshall: Could you give us a remark or two on your approach toward side force generations?

MR. IACOBELLIS: Yes. First, I would like to recall some earlier remarks, that with the larger vehicles and with the hydrogen on the bottom, you might get by with two or three degrees of gimbaling. This, of course, makes side gas injection more attractive. It also makes differential throttling quite attractive, we believe. And so we would want to do some differential throttling work with the toroid in an advanced technology program. We think we could achieve the required gimbaling with a small amount of differential throttling.

MR. HEGE: I might also point out that in the advanced air-augmentation aerodynamic spike configuration we intend to gimbal the linear toroidal segments so they can also be used for gimbaling. If you rotate a segment about its center, a section say 6 feet long wouldn't have much moment of inertia. So it would make gimbaling weight satisfactory for thrust vector control.

MR. MAHAN, Manned Space Flight Center: You indicated on your air liquefaction setup that you obtained better performance than without it. I wonder if you took into account the added weight, complexity, and reliability of the air liquefaction equipment.

MR. IACOBELLIS: We took weight into account, but we did not consider the complexity and reliability factors. Of course, there is a trade-off there. At this point it appears that we can get these gains with fixed inlets and fixed augmenter throat area, which simplifies the system but somewhat complicates our studies because we are taking varied amounts of air on, and, of course, the total pressure varies. We have flown trajectories such that our total pressure is not lower than 10 psi to prevent freezing, nor greater than 20 psi to avoid taking in too much air; otherwise the P_c of the fixed throat augmenter will exceed the design value.

MR. MAHAN: That is, in effect, operation at one point only, correct?

MR. IACOBELLIS: That was a one-point design for the inlet. But, no, we flew and integrated over the trajectory. We went through the whole bit, the total pressure versus Mach number, and actually flew a trajectory within this corridor. In fact, maybe to add emphasis to how detailed it was, it was estimated that it took us one man-month to run this trajectory, and we have what we think to be a well-developed 7090 trajectory program. So there were a lot of iterations required.

MR. SLOOP: In testing or using this configuration, aren't you going to have dividers more often than, say, one-fourth segments to get away from instability? And if you do, wouldn't you have something that looked perhaps like a frankfurter or a bent shape with an elliptical end? And wouldn't you want to test a complete segment like this?

MR. IACOBELLIS: Yes. First we are going to do work with just one tube, with plates on each end. Then, we plan to move on to four tubes. This is our NASA program as it is now envisioned, 80K without the shroud. Then we would like to go to Phase II of the development program at something like 400K. In this phase we would learn about the coupling effects. I believe when we get out to 20 tubes, or a number in that region, we will have it pretty well nailed down.

We are going to get quite a bit of knowledge of the stability in the present program, certainly these two modes. As you go to greater numbers of tubes you may have a race-track effect. I might make a qualitative statement here that in the turbojet engines I believe they have preferred this tangential flow. I understand that G.E. went from a multicombustor configuration to an annular combustor. I realize the heat problem is much less severe with the lower pressures that we are dealing with there, but it is a probability it will not hurt us. We believe we will learn quite a bit about the tube coupling effects when we test at a thrust level of approximately 400K. We may or may not need baffles. You might think of partial baffles in a 400K sector, and certainly the larger sector would be delivered with end plates that would be fixed. We don't envision a full 360° continuous toroidal combustor.

MR. SLOOP: Mr. Hege, your $3\frac{1}{2}$ years of technology intrigued me. It looked as though everything had to start at the same time. I wonder, can you sort out in relative orders of importance on your PERT diagram what is the most critical item. Is it the pump, the combustor, or the combustor with the nozzle, or have you done it this way?

MR. HEGE: At least from where we are today - and we are in the process of trying to nail this down - I think that the pump and the combustor are running neck and neck. These seem to be the two big items. If you are going to build the big pump, I think you should do some work now. In fact, we have some in-house work and we have some work with NASA on some of the things that are required to be done in the pump area. Some problem areas are becoming quite obvious. Seals and bearings are very limiting factors on a big turbopump. I am not sure hydrodynamics are as critical. In the combustor, I think we are going along with the toroid if it continues to run as well as it has. Those are the two critical items at the moment.

Maybe I an not smart enough to foresee which one will be the critical item $3\frac{1}{2}$ years from now.

MR. SUDDRETH: Have you taken a look at this particular system at a reduced pressure operation, say 10 or 25 percent pressure level, for stability?

MR. IACOBELLIS: Throttling? We have run just the two toroidal programs that I mentioned. We ran one model at 450 psia and then ran it as low as approximately 300 psia. So we varied the thrust that much. But that was not intentional. It was done while calibrating the mixture ratio, but it ran stably in all cases. I don't think there should be any difference here any more than with conventional bell injectors as far as being able to throttle over the range that we do now with conventional engines. We have throttled all of our engines down to 2 to 1 or 3 to 1.

SESSION III

ENGINE COOLING

Chairman: Donald Bartz,
Jet Propulsion Laboratory

12. INTRODUCTION AND OBSERVATIONS

By Donald Bartz

Jet Propulsion Laboratory

At the end of this session we will excuse the contractors and permit discussion of the items that might be proprietary that deal with the heat-transfer problems.

My remarks will, hopefully, set the stage with regard to the problems associated with cooling of high chamber pressure engines.

Briefly, the heat flux problem is generated in large high-pressure engines because the heat transfer increases as a power of the chamber pressure very close to 1, somewhere between 0.8 and the first power. Thus, there is the problem of protecting the walls against an environment in which the heat transfer is increasing very rapidly, increasing to levels of perhaps 50 to 75 Btu per square inch per second. This represents a factor on the order of five greater than most conventional engines have run. As has already been indicated, the problem gets significantly worse above the conventional range of 1,000 psi, and very difficult above 1,500 psi.

It is clear to me that the cooling problem will dictate the thrust chamber design to a very large extent and that in many of the designs presented earlier the thrust chamber configuration was essentially determined by the problem of cooling.

There are several methods for protecting walls against the environment of a rocket combustion. There is regenerative cooling, transpiration cooling, film cooling, ablation, and radiative cooling. Three of these, transpiration and film cooling and ablation, I would like to call mass transfer cooling to differentiate from regenerative cooling. Each of these methods has its own particular limitation, either a practical or theoretical limit. For those for which we have data, there are practical limits, limits set by one or another physical process.

In the case of other cooling method we have no cooling data. There is, therefore, a theoretical limit. We will try to emphasize the things we don't know as well as what we do know as we go along. For only one of these methods of cooling is there really any data for practical limit and that is regenerative cooling. There is not just one limit but actually two. One is the pressure drop that you have to invest to provide enough velocity to get the regenerative cooling up, and the other is the thermal stress upon the metal walls that you are using to protect the metal walls that constitute the thrust chamber. For this, because it is easy to make tests in heated tubes, there are data, and one can establish the limit of regenerative cooling fairly precisely. I would say we know it to the order of 10 percent.

For radiative cooling, there is a fairly certain theoretical limit. We are able to calculate the radiationability of a surface fairly well and we therefore can predict theoretically with fairly good certainty what the theoretical limit of radiative cooling is. Unfortunately, it is so low it really isn't of much significance to these problems.

For the three mass transfer cooling methods, there are no data at high pressure. The data that we have are all at very low pressures compared to the conditions of interest. For that reason one has to rely on theory. The theory is very shaky in all three of them. Of the three the least uncertainty probably is transpiration cooling because it is a more uniform process than are the other two as you attempt to introduce mass transfer in a fairly uniform manner. But the desire to do this creates its big drawback and that is that it puts very stringent material requirements on the design, so stringent that I personally am very pessimistic about having material that will satisfy the requirements.

With film cooling there is no comparable materials problem, but there is a very great problem in predicting its performance from a theoretical standpoint because we have processes here which are very difficult to determine and predict, even to create a good model. Therefore, the determination of the success or applicability of the application of evaporation or film cooling must await the procurement of data. For this reason I would place a very high priority on advanced technology programs in that direction. There is at least one program of that type underway at present; some preliminary results will be presented from that program, which is funded by the Air Force, by Pratt & Whitney.

Finally, ablation probably has the worst combination of situations. It is difficult to make predictions, again for similar reasons. It has a very difficult materials requirement. The theory is uncertain and materials demands are very great. At present it is really not possible to predict with any reliability even the mechanism by which the ablation occurs. There is a wide range of opinions from steady state ablation to charring ablation. These methods have very large differences in the rate at which the wall disappears. One can just hypothesize; until an engine is run at these conditions, I don't think that we can, with any confidence, predict what will happen.

There are other influences on the selection of the cooling that are affected by such things as the propellant, the nozzle configuration, and the question of whether the thrust chamber is to be reused. The propellant selection (by propellant selection I mean only the two that are of interest from the discussions presented - lox/RP or lox/hydrogen) surprisingly has had little effect for regenerative cooling but may have a very large effect in the mass transfer methods. This will remain to be seen as we learn more about the mass transfer cooling methods.

The nozzle configuration also has a surprisingly small effect on the cooling requirements, much less than I would have expected previously. In one case, the conventional type of nozzle, there is more total area but less at high heat flux, and in the more unconventional nozzles, there is less total area but more at high heat flux, and they tend to balance one another. It is not obvious with which we have the greater problem. The uncertainties that remain there are quite large, too, until we know more about the flow patterns from determinations in wind-tunnel tests.

I think, finally, the question of reuse would have some effect on what would be done about cooling. I think it would discourage the use of ablative materials because if you want to recover and reuse you would certainly have to refurbish by starting out with a new thrust chamber. This might be beyond what one would do reasonably. Furthermore, the use of coking regenerative coolants might be discouraged by a reuse requirement.

Regenerative cooling with RP will have a coking problem but it doesn't look so severe that it would preclude its use, at least according to the data Rocketdyne has. I think the question of extended ground testing and reuse would play a role; that kind of requirement would have a large effect on what was done about regenerative cooling.

These are all obviously qualitative statements. These are backed up, I hope, to a certain extent by quantitative studies that have been made by Rocketdyne. A report on these studies is available; its contract number is NAS-8-4011. It is entitled "Investigation of Cooling Problems at High Chamber Pressures."

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13. ENGINE COOLING STUDIES

By W. R. Wagner

Rocketdyne Division
North American Aviation, Incorporated

I would like to discuss some of the five facets of this particular NASA contract study which we made for conventional chambers primarily, from the standpoint that what is applicable insofar as limitations for conventional chambers is also applicable to a certain extent to larger or unconventional nozzle design. This particular contract is NAS-8-4011. It also has an internal report number, R-3-999.

The propellants used were oxygen/hydrogen and oxygen/RP-1. The intended scope of the study was to evaluate the feasibility of thrust chamber cooling within the chamber pressure range of 1,500 to 5,000 psia. We also did look at higher values, and I will discuss that later. The thrust level was 2 million to 6 million pounds. The thrust level is not such a significant factor in terms of heat transfer.

This particular study was of an analytical and experimental type, with the experimental part handled by research and the analytical part by advanced projects, aerodynamics, of which I am a part.

The following cooling methods were investigated: regenerative, film, transpiration, ablation, dump, radiation, and combined methods.

The factors defining the upper limit of regenerative cooling are:

(1) Coolant pressure drop
(2) Wall material temperature
(3) Wall thickness requirement
(4) Thermal stress
(5) Wall heat flux conduction limit

Coolant pressure and wall temperature depend on whether one chooses a refractory metal or conventional material. Wall thickness requirement depends on what is a reasonable, say, fragility limit for a wall thickness, and what in terms of reliability is a practical minimum. The thermal stress is a very active restriction.

Results of the regenerative cooling analysis are as follows:

Conventional material:	
Chamber pressure limit	2,000 psia
Limiting factor	Coolant pressure drop
Limiting wall temperature	800° F (coolant coking)
Limiting heat flux	20 Btu/in.^{2}-sec
Thrust dependence	Slight
Refractory material:	
Same limits due to coolant wall temperature limitation (800° F)	

The limiting wall temperature of 800° F is fairly well substantiated. The coking rate appears to be heat-flux and wall-temperature limited.

Results of the oxygen/hydrogen regenerative cooling analysis are as follows:

Conventional material:

Chamber pressure limit	1,750 psia
Limiting factor	Coolant pressure drop
Limiting wall temperature	1,600° F
Limiting heat flux	36 Btu/in.^{2}-sec
Thrust dependence	Slight

Refractory metal:

Chamber pressure limit	5,000 psia
Limiting factor	Coolant pressure drop
Limiting wall temperature	2,400° F
Limiting heat flux	75 Btu/in.^{2}-sec
Thrust dependence	Slight

Conventional materials such as stainless and Inconel-X and refractory metals, tungsten and molybdenum, were studied. It can be seen by comparison that we are limited very severely in terms of high chamber pressure with conventional materials, whereas if we can utilize the refractory metal the limit can be extended significantly to 5,000 psia and perhaps even above, depending on where one wants to cut off the coolant pressure drop. Note the difference in limiting wall temperatures for conventional and refractory material.

Film cooling analysis results with oxygen/hydrogen and oxygen/RP-1 are given as follows:

O_2-H_2:

Chamber pressure limit		>5,000 psia
Flow requirement	1,000° F	1,600° F
	$0.6\% < \frac{\dot{w}_c}{\dot{w}_t} < 1.0\%$	$0.3\% < \frac{\dot{w}_c}{\dot{w}_t} < 1.0\%$
Thrust dependence		Moderate
Performance loss		Small
Manufacturing problems		Small

O_2/RP-1:

Chamber pressure limit		>5,000 psia
Flow requirement	1,000° F	1,600° F
	$5\% < \frac{\dot{w}_c}{\dot{w}_t} < 14\%$	$2\% < \frac{\dot{w}_c}{\dot{w}_t} < 7\%$
Thrust dependence		Moderate
Performance loss		Large
Manufacturing problems		Small

Note that the percentages of the coolant flow relative to the total flow are quite moderate for O_2/H_2 within the range of perhaps a maximum of 1.5 percent. This is for a condition of cooling only the combustion zone. We feel, in general, that it is unnecessary to go to the extent of cooling the nozzle section.

The requirements for O_2/RP-1 are, however, much higher because of the inferior coolant capability of the RP-1 as compared with the hydrogen. The significant point to be noted here is, we feel, that the performance loss is quite large, percentagewise, in the case of RP-1, as opposed to the hydrogen case, where the molecular weight is quite small, and hopefully the gain in I_{SP} will be much greater.

Results of the study of combined cooling methods are as follows:

- Mass transfer - regenerative
  - Method advantage
    - Low performance loss
    - Minimization of pressure drop
  - Method evaluated
    - Film - regenerative cooling

- Mass transfer - radiative
  - Method advantage
    - Useful only at low heat fluxes
  - Method evaluated
    - Film - radiative

Here lies the hope for RP-1 at intermediate pressure chamber ranges, say 3,000 psi. We don't see much use of radiative mass transfer over the conventional mass transfer technique. The radiative method can handle only a small heat flux.

Some results of the experimental phase of the study of hydrogen are presented as follows:

Hydrogen Regenerative Cooling
- Purpose: Establish surface roughness effects
  - Coolant equation at high Reynolds numbers
- Method: Electrically heated test sections
  - Moderate heat flux - stainless steel
  - High heat flux - molybdenum
- Results: Surface roughness very important
  - Heat flux 38 Btu/in.^{2}-sec (maximum)
  - Pressure, 2,500 psia (maximum)

Hydrogen Pressure Drop Study
- Purpose: Evaluate correctness of analytical method
- Method: Experimental checkout of shaped coolant tube
- Results: Close agreement with analytical approach
  - Frictional pressure drop
  - Momentum pressure drop
  - Roughness effects

We studied very high Reynolds numbers, high heat fluxes, substantiated equations for roughness effects, and coolant pressure drops to see whether we could with our present analytical techniques extrapolate to higher chamber pressure conditions.

The following results are for the RP-1 study:

RP-1 Regenerative Cooling
- Purpose: Development of coolant equation for high pressures and velocities
- Method: Electrically heated Inconel and nickel tubes
- Results: Successful demonstration at heat fluxes 20 Btu/in.^{2}-sec
  - Coolant velocity maximum, 317 ft/sec
  - Pressure maximum, 3,170 psia
  - Derivation of applicable heat-transfer equation
  - Determined coking limitation

RP-1 Coolant Recovery Temperature Evaluation
- Purpose: Establishment of proper coolant bulk driving temperature for heat transfer at high Prandtl number
- Results: Recovery factor temperature
  - No agreement with $\sqrt[1/3]{N_{Pr}}$
  - Compared favorably with more complex analytical theory

The design conditions imposed for this study were:

Thrust, million pounds	2 to 6	
Chamber pressure, P_c, psia	1,500 to 5,000	
Expansion area ratio, ϵ	16 to 30	
Tube wall materials	347 stainless, Inconel-X, molybdenum	
Propellants	lox/RP-1	lox/LH_2
Characteristic length, L^*, in.	35	25
Mixture ratio	2.35	5.0

It makes very little difference what the choice of the expansion ratio is, because of the relative ease of cooling the nozzle section. What was chosen here were three materials, including a low-conductivity low-strength material, a high-strength low-conductivity material, and a refractory which had high strength and high conductivity. Mixture ratio points were fixed for each propellant.

Figure 1 shows the variation of heat flux with chamber pressure for lox/RP-1 engines. It is fairly insensitive to thrust. Note the peculiar shape of the curve. In the region shown by the dashed line there is a significant effect of the carbon layer which is deposited on the wall. As a result the heat fluxes are quite low until higher mass velocities are reached which correspond to a higher chamber pressure; it appears that the carbon layer scrubbed away from the wall and thus the heat flux goes up quite markedly.

The lox/hydrogen (fig. 2), on the other hand, starts out with significantly higher heat flux and continues throughout the chamber pressure range with a higher value. A comparison is made here between different wall temperatures and different thrusts. The two engines with refractory metal $T_{wg} = 2{,}400^\circ$ F would lower the effective heat flux due to the higher wall temperature, but not significantly.

We are talking about values of heat flux on the order of 50 Btu/(sq in.)(sec) which are quite high in comparison with, say, the lox/RP-1 engine. Presently the F-1 has about 3 to 4 Btu/(sq in.)(sec) at the throat.

The principal limitation for both of these propellants is the coolant pressure drop. Figure 3 is a summary of coolant pressure drop up to a chamber pressure of 5,000 psi. One can see that the two curves on the left should be smooth continuous curves. In the case of lox/hydrogen the curve rises sharply in the vicinity of 1700 psi; for lox/RP, with conventional material, the curve rises sharply in the vicinity of 2,000 psi. Imposed here are thermal conduction limits; that is, the wall with an allowable differential temperature across it can't transfer any higher. This corresponds to an infinite pressure drop. With the utilization of a refractory metal, one can see how significantly the chamber pressure range can be extended.

One of the basic limiting factors in the case of regenerative cooling is thermal stress. (See fig. 4.) Two principal thermal stresses are induced. The first is from the thermal gradient imposed by the heat flux from the hot surface. The other is the longitudinal thermal stress, from the hot surface to the cold surface, which is essentially at coolant ambient temperature.

There too are very serious limiting factors in terms of reliability when one of these engines is fired at a higher chamber pressure. The stresses are such that over a finite operating life failure can be expected.

Figure 5 shows a comparison of materials in terms of heat flux to the yield point for tangential thermal stresses. For conventional materials the heat flux is quite low, 3 to 6 Btu/(sq in.)(sec). Inconel raises this somewhat. Refractories indicate a marked superiority, again for a defined minimum wall thickness from the manufacturing consideration standpoint.

Figure 6 presents a comparison of the same materials in terms of temperature differential due to longitudinal thermal stress. The refractories again are much higher than Inconel and 347 stainless steel. We did not investigate the limitations with these particular materials, as well as the definite limitations with refractories.

Figure 7 is a short presentation of comparative methods with mass transfer cooling techniques. We have three basic methods: Transpiration cooling, film cooling, and ablation cooling, with the first two differing only in terms of perhaps the finite or discrete number of slots along the length of the chamber. That is, if we extended the film cooling case to an infinite number of slots, we would in essence have a transpiration cooling method.

We studied these three types and, of course, there are manufacturing problems associated with them, but we haven't for this particular study worried about that in great detail. For transpiration cooling, we find very severe problems in terms of getting a reasonable thickness or weight. The weight always seems to come out to be much higher than we would like to have.

There are mechanical problems with introducing the film cooling which go hand in hand with utilization of the film cooling. We would like to minimize the percentage flow that is introduced. Ablation, of course, is a limited method in terms of the time. It can operate only in a defined limit of time, depending on wall thickness.

Figure 8 shows a variation of ablated thickness and char depth with area ratio for two material types, Teflon and phenolic Refrasil. This particular study was of 200 seconds duration with lox/hydrogen. It doesn't, based on our present theory, make much difference whether the propellant is lox/hydrogen or lox/RP-1 because of similarity of flame temperatures. If we assume a char type of material the wall thickness is fairly limited. Because of the conduction-limited nature of the ablation mechanism there is no appreciable effect at these high film coefficients of area ratio, so that the variation of thickness with area ratio is essentially constant for phenolic Refrasil.

In contrast, the ablative thicknesses for a noncharring type of material, Teflon, are way out of reason.

Figure 9 presents substantiation for some of the transpiration cooling percentages that you see. What we did was compare all the available theories and modify what we thought was the most suitable equation. The curve is a fit of some transpiration data in terms of a nondimensional parameter and mass velocity ratio, mass velocity of the coolant relative to the mass velocity of the gas. We got fair agreement with some modification of the equation. The principal unknown here in terms of transpiration is the effect of variation of coolant properties, that is, the difference between, let's say, the heat capacity of the coolant and the gas. Because of the more defined nature of transpiration, the analysis or analytical development is accomplished much easier than in the case of film cooling.

Figure 10 is a typical summary of the transpiration cooling requirements for lox/hydrogen. This is for the combustion zone only. The curves are for two wall temperatures, $1{,}000^{\circ}$ and $1{,}600^{\circ}$ F. In the case of transpiration the average wall temperature on the gas side surface is the peak value. Of course, this results in a much better utilization than can be accomplished with the film cooling technique. But the percentages are very small, 0.5 percent.

Note that thrust is a very significant factor. You can see the degradation in requirement as thrust increases. At these high thrust levels, because of the diminishing surface area relative to thrust, a very small percentage is required.

Figure 11 is a summary of transpiration cooling requirements for lox/RP-1, with again a little higher requirement; on the order of 1 percent of the total flow is required.

The last of the three mass transfer cooling techniques is that of film cooling. (See fig. 12.) What we did was go through all suitable available data, of which these are part, to determine what sort of a model would be acceptable for extrapolation to the higher chamber pressure condition. These represent some NASA data which we recorrelated in terms of an efficiency of cooling. We have modified this model to a certain extent. What it seems to point out to us is that at the higher mass velocity condition, that is, if you bring in a higher coolant mass velocity relative to the gas mass velocity, there is a definite diminishing in the effectiveness or efficiency of the film coolant. There are a certain number of limitations here, and we feel there are a certain number of limitations in the data.

In terms of our analysis, under the imposed conditions, the film cooling requirements for lox/hydrogen do come out much higher than in the case of transpiration cooling. (See fig. 13.) Approximately 1 to 1.5 percent is required again for cooling the combustion zone alone. We looked at the analysis for the nozzle section but, as I pointed out, we didn't feel there was any need for cooling the nozzle section with this method. The surface area or the combined integrated heat flux times the surface area would require a significantly greater number than this if one did want to cool the nozzle section.

Figure 14 presents a summary of the recorrelation of the effectiveness or cooling efficiency parameter versus combined mass velocity ratio parameters. This points out to us that we perhaps would like to have a lot of film cooling slots with a very small amount introduced in each coolant section as one would anticipate for the greatest efficiency.

Figure 15 shows film coolant requirements for lox/RP-1 in terms of total flow. Ten percent of the total flow is required for cooling the combustion zone at 2000 psia increasing with chamber pressure increase. Thrust again is a very strong variable. One would like, in order to minimize flow requirements, to tolerate as high a wall temperature as possible.

Great promise is held for intermediate chamber pressure ranges. Figure 16 shows the trade off between percent of regenerative cooling (heat-transfer coefficient) and percent of film cooling for lox/hydrogen. It turns out that one can make a kind of intermediate trade off between the pressure drop and the percentage of film coolant requirements. So for the intermediate chamber pressure ranges in the vicinity of, say, 1,500 to 3,000 psi for lox/RP-1, this technique would be the most advantageous in our opinion.

Table I is a summary of pump pressures for increasing chamber pressure. For lox/RP-1 we don't feel that regnerative methods can be utilized at these higher chamber pressures. One has to go to a film or combined film-regenerative technique at the upper chamber pressures.

For lox/hydrogen with conventional materials we are limited to 1,700 psi, as previously mentioned. With refractory metals we can extend this limit depending on what we want to accept as the pressure drop or pump outlet requirement. With film or combined film and regenerative techniques the pressure drop requirements can be significantly reduced.

Figure 17 illustrates the validity of using radiation cooling at high chamber pressures. It can be seen that in terms of wall temperatures one would have to go out to a considerable area ratio before reasonable values of wall temperature occur. As a result, this method is probably not advantageous for use until the area ratio is perhaps 50 or 100.

These two sets of curves are a comparison of lox/RP-1 with lox/hydrogen. You would have to go out much, much farther if you wanted to stay in, say, the 3,000 to 3,500 range of wall temperature. The lower curves are for lox/RP-1 where we are assuming some benefit of carbon. There are other methods, and one can combine all of these methods together, actually.

In terms of having a film-cooled combustion zone and a regenerative cooled nozzle or a regenerative combustion zone, one can think of a number of varieties and combinations of cooling methods such as:

Other Cooling Methods:
- Simultaneous film + radiation cooling
- Dump cooling

Combined Methods for Combustion Zone and Nozzle:
- Film
- Transpiration
- Ablative
- Regenerative

In the case of ablative cooling the serious unknown factor there is the stability of the charring surface.

QUESTION PERIOD

MR. WOODCOCK: I would like to know what you mean by "combustion zone" and what you mean by "nozzle."

MR. WAGNER: The combustion zone includes from the injector phase to the throat, and nozzle includes from the throat on out to the expansion section.

MR. WOODCOCK: Don't you feel that the heat flux loads are pretty high at least to some small distance?

MR. WAGNER: That certainly is true. We had to define a point. However, perhaps if one went down to 3 to 1 expansion ratio or 5 to 1 these results would not change significantly.

MR. HALL: Did you include in your studies also cooling of downstream portions by turbine exhaust gases?

MR. WAGNER: No, in this particular study we didn't. We are presently on our own examining parametrically the studies of utilization of turbine exhaust gas in the nozzle section. It looks very attractive. However, at high area ratio, say 10 or 20, cooling with turbine exhaust is not of any value because the temperature is quite high to begin with, so we can't have much available temperature rise before you reach wall temperature limits.

MR. BARTZ: How do you think refractory metals can be used in a practical sense, with regard to spinning, handling of thin sizes, and availability of metals that you can use? In other words, are we whistling in the dark when we talk about these refractory metal tubes or are they around the corner.

Secondly, what might this do to the weight of the system compared with the weights of, say, stainless steel or Inconel?

MR. WAGNER: One of the principal limitations with refractory metals is, of course, the disilicide or whatever coating one must impose upon it for oxidation resistance. The other problem is the recrystallization point which occurs at about 2,500° F; one can't utilize the full potential near the melting point of the material. Then, perhaps, in the case of hydrogen, the embrittlement problem occurs. We have underway development programs on a small scale of forming and so forth with these materials. They are several years off, I am sure, both with regard to the forming problem and the coating.

MR. BARTZ: I notice in your work you had to get tubes by drilling out rods. That didn't seem practical for thrust chamber construction. When did the manufacturers say you are going to get tungsten tubes or molybdenum tubes?

MR. WAGNER: If you present to them an attractive enough long-range program they are always interested. They seem to feel in the case of molybdenum, or perhaps these two materials, that within a short range, say 5 years or so, they might have reasonable feasibility.

MR. IACOBELLIS: I would like to add to that. In the model I showed earlier, we actually were able to test some TZM tubes that were supplied to us. They were supplied in inches. It is in its infancy, but they are starting to do some work with supplying tubes.

MR. BARTZ: It sounds like the schedule is such that you would be hard put to make this Phase II or Classification II, that we discussed previously. It is almost like a third generation, isn't it, that we are talking about?

MR. IACOBELLIS: I don't want to confuse two things now. The presentation we made earlier did not use refractories. I want to make that clear. From there I agree with you that they are not right around the corner as far as being able to use them. But all I wanted to point out is that we have received some of these refractories in tube form. They are straight tubes. When you start bending them and swaging them, that is something else.

MR. BARTZ: I pursue the point because your limit analysis showed that you can get 2,500 psi regenerative cooling with hydrogen. It seems to me that of the various methods of cooling, we know how to do that the best. Therefore, the use of that method of cooling really depends on the availability and practicality of these refractory tubes. If they are available and practical, it seems to me that is the most straightforward way of going. If they are a sort of theoretical chariot in the sky, we ought to realize that and eliminate that.

MR. WAGNER: In looking at the mass transfer cooling methods, and particularly film cooling, we have run studies which indicate that film cooling with

[redacted]

hydrogen is particularly attractive. These studies that have been run on a smaller size indicate that this method is practical. We have data to confirm some of these film cooling results with the hydrogen.

MR. BARTZ: But you still are nowhere near the kind of chamber pressures and conditions you are talking about.

MR. WAGNER: That is true.

MR. BARTZ: It is too easy to fall into the trap of saying this is a nice curve - I have some data here and the data seem to follow this curve ±20 percent and we can extrapolate it out to whatever we want. I think that is a hazardous occupation for projecting engines on the schedule we are talking about until we really have some solid high-pressure film-cooling data. I feel that maybe your approach is a bit on the optimistic side.

MR. IACOBELLIS: I would like to add this point. We have come to this crossroad just as you indicate. We have selected the film cooling with the more conventional methods as opposed to going further with the regenerative cooling technique with the refractories.

MR. BARTZ: But you did it on faith and now we are going to have to see what the data show.

MR. IACOBELLIS: Right.

MR. BURLAGE: There are a few programs underway in the country now to bring refractory metal technology into being, particularly in the Navy. It is one of those things that may fall out to us one of these days.

MR. BARTZ: Do you think the time schedule is compatible with what we have been talking about?

MR. BURLAGE: I don't think that is the point in the advanced technology. One would look to see what one could do with it. I think that is what we are after. It is not one of these pie-in-the-sky things. There is a definite attempt to make this technology available.

MR. BARTZ: I have another comment. It may be obvious, but the thing ought to be recognized, that the kind of film cooling we are talking about occurs at such high pressures that we don't get the heat of vaporization as you might think. It is like injecting gaseous propellant into the system. It is a sensible entropy kind of thing and not heater vaporization. So that the data most applicable to these correlations were gaseous injection in the hot gas flows and not the kind of film cooling you might be familiar with where you dump a liquid like water or RP into a liquid system, because there is no real phase change.

MR. WAGNER: To the extent that with hydrogen we have a rapid change of phase, in other words, it is introduced at 50° R, it behaves very much as a perfect gas; it is very similar.

[redacted]

MR. BARTZ: There is no big heat sink there?

MR. WAGNER: No, one cannot use the latent heat.

MR. BARTZ: I would like to take exception to your conclusions about charring ablation. I think that is a very optimistic assumption. Practically all the data are at higher pressures. Most of the ablation work around the country has been at lower pressures. The small amount of data that have occurred at higher pressures seem to confirm the fact that you approach steady-state ablation as confirmed by the article on ablation by Adams in the ARS Journal. There you reach ablation rates which are considerably higher than you are talking about here. In fact, we made some calculations in, I think, 10 inches of ablation under the conditions you are talking about at about 2,500 psi.

If you assume there is no spalling off, we agree the char depth is very encouraging. I think the data show that there is spalling off; a steady state of ablation process is approached and one must be ready to expect very high ablation rates at these pressure conditions. So nobody is really very serious about ablative cooling for these kinds of thrust chambers. It is an academic point. However, it ought not to go without comment.

MR. BURLAGE: That report is available and has been distributed to all NASA centers and to other contractors.

TABLE I

MINIMUM PUMP DISCHARGE PRESSURE FOR VARIOUS THRUST CHAMBER COOLING METHODS

LO_2/RP-1

COOLING METHOD	MINIMUM FUEL PUMP DISCHARGE PRESSURE, PSIA		
	P_c, 1500 PSIA	P_c, 3000 PSIA	P_c, 5000 PSIA
REGENERATIVE WITH CONVENTIONAL METHODS	2220	—	—
REGENERATIVE WITH REFRACTORY MATERIALS	2220	—	—
FILM COMBUSTION ZONE–REGENERATIVE NOZZLE WITH CONVENTIONAL MATERIALS	2160	4210	6350
FILM, TRANSPIRATION, OR ABLATION	2120*	3910*	6050*

* THESE VALUES ALSO HOLD FOR THE OXIDIZER PUMP.

TABLE II

MINIMUM PUMP DISCHARGE PRESSURE FOR VARIOUS THRUST CHAMBER COOLING METHODS

LO_2/LH_2

COOLING METHOD	MINIMUM FUEL PUMP DISCHARGE PRESSURE, PSIA		
	P_c, 1500 PSIA	P_c, 3000 PSIA	P_c, 5000 PSIA
REGENERATIVE WITH CONVENTIONAL MATERIALS	2520	—	—
REGENERATIVE WITH REFRACTORY MATERIALS	2220	4310	7550
FILM COMBUSTION ZONE-REGENERATIVE NOZZLE WITH CONVENTIONAL MATERIALS	2290	4560	6700
FILM, TRANSPIRATION, OR ABLATION	2120*	3910*	6050*

* THESE VALUES ALSO HOLD FOR THE OXIDIZER PUMP

MAXIMUM HEAT FLUX VS CHAMBER PRESSURE FOR LO_2/RP-1 ENGINES

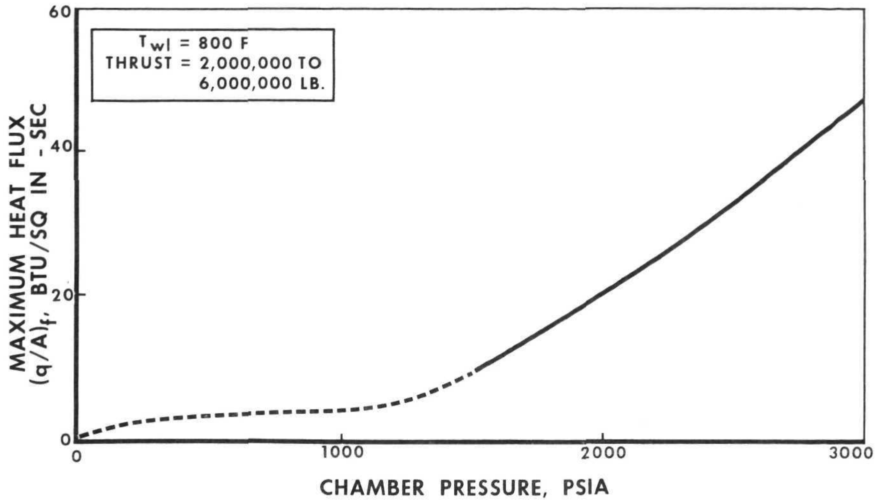


Figure 1

MAXIMUM HEAT FLUX VS CHAMBER PRESSURE LO_2/LH_2

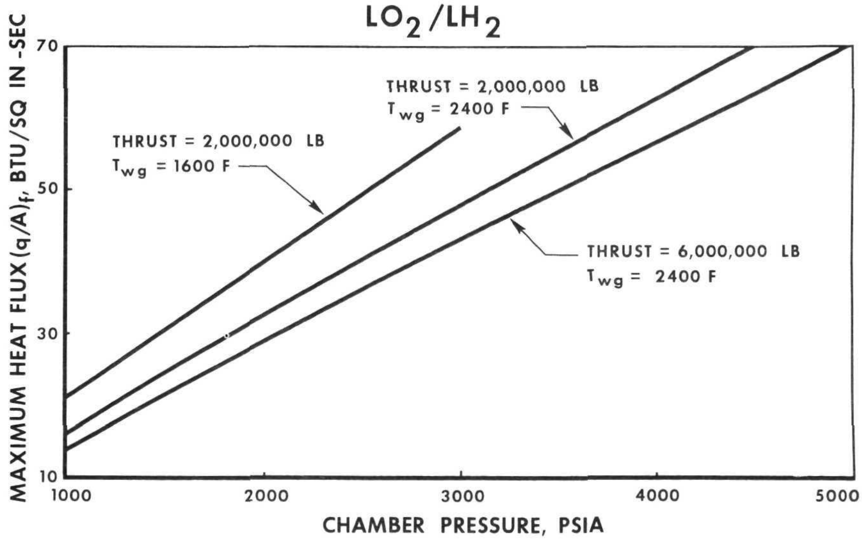


Figure 2

MINIMUM COOLANT PRESSURE DROP VS CHAMBER PRESSURE

LO_2/RP-1 AND LO_2/LH_2 6,000,000 POUNDS THRUST

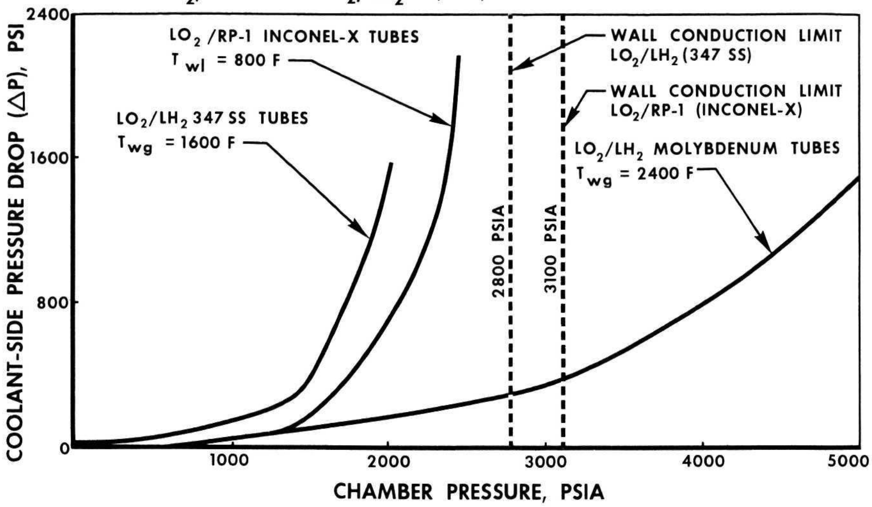
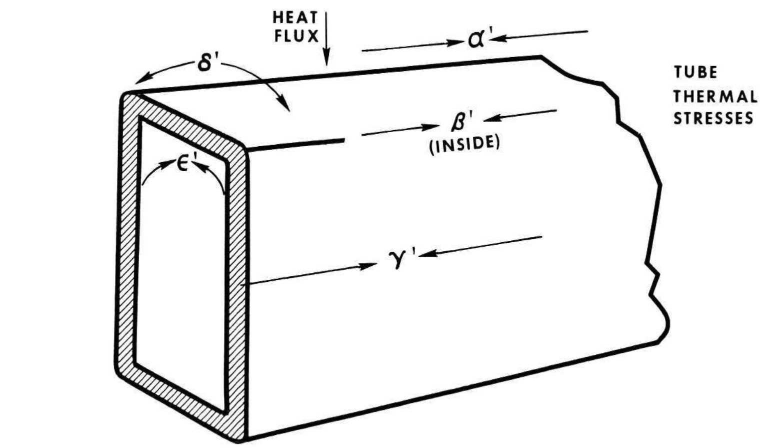


Figure 3

WALL MATERIAL THERMAL STRESS



LONGITUDINAL STRESSES:

OUTSIDE SURFACE α' EXPANDS RELATIVE TO INSIDE SURFACE β'
INSIDE SURFACE β' EXPANDS RELATIVE TO INTERIOR SURFACE γ'.

TANGENTIAL STRESSES:

OUTSIDE SURFACE δ' EXPANDS RELATIVE TO INSIDE SURFACE ϵ'

Figure 4

[redacted]

HEAT FLUX AT THE YIELD POINT DUE TO TANGENTIAL THERMAL STRESSES

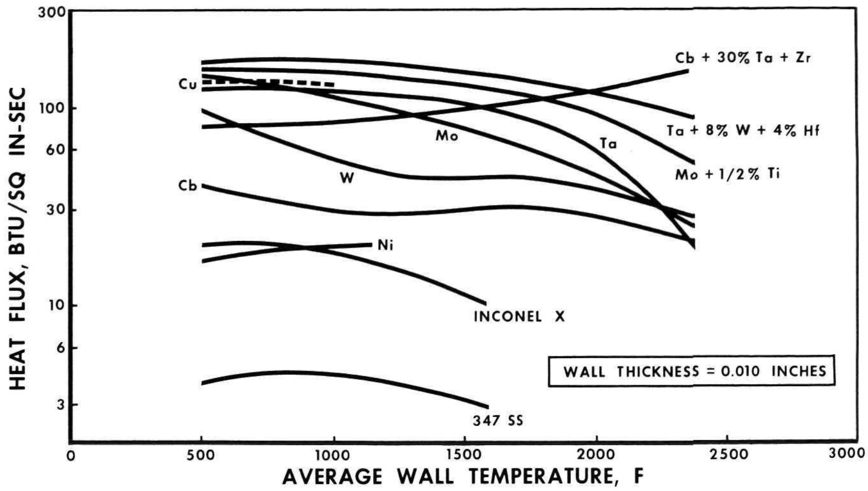


Figure 5

ALLOWABLE TEMPERATURE DIFFERENTIAL AT THE YIELD POINT DUE TO LONGITUDINAL THERMAL STRESSES

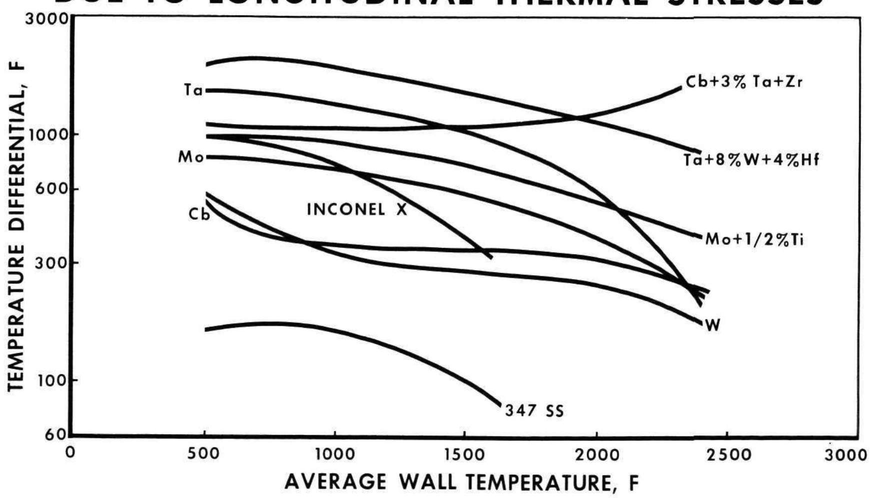


Figure 6

[redacted]

MASS TRANSFER COOLING METHODS

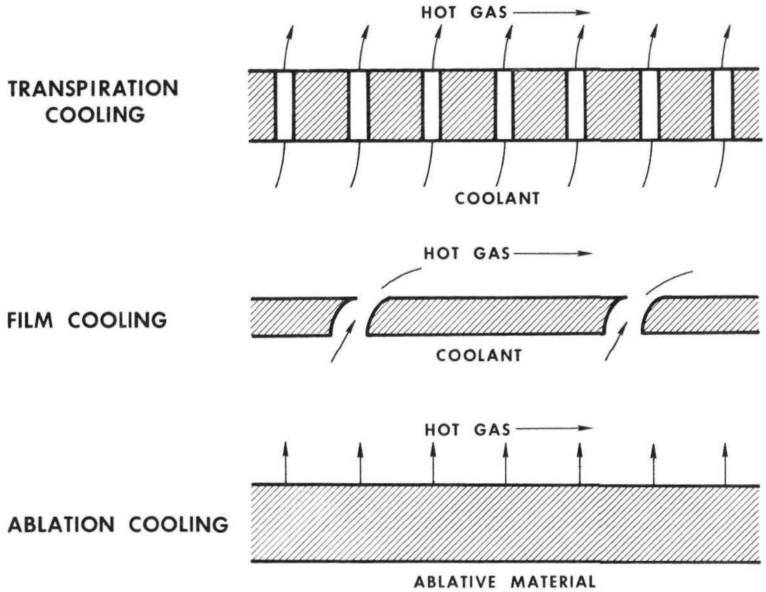


Figure 7

ABLATED THICKNESS OF TEFLON AND CHAR DEPTH OF PHENOLIC REFRASIL

200 SECONDS, LO_2/LH_2, ENGINE
2 MILLION POUNDS THRUST

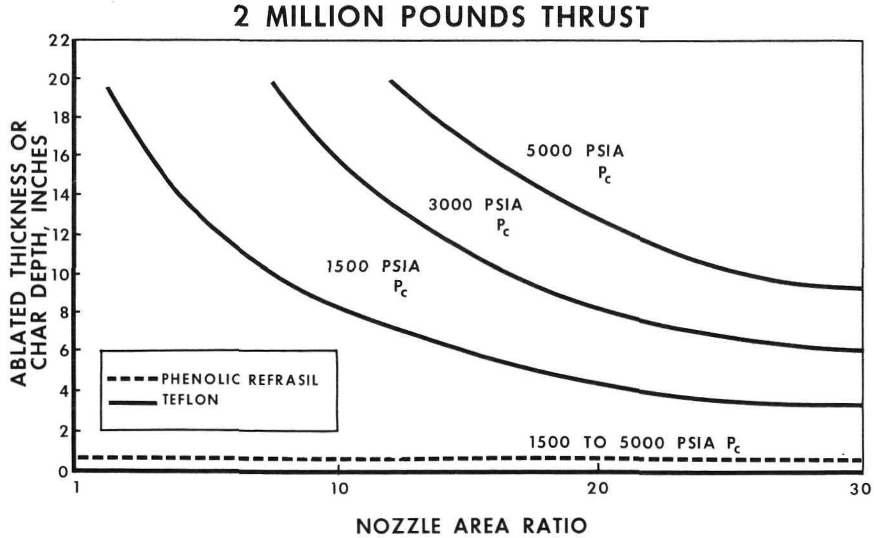
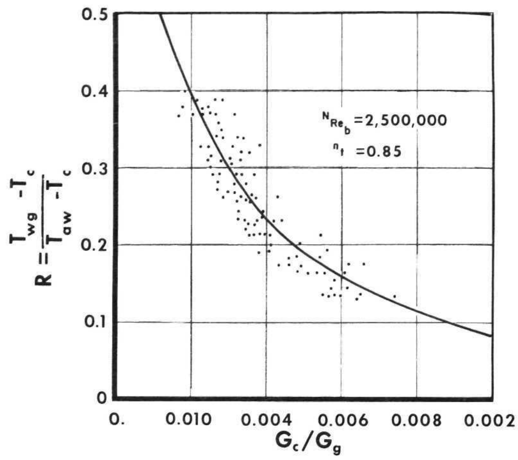


Figure 8

TRANSPIRATION COOLING
COMPARISON OF MODIFIED RANGE RANNIE EQUATION TO EXPERIMENTAL DATA OF BRUNK



$$\frac{1}{R} = \frac{T_{aw} - T_c}{T_{wg} - T_c} = \left[1 + \frac{c_{p_c}}{c_{p_g}} \left(1.18\, N_{Re_b}^{\;0.1} - 1\right)\left(1 - \epsilon^{-37 n_t \left(\frac{G_c}{G_g}\right) N_{Re_b}^{\;0.1}}\right)\right]\left[\epsilon^{37 n_t \left(\frac{G_c}{G_g}\right) N_{Re_b}^{\;0.1}\, N_{Pr_m}}\right]$$

Figure 9

TRANSPIRATION COOLING
REQUIREMENTS-LO_2/LH_2
(COMBUSTION ZONE)

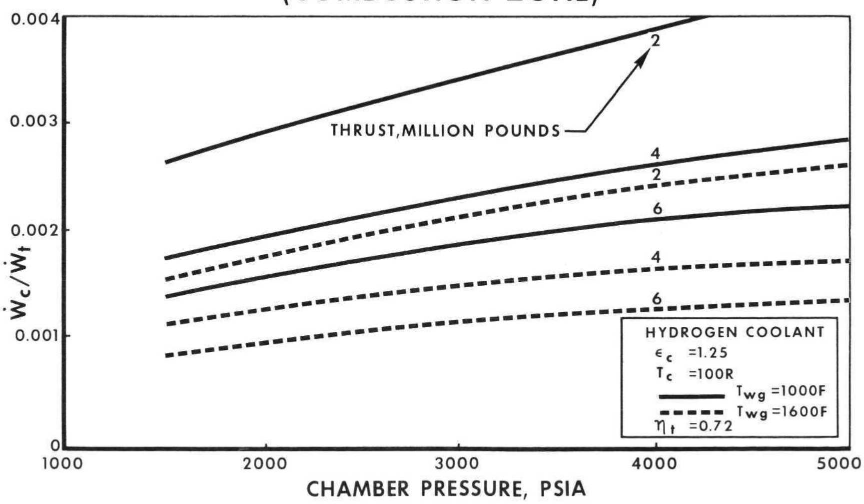


Figure 10

TRANSPIRATION COOLING REQUIREMENTS - LO_2/RP-1
(COMBUSTION ZONE)

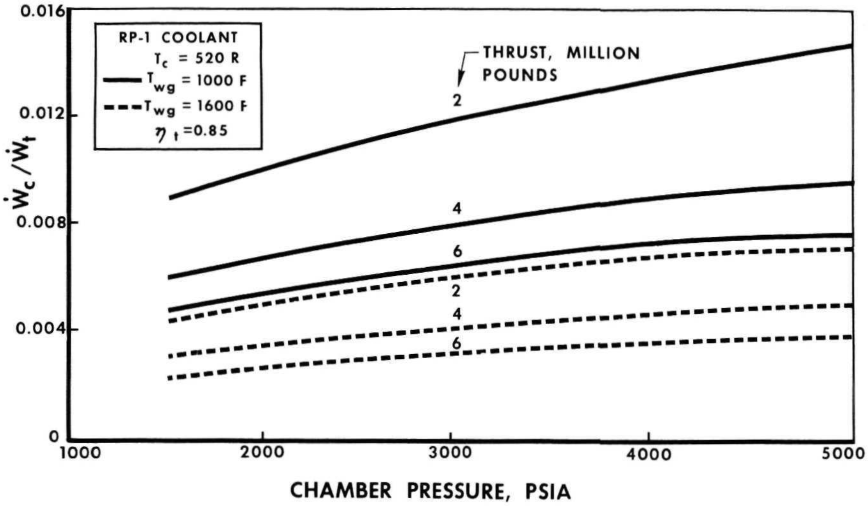


Figure 11

AIR FILM COOLING DATA OF PAPELL AND TROUT
($V_C < V_G$)

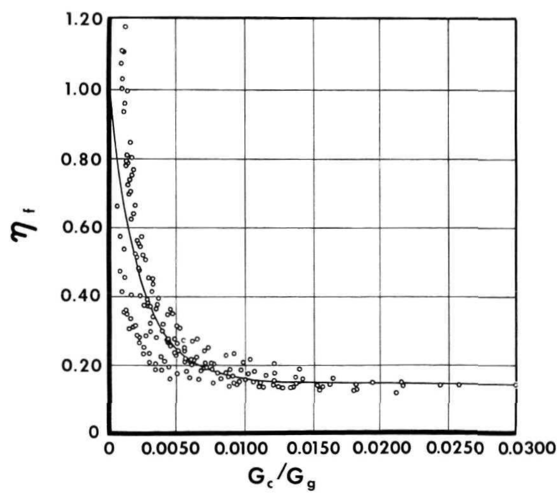


Figure 12

FILM COOLING REQUIREMENTS-LO_2/LH_2
(COMBUSTION ZONE)

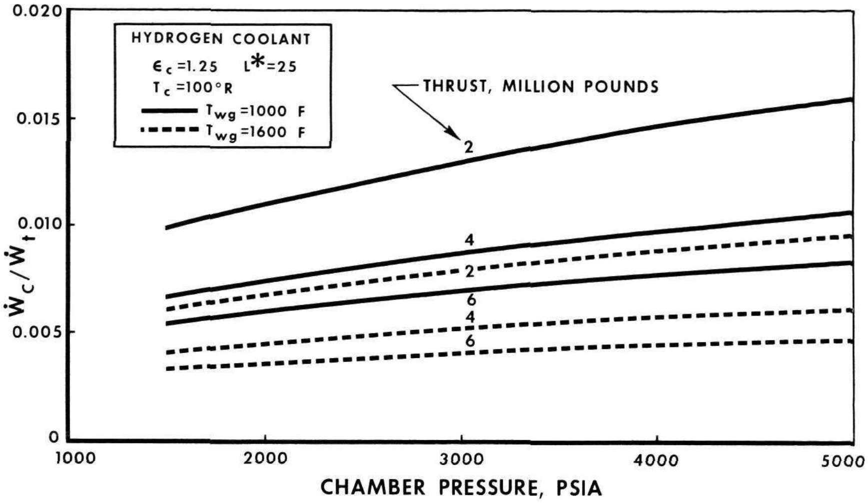


Figure 13

FILM COOLING CORRELATION

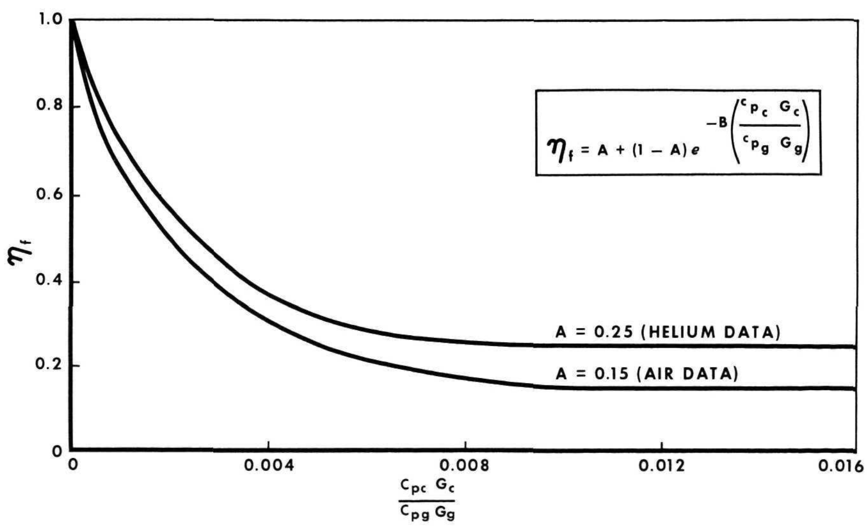


Figure 14

FILM COOLANT REQUIREMENTS - LO_2/RP-1
(COMBUSTION ZONE)

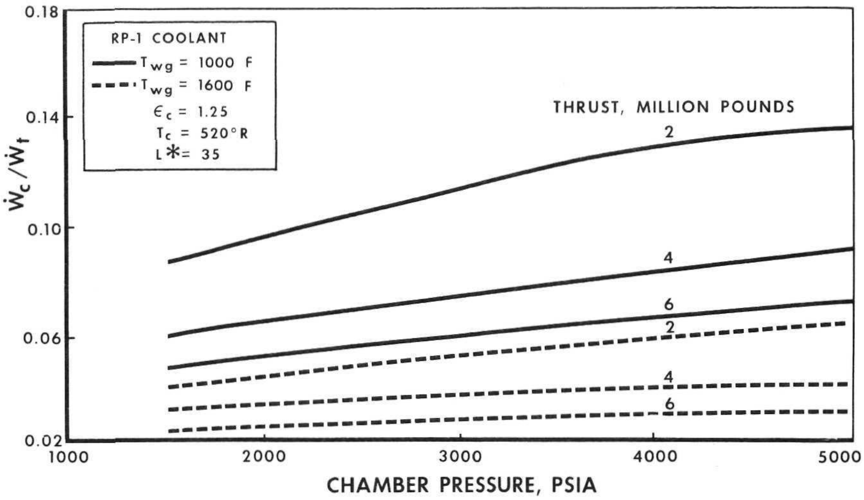


Figure 15

HEAT FLUX DISTRIBUTION
VARIOUS DEGREES OF SIMULTANEOUS FILM AND REGENERATIVE COOLING

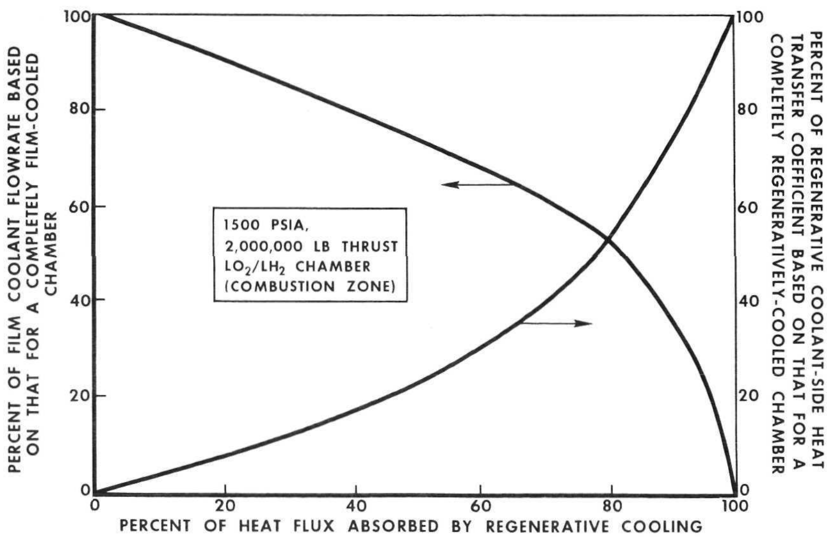


Figure 16

GAS SIDE WALL TEMPERATURES REQUIRED FOR RADIATION COOLING

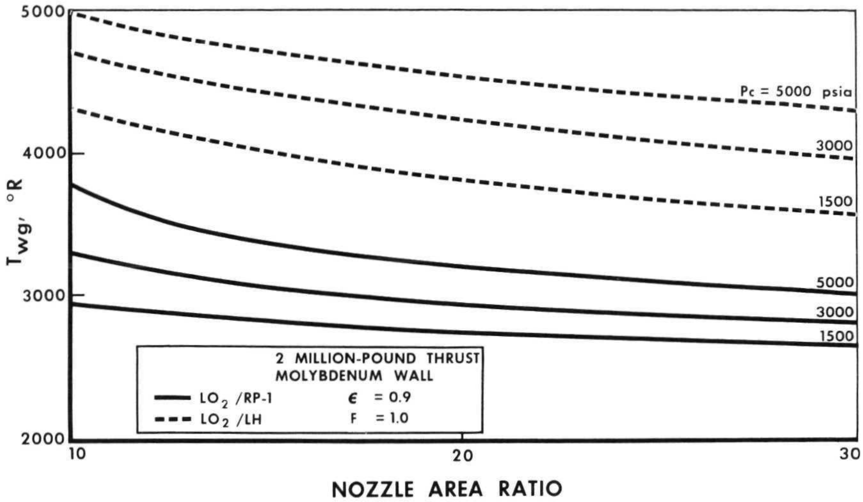


Figure 17

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14. ENGINE COOLING EXPERIMENTS

By John Chamberlain

Pratt & Whitney Aircraft Division
United Aircraft Corporation

We were not expecting to give a presentation and have no figures to show. Furthermore, I don't happen to be connected with a project so I can talk freely. However, I want to note that this work on film cooling at high pressures under combustion conditions was done under Air Force Contract 046177435. The progress reports should be checked for the details. If there is a question about the correctness of something, you can check me there.

The things we have been trying to do in this program are first to measure heat-transfer rates in oxygen/hydrogen rocket engines running at the order of 3,000 psi. Actually there you have a range of chamber pressures from 300 up to 3,000 psi. And this phase of the work I believe is pretty much complete.

We are also just getting into tests of the effectiveness of film and transpiration cooling under this same range of conditions with oxygen/hydrogen propellants. We started out some of this work in a 5K chamber and are continuing it in a 10K thrust chamber.

To reduce the costs in the program it was decided that it would be desirable to make measurements and test samples on a probe going down through the middle of the throat of a rocket nozzle rather than to build the whole nozzle out of the material of interest. So, much of the work has been done in this kind of a configuration.

Figure 1 shows a probe bore in the middle, and a small slot for liquid oxygen flow, surrounded on both sides by small slots for gaseous hydrogen. These dimensions are fairly small, like 0.012- or 0.014- or 0.016-inch hydrogen slots.

The flow was purposely made symmetrical in an attempt to get a uniform heat-transfer rate onto the probe and onto the outer wall. Heat-transfer rate was measured by measuring the rate of temperature rise in the wall material. I believe they used copper for local heat sinks. As a matter of interest they measured heat-transfer rates out to the order of an area ratio of 100 in the nozzle, since this chamber pressure is sufficient to make a nozzle of that area ratio flow full.

The order of heat-transfer rates that have been measured are of the order that the Bartz equations would predict, something like 75 Btu/(sec)(sq in.) in the throat areas.

One of the first steps was to compare the heat transfer on the probe with that on the nozzle so that one could then proceed further with work on the simple probe in the middle. This turned out to be a little bit more difficult than some might have hoped.

[redacted]

At low chamber pressures, where an injector was run with gaseous hydrogen and gaseous oxygen, one of the walls appeared to pick up about half again as much heat as the other wall. At higher pressures, with liquid oxygen and gaseous hydrogen, the reverse occurred. I am highly suspicious that this reveral was due to characteristics of the injector rather than with the gaseous or liquid oxygen or the chamber pressure changes. But it does point up one thing and that is that it takes a lot of work on an injector before you know what you have when you measure a heat-transfer rate.

The correlation of heat transfer to the probe compared with heat transfer to the nozzle is pretty well complete. Very recently, tests were started in which a film-cooled section was added onto the probe, and a 0.040-inch-high slot was provided for injection of gaseous hydrogen which flowed down through the probe. These tests were run like the previous tests, very short duration, and the temperature rise was measured in order to obtain a heat-transfer rate to the probe. It was not allowed to run to equilibrium.

Some of the data just obtained are shown in figure 2, in which heat transfer is plotted against distance from the injector face.

In this particular case the probe slot was located 2 inches upstream of the throat and the coolant flow from the slot was 3 percent of the total propellant flow. So measurements were taken in an accelerating flow region.

The theoretical heat-transfer rates for a regeneratively cooled chamber are also shown in figure 2. As you approach the throat the heat-transfer rate increases.

It is worth remembering that this is a pretty small engine. But the throat without the probe would be 1.6 inches in diameter. With the probe the throat diameter is larger but the probe takes up most of it, so the annular opening is probably of the order of 0.4 inch, or something like that. So there is a lot of surface area in here and one would expect fairly high amounts of coolant to be required.

There is another bit of comment required here. The thermocouples buried in the probe could not all be alined axially, so each point was obtained at a different spot around the circumference in a helical pattern. So local variations could have been caused by such things as a very slight blockage in the upstream cooling slot, or they might have occurred because of a high mixture ratio in that particular area in the injector.

The main thing we can do at this point is state that we are beginning to do some work in this area, not that we have anything useful at the moment.

It is worth pointing out one other item here. If we plot a graph of mixture ratio against theoretical heat transfer to a wall of low temperature, we get a curve not quite as flat at the top but one that over quite a wide range of mixture ratios has a roughly constant heat-transfer rate. Thus, you can take an injector and put it in a chamber and if the mixture ratio varies in some spots from 3 to 7, this won't change the heat-transfer rate on the wall a great deal, and you can measure the heat-transfer rate even if the flow from the injector is not really uniform. But when you dilute this mixture, that same variation in mixture ratio in the injector will result in extreme variations in temperature on your test piece. So to do any work one has to have an injector where the mixture ratio is very uniform or at least you have to know what it is.

Every time we build an injector, we put that injector in a special rig and measure the local flow at each specific spot around the perimeter of the oxygen flow by flowing water through the injector. We turn off the water and turn on air or nitrogen and measure that flow around the perimeter of the injector. We will remove the face material and grind away slightly here and slightly there to try to make it as uniform as we can. And even after we get done with this process, if we get to the point where we have variations in flow around the circumference of the injector with a spread of less than 30 percent or so, we feel we are doing fairly well.

It may sound a little bit easy, but you get at it and it gets worse; you run the injector and the lox portions of the injector run cold and the other portions run considerably hotter. They will move as much as one-third of the gap width relative to each other, and throughout the period of the run this is a transient all the way. So the lox parts are continually cooling down and the hydrogen parts are continually warming up. It takes a lot of care. We hope we will get some better information. When we are done it will still look fairly crude but I hope it will be helpful. Information is almost nonexistent at present.

QUESTION PERIOD

MR. GRAHAM: I wonder why you would expect that probe to have the same kind of boundary layer as you would have on the nozzle, and why you would ever expect these two things to match.

MR. CHAMBERLAIN: We don't expect them to be the same. In fact, there has been a fair amount of analytical work. The flow is not one dimensional in the nozzle. The rate of acceleration of the flow is different on the probe than on the outer wall. Analytical work has been done to try to remove this. Part of the tests were made to confirm whether this comparison would hold or not. So far as we can see the effects of the injector far outweigh the effects of whether you are on the inside or on the outside. These are minor as far as I can see compared with imperfections in the injector uniformity.

MR. GRAHAM: How stable was that probe?

MR. CHAMBERLAIN: Another minor problem. The probe doesn't stay straight. So though the technique, theoretically, might be very nice, it has a lot of practical problems.

MR. GRAHAM: I have a comment on Mr. Bartz's presentation, which I think perhaps the contractors would like to get into.

He made a statement that correlations on the liquid side we know within 10 percent. Heat-transfer people are pretty lax sometimes. They like to put correlations on log-log plots. Then they put scatter and everything looks nice. And they come up with missile Reynolds number correlations. Sometimes they throw in a Prandtl number. Then on the back end of that they put a wall-to-bulk ratio. The poor designer is interested in a wall temperature when he comes to design his tube. If he takes these correlations that have these nice 10, 20, and 30 percent scatter, and makes a back calculation, and calculates the wall temperature from the correlation, he might find out that he can't hit that wall temperature within 300 or 400 percent. This we have done with some of the correlations that exist, and also some of the correlations that we are trying to do on heated tube experiments.

This is particularly true when you put this wall-to-bulk ratio on the back end and put it up to maybe 0.8 or 0.9. You find out that the missile correlation then becomes independent of wall temperature. And then when you back calculate you don't have any footing when you come to wall temperature.

I might point out one or two other things in getting this kind of liquid side data that we have experienced. One of the things, and I think Rocketdyne has experienced this, too, is that with any tube that we ever run we get into some kind of an oscillation. Some of these may be down in almost the chugging mode, and may be tied into the system. Others are acoustical. These things can be real bothersome because they can give us different levels of heat transfer. I don't know of any correlation that really takes this into account. And so I think we always have to say when we give somebody a correlation, "Look out, because there are some unknowns in this thing."

Another unknown that has cropped up, too, is acceleration in flow. We often do these heated tube experiments with constant-diameter pipes. But we find that if we accelerate the flow for any reason, or decelerate it, that we can get into different modes of heat transfer and those that don't fit these nice missile correlations that we put on log paper. Also with hydrogen you have to be careful how close you are to critical temperature, because around the critical temperature (this may not be as true at high pressures) if you get down to 800 psi, you can find terrific enhancements in heat transfer around this point. So I caution anyone, at least from our experience, on these nice correlations on liquid hydrogen, because they can be very troublesome.

INJECTOR LOX RING DEFORMATION

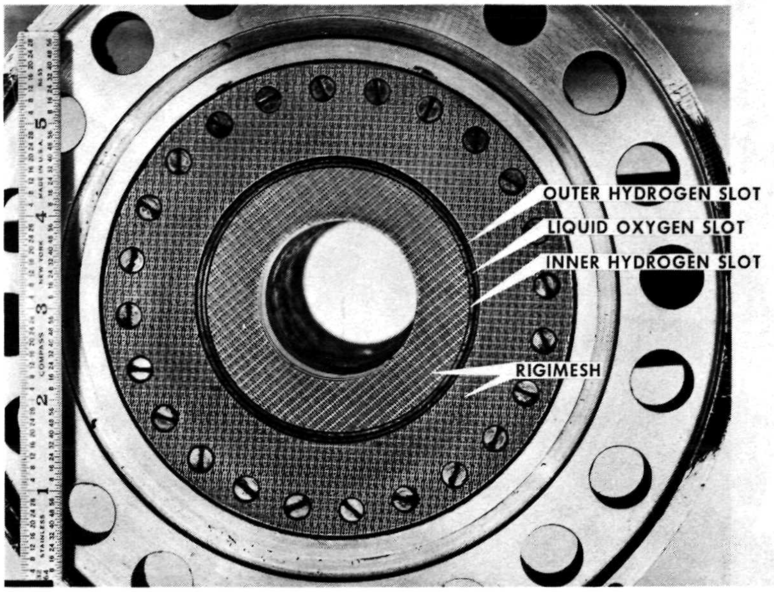

Figure 1

PRELIMINARY REDUCED DATA FROM TEST NO. 10H2

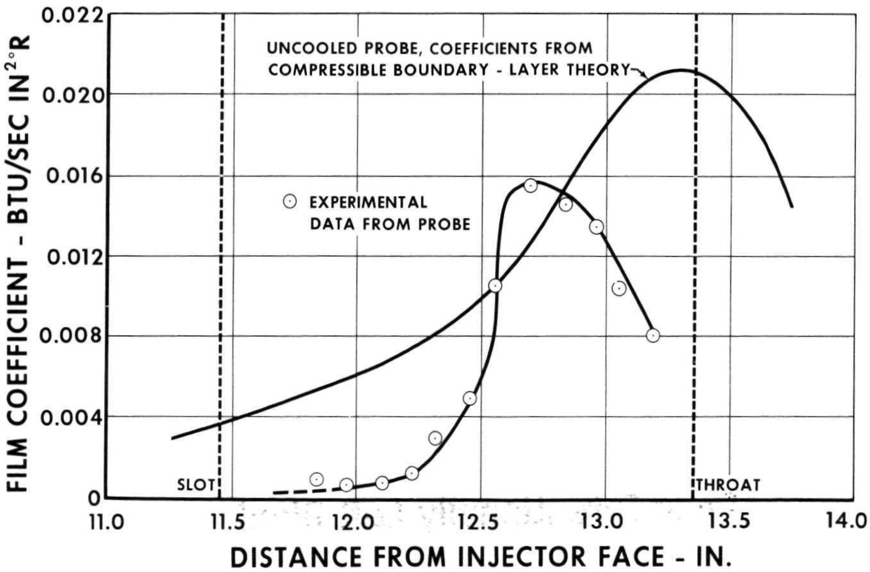

Figure 2

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15. SUMMARY DISCUSSION

MR. BARTZ: We will now excuse our contractors. There may be another point or two that might be discussed on proprietary items. I want to thank you again for your help and assistance. I think we can finish up quickly on the cooling session. I think maybe the only point that is of proprietary interest is the discussion by Mr. Mulready on this combination film and transpiration cooling. I had not seen this before and was not aware of it. It apparently is a new thing that they are considering and that they are rather optimistic about.

I was rather startled by their claims of competitive weights. When you have that much thickness of metal and you compare that to a hollow tube, or some of the other methods that we talked about, it is just astounding to me that they can end up with comparable weights.

COMMENT: They are not comparing with a hollow tube but with a Rigimesh.

MR. BARTZ: It seems to me that as long as a hollow-tube regenerative idea is possible, or film cooling is possible, you have to go back and compare with those things. It is not fair to compare it with those things that are not equally applicable.

I think it is an interesting way of looking at it. I am just fearful that it is going to be awfully heavy. You can see by the time you have solid copper or any other metal for that matter that has to be a half inch thick, even restricted to the combustion chamber, it will be of considerable weight. Does anyone else want to discuss this or any other matters?

MR. WOODCOCK: I don't remember too much about the detailed numbers that Pratt & Whitney gave. It appeared that in their engine concept, and I suppose in most engine concepts around this chamber pressure, the pump and associated hardware make up a pretty large part of the total engine weight. As I remember they said that for this concept engine weight is 4 percent greater than for the original base concept. I have no idea how it compares with the tube wall engine. It seems that if the pump is a major portion of the engine weight the concept is not too bad from the weight standpoint.

MR. BARTZ: Is it major in the sense that you could say that the thrust chamber itself is, say, 20 percent or less of the total? I have no figures but intuitively I guess that the thrust chamber itself, including injector, skirt, and all, must be 40 percent of the total mass of the engine. Does anybody have any figures?

MR. GINSBURG: Aerojet's figures show the turbopump to be only about 30 percent of the total.

MR. BARTZ: If the turbopump is 30 percent, maybe the engine is another 40 percent, and the rest would be controls. I think as a significant part of the total it is not a negligible part, and therefore big effects on the total weight of the thrust chamber would have a large effect on the engine weight.

MR. WOODCOCK: They have only been working on this idea, I believe, for something of the order of a couple of months.

MR. SUDDRETH: Rocketdyne showed a couple of limitations on the kerosene, through the coking and carbon, a little lower than shown by GD/A. GD/A, I think, referred to 2,000 psi; Rocketdyne referred to 1,750 psi as maximum. It is a rather daring step in terms of technology if you talk about a 250 psi step, and it is up in the area of marginal or more marginal design. The same thing is true with stainless-steel construction for hydrogen/oxygen. The point here is that GD/A was rather enthusiastic about these higher pressure levels with the current technology. This is in the realm of uncertainty. I think it is a pacing item.

MR. BARTZ: I completely agree. I think they are overly optimistic. It seems to me that these film cooling figures that they quote here are pretty glib. What we know about film cooling is very little and has been taken from data that have very little relevance to what we are doing. The situation could get very optimistic or black, depending on how the data turned out.

MR. SUDDRETH: For a realistic study it would seem to be worthwhile to begin to follow these things back into GD/A and Martin studies and temper the enthusiasm of the contractors in this case.

MR. BARTZ: I think it all relates to the time schedule we are talking about. If you can sit around and wait for Pratt & Whitney to get some definitive cooling data in $1\frac{1}{2}$ or 2 years, that is one thing. If you can't, then I quite agree with you, you have to back them down on the chamber pressure.

MR. BURLAGE: For what it is worth on relative weights, I was reviewing what our studies showed; in almost all cases the thermal pump is about one-third.

MR. BARTZ: Thrust chamber one-third and controls another?

MR. BURLAGE: I don't know how this breaks down. Thermal pumps are about one-third.

MR. WOODCOCK: If, when you talk about the engine, you consider the extended nozzle, the big skirts, and so forth, I think it will turn out to be a fraction of the total system weight.

MR. TISCHLER: I think we are faced with two different issues in these discussions. One of them has to do with the development program to be undertaken sometime in the future, and trying to decide at this early date what the nature or form of that development program is likely to be. And the other issue quite distinct is the issue of what should we be doing to develop technology leading up to advancements in this development possibility. I think there should be no limitation on what we explore in this area. I think it is clear that if we have areas of nonavailability of data, that is, hollow areas, we should be trying to fill them. There should be no predetermined limit to how far we can go in this area until we have actually impounded that limit

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experimentally or at least theoretically. I think we should push agressively into these particular areas, and not try to define studies that fall short of it.

MR. BARTZ: I agree on the last point. I think the contractors have a tendency to have a glib answer for everything and not face up to holes in the data, whereas I think one has to face up to these and actively seek programs to define these and not gloss them over. I think they have to do that, and I think that is too bad.

MR. WOODCOCK: Some of the contractor studies were more on the basis, assuming that high chamber pressure can be achieved, of what looks the best from an optimization standpoint. I don't think they pursued at any depth whether 2,000 psi was feasible. Also, their optimization was relatively flat, and if you have to back down to 1,750 psi for the RP stage I don't think it will make very much difference in their performance.

MR. SUDDRETH: If you use the Nova study to determine a national booster, and look at these things without weighting them technically, you will end up with a system which may not be what you might want to buy at that time. My point here is that the weighting or evaluation will be done by people maybe other than technical people, and you have to be realistic in the approach.

I have no objection to the 2,000 psi number, but somebody who might make the decision would say 2,000 is high enough; let's buy it.

MR. BARTZ: Of course, they have been talking 2,500 psi most strenuously. That seems to be the magic number.

MR. SUDDRETH: That selection might not buy us a brand new engine like a toroidal or multichamber engine. It might be another regular type of engine.

SESSION IV

IGNITION AND COMBUSTION

Chairman: Gerald Morrell
Lewis Research Center, NASA

16. IGNITION AND COMBUSTION

By Gerald Morrell

Lewis Research Center, NASA

It is evident from the preceding discussions that given enough boundary conditions and variables with which to work, a gaussian distribution of designs can be obtained, and there is no apparent way to select an optimum design from the results of uncontrolled studies. As requirements become more definite, we can expect the selection of an optimum design to become an easier task provided the research and advanced technology programs are providing the information needed to carry out and to evaluate proposed designs. Such technical information is also necessary for establishing realistic requirements in the first place.

In this session, I shall try to show how current theory can be used to assess the problems which may arise in the development of large-thrust rocket engines whether the large thrust is obtained by physical scale-up, by an increase of operating pressure, or by both methods together. I shall discuss ignition, combustion, and oscillatory combustion, starting with ignition which seems to offer the fewest problems.

Although the kinetics of the ignition process are not well understood, the classical approach of considering the thermal inputs and outputs up to the point where reaction is self-sustained permits an evaluation of the influence of changes in experimental variables on the parameter of interest whether it be ignition temperature, ignition lag, or minimum ignition energy.

The simplest expression for the rate of change of pressure in a combustor is given by:

$$\frac{dP}{dt} = \frac{kRT}{MV} - \frac{P}{t_c}$$

where k is the propellant flow rate $\frac{dm}{dt}$, R is the gas constant, T is temperature, V is volume of the combustor, M is molecular weight of the combustion products, and t_c is the first-order depressurization constant or blowdown time. If ignition takes place at $t = 0$, $P = P_o$, and no mass loss is assumed, then $k = \frac{m_o}{\tau}$, where m_o is the mass of propellant entered over the ignition lag period τ, and

$$P_o = \frac{k\tau RT}{MV}$$

At steady state, $t = \infty$, $\frac{dP}{dt} = 0$, and

$$P_\infty = \frac{kRTt_c}{MV}$$

With these boundary conditions, the rate equation may be integrated to give:

$$\frac{t}{t_c} = \ln \frac{1 - \frac{\tau}{t_c}}{1 - \frac{P_o}{P_\infty}}$$

At the ignition point, $t = 0$, and

$$\frac{\tau}{t_c} = \frac{P_o}{P_\infty}$$

This equation indicates that in scaling to larger thrust, a similar ignition pressure profile is obtained by maintaining a constant ratio of ignition lag to blowdown time. Also, to avoid overpressures, the ignition lag should be less than the blowdown time. For systems with considerable mass loss during the ignition period, as in low-contraction-ratio engines, at low ambient pressures, these conclusions need to be modified somewhat in that the initial pressure and pressure rise time will be lower unless the flow rate is increased to compensate for the losses.

There are many different ways to achieve ignition in rocket engines and most of them have been tried including spark ignition, hot surface ignition, and chemical ignition.

For laminar flow conditions, the minimum spark ignition energy is given by:

$$E_{ig} \propto \frac{U_g t_s}{P \log\left(\frac{2U_g t_s + s}{s}\right)}$$

where U_g is the approach gas velocity, t_s is the spark duration, P is pressure, and s is electrode spacing (assumed to be greater than the quenching distance). For turbulent flow at constant pressure, the analogous expression is

$$E_{ig} \propto \frac{U_g t_s \sqrt{\overline{u^2}}}{\log\left(\frac{2U_g t_s + s}{s}\right)}$$

where $\sqrt{\overline{u^2}}$ is the turbulence intensity. If the pressure level during the ignition transient is increased, therefore, lower ignition energies will be required. For the smaller volumes characteristic of high-pressure engines such a condition should exist. There is also evidence that in high-pressure combustors, the turbulence intensity is lower which would also tend to reduce ignition energy requirements. In scaling to larger sizes, care should be taken to maintain the gas velocity in the neighborhood of the spark source constant to maintain similarity.

For ignition by hot surfaces, one theory gives the following expression for the minimum igniter surface temperature:

$$\left(\frac{Nu}{d}\right)^2 \left(\frac{T_s - T_o}{T_o}\right)\left(\frac{T_f - T_o}{T_f - T_s}\right) \approx K \exp\left(-\frac{E}{RT_s}\right)$$

where Nu is Nusselt number, d is a characteristic dimension of the igniter, T_s is igniter surface temperature, T_o is approach gas temperature, T_f is flame temperature, E is activation energy for the chemical reaction, R is the gas constant, and K is a constant which depends on the properties of the propellant. When the appropriate expression for Nusselt number in terms of Reynolds number is substituted, it can be seen that the same conclusions with respect to effects of pressure and approach velocity on igniter surface temperature, and hence igniter energy, can be made as for the spark ignition case.

Hypergolic or chemical ignition has been studied extensively and sufficient understanding has been generated for our purpose. Although temperature has been found to be the primary environmental parameter affecting ignition lag, increasing pressure also produces a slow decrease in ignition lag. The latter effect is due, at least in part, to the improved mixing provided by increased pressure drop in flow systems. In a typical case, ignition lag is decreased from several milliseconds to several microseconds by an increase in pressure from 1 atmosphere to 1,000 atmospheres.

For scale-up of engine size, the most important observation is the apparent invariance of the ratio of ignition lag τ and combustor volume V_c for a given flow rate:

$$\log\left(\frac{\tau}{V_c}\right) = aT + b$$

where T is temperature, and a,b are constants characteristic of a particular propellant system. For nonflow systems at a given initial temperature:

$$\frac{\tau}{V_c} \propto \frac{1}{m}$$

where m is the mass of the reactants added. In either case, it is apparent that τ is most dependent on the local concentration of reactants which should be kept constant to maintain τ constant. In other words, initial flow rate should be increased in proportion to the increase in combustor volume in order to maintain a fixed value of ignition lag. For the case that the steady-state flow is increased correspondingly to produce the increased thrust, this procedure will assure a constant value of τ/t_c and, hence, of the ignition pressure.

One further factor which needs consideration is flame spreading from the locus of ignition. The rate of flame spreading will determine the rate of pressure rise after ignition and should be kept constant in order to maintain the same startup time in a larger engine. An increase in pressure should have no serious effect, since all available research results indicate that laminar flame speeds are proportional to pressure raised to a positive exponent, which varies from 0.1 for hydrocarbons-oxygen to 0.4 for hydrogen-oxygen. It should be necessary, therefore, in a scale-up only to maintain a fixed distance between ignition sources, that is, a constant number of igniter elements per unit cross-sectional area of combustor.

In summarizing up to this point, it appears that there are no serious ignition problems to be expected in the development of very large rocket engines where both scale and pressure may be increased over currently contemplated levels.

With respect to steady-state combustion efficiency most rocket design proposals are optimistic in spite of limited experience at very high and very low pressures. Our present state of knowledge does not warrant such optimism, in my opinion, especially in the high-pressure regime. There is no question, however, that with sufficient development effort a satisfactory level of combustion efficiency can be achieved consistent with stability and tolerable heat flux rates. It should be one of the objectives of the research and advanced technology program to provide the information which will help to minimize the development requirements. Another objective should be that of providing information which will help in evaluating the technical merits of alternative proposals.

In an earlier session, one contractor voiced the view, with which I concur, that we should give serious consideration to low-pressure systems in spite of high structural weights because of the inherently higher reliabilities which could be achieved. With this view in mind it seems appropriate to discuss several rate processes, since it does not appear that a single process can be rate-controlling over the range of pressures which should be considered. I shall discuss chemical space heating rates, turbulent mixing, and droplet vaporization.

Chemical space heating rate is dependent on the following parameters:

$$Q_{max} \propto \left(\frac{E}{RT_f}\right)\left(\frac{T_f - T_o}{T_f}\right)\left(\frac{C_p}{2\lambda_f}\right)\left(\frac{P}{RT_o}\right)U_f^{2}\Delta H_v$$

where E is the activation energy, R is the gas constant, T_f is flame temperature, T_o is reactant temperature, C_p is the average molar heat capacity of the reaction mixture over the range T_o to T_f, λ_f is the average thermal conductivity of the reaction mixture, P is pressure, U_f is laminar flame speed, ΔH_v is the volumetric heat of combustion of the reactants, and Q_{max} is the upper limit of the chemical space heating rate. From other research it is known that the pressure dependence of the laminar flame speed is given by

$$U_f \propto P^{0.114}$$

for methane-oxygen and

$$U_f \propto P^{0.43}$$

for hydrogen-oxygen, so that to first order, the pressure dependence of the space heating rate is given by

$$Q_{max} \propto P^{1.23}$$

for methane-oxygen, and

$$Q_{max} \propto P^{1.86}$$

for hydrogen-oxygen. It appears, then, that increasing pressure should increase the burning rate thereby increasing the combustion efficiency for a given length combustor. This phenomenon may account for the face burning observed in many high-pressure combustion experiments. As the following table shows, however, actual space heating rates fall far short of the maximum calculated value.

Propellant system	Pressure, atm	Space heating rate, Btu/hr-ft^3 (measured)	Maximum space heating rate, Btu/hr-ft^3 (calculated)
n-heptane—liquid O_2	18.4	7.1×10^8	4×10^{14}
g-H_2—liquid O_2	20.4	8.4×10^8	2×10^{15}
g-H_2—liquid F_2	25.2	8.5×10^8	7×10^{17}

The large discrepancy between measured and computed space heating rates has been attributed to the dominance of physical reactions in determining overall conversion rates. It is possible, of course, that for sufficiently low pressures chemical reaction may become the slow step in the conversion process, but it does not seem to be rate-controlling at pressures above 10 atmospheres or thereabout.

Another process which should be considered is turbulent mixing which, as will be shown later, may be especially important at very high combustion pressures. The results of a linearized analysis based on a grid of point sources is shown in figure 1, where the ratio of minimum to maximum concentration C_{min}/C_{max} is plotted as a function of the ratio of downstream distance to source or grid spacing x/s for various values of turbulence intensity $T = \sqrt{\overline{u^2}}/U$. The familiar growth curves are obtained with mixing length becoming smaller as turbulence intensity is increased or grid spacing is decreased. Typically, grid spacing might be the mean distance between injector elements or the distance between hydrogen and oxygen slots. The effects of finite source dimensions and jet impingement momentum have not been included in the analysis, but, qualitatively, it might be expected that the effect of the former would be to increase mixing length, while the effect of the latter would be to decrease mixing length. These effects should be the subject of additional research. The meager experimental data which have been obtained in small research combustors indicate that average turbulence intensity may vary from about 10 percent near the injector to about 5 percent near the nozzle entrance for the hydrogen-oxygen system.

It is apparent from figure 1 what must be done in scaling up at constant pressure in order to maintain the same mixing efficiency. It is necessary to retain the same grid spacing and gas velocity (contraction ratio). However, such a procedure will not retain similarity with respect to the acoustic mode of instability, and compromises will have to be made, for example, increasing combustor length.

The effect of increasing pressure is not so clear in this case. In general, as pressure is increased, it is necessary to decrease the effective number of injection elements in order to obtain the required mass flow in a reduced cross-sectional area. All the designs presented in preceding sessions showed this characteristic. This procedure has the effect of increasing the grid spacing and would be detrimental to mixing efficiency.

Turning now to vaporization as a rate process, we have had considerable success in recent years in relating combustion rate with the rate of vaporization of one or both propellants in the pressure range of about 10 atmospheres to 70 or 80 atmospheres. The final result of this work is the following equation for the percent of mass vaporized:

$$\text{Percent mass vaporized} = \frac{l_c}{A^{0.44}} + \frac{0.83 l_n}{A^{0.22} s^{0.33}} \; \frac{K(P/300)^{0.66}}{\left(1 - T_{l,o,R}\right)^{0.4} \left(\frac{r_m}{0.003}\right)^{1.45} \left(\frac{V_o}{1200}\right)^{0.75}}$$

where l_c is cylindrical chamber length, l_n is nozzle length, A is contraction ratio, s is nozzle shape factor, $P/300$ is a normalized pressure, $r_m/0.003$ is a normalized mass median drop radius, $V_o/1200$ is a normalized injection velocity, $T_{l,o,R}$ is the reduced liquid temperature, and K is a constant which depends on the propellant properties. The characteristic velocity efficiency is then given by the expression:

$$\eta_{C^*} = \frac{\left(C^*_{th}\right)_{O/F}}{\left(C^*_{th}\right)_{o/f}} \left(\frac{O\dot{w}_o + F\dot{w}_f}{\dot{w}_o + \dot{w}_f}\right)$$

where $\left(C_{th}^{*}\right)_{O/F}$ is theoretical characteristic velocity for the vaporized oxidant-fuel ratio, $\left(C_{th}^{*}\right)_{o/f}$ is theoretical characteristic velocity for the input oxidant-fuel ratio, $\dot{w}_o$ is the oxidant flow rate, $\dot{w}_f$ is the fuel flow rate, O is the fraction of oxidant vaporized, and F is the fraction of fuel vaporized. From these equations it is possible to estimate the effects of design changes on combustion efficiency. If pressure alone were increased, there should be a corresponding increase in combustion efficiency, but, as we have noted, an increased operating pressure usually means fewer injection elements per unit of propellant flow and much larger drop sizes. The net result could be a decrease in burning rate and a decrease in efficiency for a given length of combustor. Scale-up by multiplying the number of unit injector elements (i.e., maintaining r_m constant) is also hazardous since such a procedure is likely to increase the tendency towards oscillatory combustion, especially in the transverse modes.

At combustion pressures above the critical pressure of the reactants the vaporization theory runs into a serious difficulty. When the liquid temperature reaches the critical value, there is no longer a liquid boundary and some process other than vaporization must be invoked to describe the combustion rate. Turbulent mixing is probably the process which becomes dominant, but this combustion realm needs a good deal of research before definitive design criteria can be expected. An example of the magnitude of the difficulty is shown in figure 2 where the fraction of propellant vaporized when the critical temperature is reached is plotted as a function of the ratio of combustion pressure to critical pressure. It is apparent that for combustion pressures as low as 100 atmospheres (1,500 psia) only a small fraction of the mass will have vaporized when the critical point is reached. At this pressure and above, vaporization theory can only describe the initial heating period and is not applicable to the steady-state combustion regime.

Summarizing briefly, it can be said that chemical reaction rates are probably not a limiting factor in large engines. Vaporization theory is adequate for describing steady combustion up to about 70 to 80 atmospheres. Beyond this pressure limit, turbulent mixing probably becomes the dominant process, but this regime of combustion in the supercritical range requires much more research before definitive design criteria can be obtained. An increase in pressure, per se, should cause no difficulties except for the possibility of increased injector face burning caused by the flame stabilizing nearer the injector due to the increase in flame speed. The necessity for decreasing the effective number of injection elements per unit of mass flow as pressure is increased, however, may actually decrease combustion efficiency due to an increase in mass median drop size or grid spacing, depending on which mechanism is operative. This latter effect is already noticeable in the high-pressure hydrogen-oxygen engine work of Pratt & Whitney, and in the large thrust per element work of Aerojet. One way out of this dilemma is to make use of staged combustion cycles which may improve liquid atomization. Another promising approach is the toroidal concept of Rocketdyne which does not require such a large decrease in injection plane area as pressure is increased thus permitting the use of smaller injector unit elements. Much technology work remains to be done in these areas.

Scale-up with respect to thrust level only does not appear to be a serious problem as long as consideration is restricted to combustion efficiency. However, as I will show later, the similarity criteria for combustion rate are not compatible with those for combustion stability, so that the scale-up cannot be a straightforward procedure. Nowhere in the present proposals do I see any serious consideration of this difficulty. The combustion difficulties which have been encountered in the F-1 development give ample warning that the scale-up procedures used by prospective contractors in proposals for future large rocket engines should be given careful scrutiny.

Finally, I would like to discuss what I believe to be a crucial question with respect to the development and reliability of future large rocket engines. That is the question of combustion instability, which is a generic term for a variety of phenomena ranging from pure hydrodynamic modes of oscillation to pure acoustic modes. Only the hydrodynamic and acoustic modes have been studied extensively. A complication which is likely to arise in the very large engines of the future is the interaction of these two modes due to the fact that acoustic frequencies are approaching those of the hydrodynamic modes. Interactions of this kind have not been studied because of the great experimental and mathematical difficulties that are involved. The best that can be done for now is to treat the pure modes separately and to take care of the interactions during the hardware development phase.

The results of a linearized analysis of the hydrodynamic mode of instability (chugging) are presented in figure 3. In this figure, t_c is the critical time lag or combustion dead time, L^* is characteristic length of the combustor, ΔP is the feed system pressure drop, P_c is combustion pressure, and Z is the line relaxation time. The expression for Z is:

$$Z = \frac{l \rho_L V_L}{g P_c}$$

where l is line length, ρ_L is liquid density, V_L is liquid velocity, and g is the conversion constant. This analysis agrees quite well with much of the experimental work that has been reported and may be used to predict the stability of proposed systems. The critical time lag cannot be calculated precisely but may be approximated by calculating the ratio of the length required to vaporize half the mass and the injection velocity, or by calculating the ratio of the 50 percent mixing length and the average gas velocity, whichever mechanism is applicable. For reference purposes, the locus of current engines is shown in the figure. The difficulty in going to higher combustion pressures arises from the requirement to keep the pump discharge pressure as low as possible. This results in a low feed system pressure drop and, therefore, a low value of $L^* \frac{\Delta P}{P_c}$ leading to a less stable system. The situation is further aggravated if a portion of the feed system such as the injector dome or annulus is decoupled due to a sudden decrease in velocity. If the latter case is assumed, then most of the proposed high-pressure engines should be less stable than current engines. Nowhere in the present proposals do I see any serious consideration of this problem. Pressure budgets appear to be based on what the norm has been for each component rather than what is required from a dynamic analysis.

One positive factor should be mentioned. If hydrogen-oxygen rather than kerosene-oxygen is used for future engines it will be easier to achieve a stable system in the hydrodynamic mode. The much lower time constant for the former system means that it will be more stable than the latter system for a given value of Z and $L^* \frac{\Delta P}{P_c}$.

Finally, we should consider the acoustic mode of instability which continues to be a troublesome factor in new engine developments. After 6 to 10 years of really serious work we are beginning to shed some light on this phenomenon; there are several linearized theories which indicate qualitatively the influence of parameter changes on system stability, and there is one non-linearized theory which permits quantitative predictions of threshold perturbations which will destabilize the system. Much remains to be done. Good experimental tests of several theories have not yet been devised. Much more information is needed about the kinetics of the various rate processes in order to permit quantitative predictions of stability for a variety of engine design and operating conditions. Work in the advanced technology area between basic research and hardware development needs to be pursued much more vigorously especially over a wide range of operating pressures. Some concept of the problems associated with scaling combustors in order to maintain similarity with respect to acoustic mode stability can be gained from a brief review of the major results of each theory.

Penner has applied Damkohler's analysis for flow reactors to the case of a rocket combustor and has found three similarity groups which must be satisifed to maintain a stable system:

Reynolds number,

$$Re \equiv \frac{\rho U D_c}{\mu}$$

Damkohler's Group III,

$$D_{III} \equiv \frac{QD_c}{U} C_p T_c t_i = \frac{QD_c \dot{w}}{\rho Y_i U C_p T_c}$$

Chemical Conversion Group,

$$\chi \equiv \frac{t_i}{t_w} = \frac{\rho Y_i}{\dot{w} t_w}$$

where ρ is gas density, U is gas velocity, D_c is combustor diameter, μ is viscosity, Q is heat of reaction, C_p is specific heat, T_c is combustion temperature, Y_i is the mass fraction of reactant i, $\dot{w}$ is the mass addition rate per unit volume due to chemical reaction, t_w is the wave time, t_i is the chemical conversion time of reactant i. From this analysis it is apparent that as combustor size is increased there must be an increase in chemical conversion time to maintain a constant X value; however, it would be difficult to maintain the other two groups constant while changing only t_i. It is also interesting to note that an increase in t_i will not maintain similarity with respect to combustion rate.

Crocco and coworkers have obtained a solution of the linearized conservation equations by assuming that the response of the combustion process to a perturbation is characterized by a sensitive time lag and a pressure interaction index. The total time lag is given by:

$$\tau = \tau_i + \tau_s$$

where τ_i is the insensitive portion of the time lag and τ_s is the pressure sensitive portion of the time lag. The sensitive time lag is given by:

$$\tau_s = \bar{\tau}_s - n \int_{t-\tau_s}^{t} \frac{\Delta P}{P_o} \, dt$$

where $\bar{\tau}_s$ is the average value of τ_s, n is the interaction index, P_o is the average pressure, ΔP is the pressure perturbation, and t is time. The perturbation of the burning rate is given by:

$$\Delta\dot{w} = n\left(1 - e^{-n\tau_s}\right)\frac{\Delta P}{P_o} \rho \frac{dU}{dz}$$

where ρ is gas density, U is velocity, and z is axial distance. Instability should occur for values of $\bar{\tau}_s/t_w$ between the following limits:

$$\left[\frac{1}{2} - \frac{1}{\pi} \sin^{-1}\left(\frac{\gamma + 1}{2\gamma_n} - 1\right)\right] < \frac{\bar{\tau}_s}{t_w} < \left[\frac{3}{2} - \frac{1}{\pi} \sin^{-1}\left(\frac{\gamma + 1}{2\gamma_n} - 1\right)\right]$$

where γ is the ratio of specific heats and t_w is the wave time. The ratio $\bar{\tau}_s/t_w$ is similar to Penner's ratio t_i/t_w, and would yield about the same conclusion with respect to scaling. It is not possible to calculate an a priori value of $\bar{\tau}_s$ or n, so that a direct test of the theory has not been attempted. Crocco has measured stability limits for rocket engines and has obtained values of $\bar{\tau}_s$ and n from one limit of the boundary curve. Using these values, he has computed the second limit in close agreement with the observed value. It appears, then, that Crocco's model should be useful for correlating experimental instability data and might be useful for indicating changes in design parameters which would improve stability. It is not applicable to systems requiring a finite disturbance to produce reasonance (nonlinear systems).

Priem has obtained numerical solutions of the nonlinear conservation equations by employing a specific equation for reaction rate and, in particular, the vaporization equation derived for steady-state combustion. The results for a one-dimensional model and the transverse acoustic mode are shown in figure 4. In this figure, the pressure amplitude $\Delta P/P_o$ is plotted as a function of the parameter $L \equiv D_c m/2A$ for several values of $\Delta V/a_c$ where P is the peak-to-peak amplitude of the wave, P_o is average pressure, D_c is combustor diameter, m is fraction of propellant vaporized per unit length, A is contraction ratio, ΔV is velocity difference between liquid and gas phase, and a_c is the velocity of sound in the combustion gas. The lower limit curves are the perturbation amplitudes required to destabilize the system and the upper bands represent the limits for the equilibrium amplitudes of the oscillation. For reference purposes most current engines have L values corresponding to the minimum region of the perturbation curves. Since not all of these engines display equal levels of instability, the differences must be attributed to effects of viscous damping, which was not included in the analysis, or to variation in the value of $\Delta V/a$ for the engines. As is seen from the figure, increasing $\Delta V/a$ stabilizes the system. This factor may account for the increase in hydrogen-oxygen system stability which has been observed as hydrogen temperature is increased. It would also indicate that the coaxial form of injector should provide a relatively stable system as compared with the impinging jet form of injector, for a system where one propellant is injected essentially as a gas.

Concerning scale-up in size, the theory requires a decrease in combustion rate m as size is increased, in order to maintain a constant value of L; it also indicates that for any size combustor, stability is improved by decreasing the combustion rate (considering only the region to the left of the minimum). Both conclusions are consistent with the results of the linearized models. So we see that all theories are in agreement in one respect: scale-up procedures which maintain similarity with respect to combustion rate are likely to produce a system which is less stable than the model. Such procedures, however, are exactly the ones which have been used in the past to scale from existing hardware to larger systems. It will be necessary to take a much different approach to the engines of the future.

So far, I have restricted the discussion to the region left of the minimum. The region to the right, however, represents the most startling result of the nonlinear solution. It predicts a trend which is just the opposite of that obtained from the linear models. This behavior has not yet been tested experimentally although we are attempting to do so at Lewis. If it should be true, it offers the possibility of obtaining absolutely stable combustion systems. The probability is rather high, however, that the complete three-dimensional solution will yield a curve which is considerably flattened in this region. Preliminary two-dimensional calculations already indicate such a trend. For the time being, therefore, it would be safer to restrict attention to the left-hand range of L values for scaling to larger thrust levels.

The effect of increasing pressure on stability is not clear-cut. Restricting attention to the left region, a pressure increase alone would increase m and would tend to destabilize the system. If, as is usually the case, however, the number of injector elements per unit of flow is decreased, the effect will be to decrease m and to stabilize the system. So the effect of pressure cannot be assessed without considering a particular design.

It may also be useful to consider the acoustic mode as a heterogeneous detonation supported by the atomization of the liquid phase behind the wave front. Penner's chemical criterion may then be written:

$$\chi = \frac{t_b}{t_w}$$

where t_b is the breakup time of the liquid phase. I have been studying the rate of breakup of liquid jets by shock waves and the theoretical model when evaluated at the conditions behind a stoichiometric hydrogen-oxygen detonation predicts breakup times of the order of magnitude required to drive such a detonation, that is, the order of 10 microseconds. For purposes of this discussion the breakup time may be written:

$$t_b \propto \frac{R_o^{1.25}}{U^{1.25}\rho^{0.42}}$$

and

$$\chi \propto \frac{R_o^{1.25}}{U^{1.25}\rho^{0.42}t_w}$$

where R_o is jet radius, U is particle velocity behind the wave, and ρ is gas density. The latter expression implies that for scale-up of thrust or pressure, similarity with respect to stability is maintained by increasing R_o; that is, the combustion rate must be decreased or the number of injector elements per unit of flow must be decreased. Again the conclusions are qualitatively similar to those of other theories. What is required, obviously, is much more quantitative information both at the research level and at the advanced technology level.

In summary, it appears that no serious ignition problems can be expected in the development of future large rocket engines.

Available theory is adequate to predict combustion rate and efficiency up to about 80 atmospheres, but for pressures above this value the nature of the rate-controlling process is not known, but is most probably turbulent mixing.

There may be some difficulty in future large engines with hydrodynamic instability if high combustion pressures are employed together with marginal line pressure drops.

Finally, several theories are available for predicting design variations which will maintain similarity with respect to acoustic mode instability. All agree that the proper scaling procedures are not compatible with those required to maintain combustion efficiency constant. None of the theories has had adequate experimental tests.

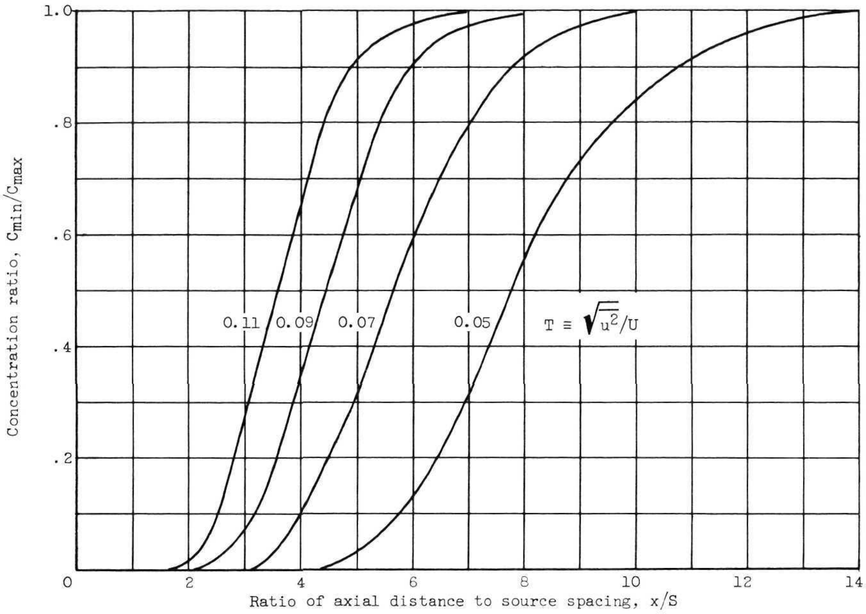

Figure 1.- Turbulent mixing from a grid of point sources.

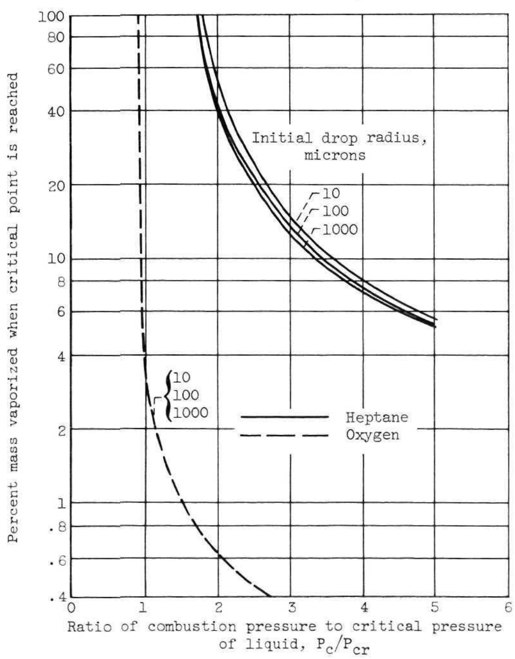

Figure 2.- Liquid vaporization at combustion pressures above the critical value.

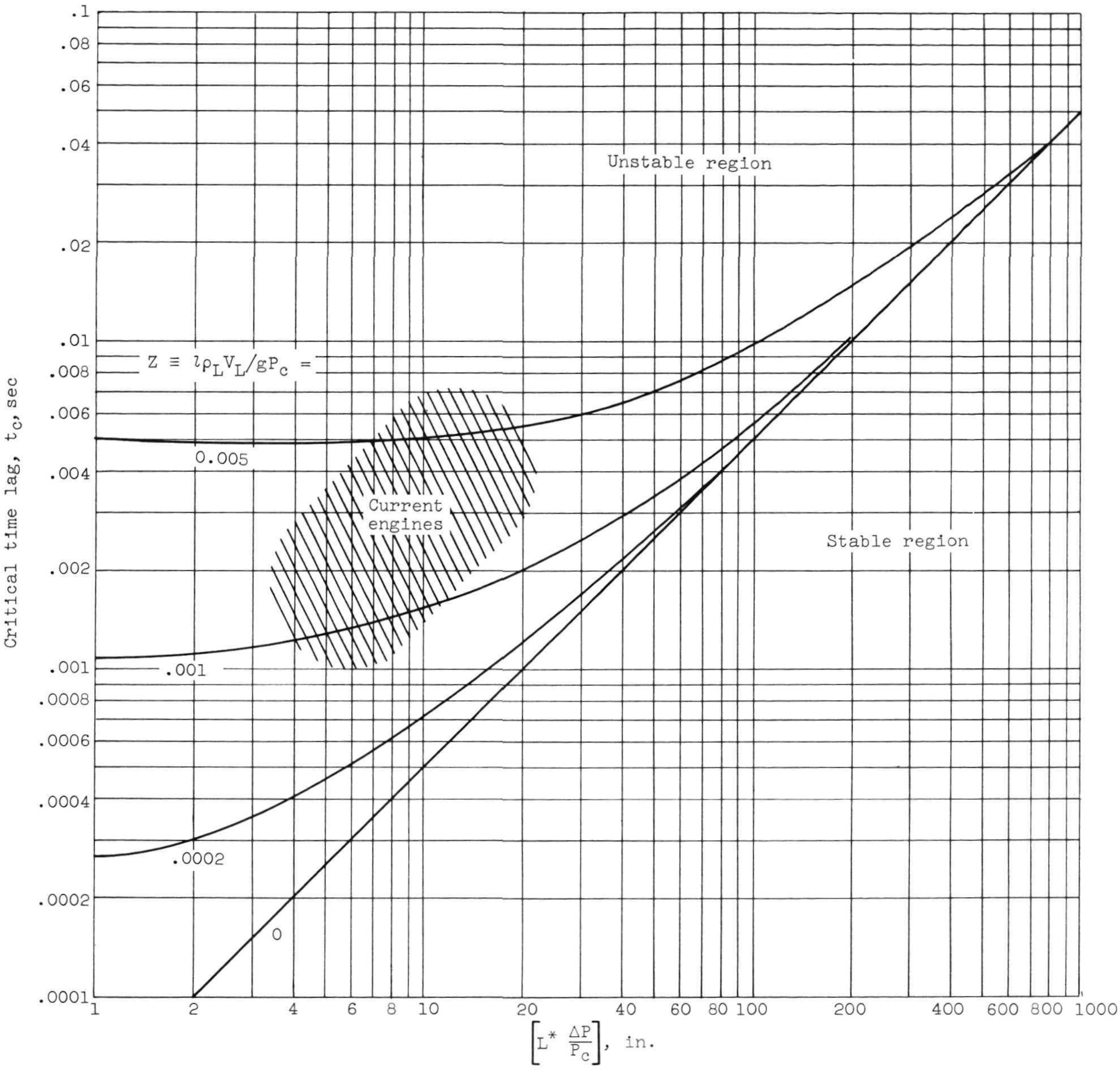

Figure 3.- Critical time lags for hydrodynamic instability (chugging).

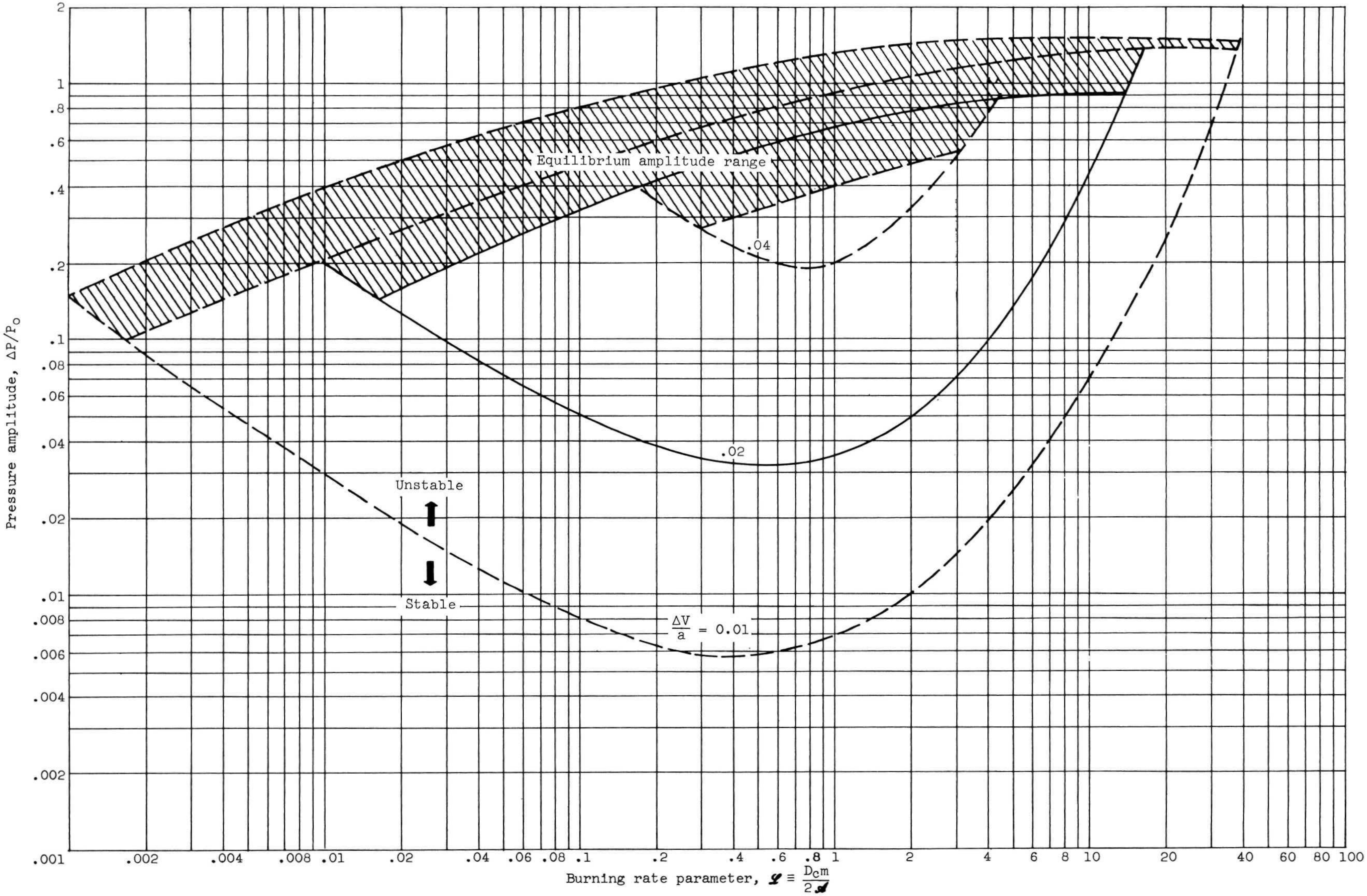

Figure 4.- Stability limits based on nonlinear conservation equations and vaporization model.

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17. SUMMARY DISCUSSION

MR. JARAMILLO: I would like to know whether you favor going to higher chamber pressures or lower chamber pressures for the very large high thrust engines.

MR. MORRELL: How can I say? We first have to decide what it is we want to do, really. And what the conditions are for what you want to do. If you tell me, for example, that you don't care what the thing looks like, just so it works, how big it is or what, just so we take the least amount of risk, obviously I would pick the very simplest system I could think of. Maybe to me that is a pressurized low-pressure system. I would say that, of all the things I could think of, this has probably the least amount of trouble with it.

MR. BARTZ: Then it boils down to the question of single-stage-to-orbit or not. That seems to be the recurring open question.

MR. MORRELL: What is so magic about that?

MR. BARTZ: There you need the impulse that you can get from high area ratios.

MR. MORRELL: The whole selling point on single-stage-to-orbit, other than some minor one about reliability, is that it is a lot easier to recover everything and maintain it, and the engineering is a lot easier if you are going to reuse it.

MR. BARTZ: Then you question the importance of it?

MR. MORRELL: I would question it for the very limited objectives NASA has. I don't see how you can launch more than 50 a year of these beasts. I don't know. Maybe that is a lot. Maybe 20 a year. We are not going to produce them like automobiles, like Volkswagons in this case or even Ford cars. You want at most 100 of these things, therefore why break your back.

MR. BARTZ: I agree.

MR. MORRELL: I am all in favor of advanced technology because it helps fatten up the budget, among other things, and it is good for the economy, and who knows, someday we may want to do more than my limited viewpoint right now sees. That is the place where advanced technology comes in. I think we tend to get confused between what it takes for success and what it takes for esthetics, so to speak.

You have a certain job to do and there are a number of ways to do it, and you can do it the simplest way, although it may also be the clumsiest, but simple enough so that it gets done. You don't have to worry about appearances or using the very latest things, but about those things which you know the most about. And you can sledge-hammer your way through the problem; or you can make the argument that this technology will help me some day later, somewhere; or it may turn out that we will decide it is worthwhile doing. I am not against learning new things and learning new ways of doing things, but I think we ought to keep that in perspective with what you have to do, and, in my opinion, you don't put new things in simply because they are new. You put them in when you are sure they will work, because if we don't succeed, nobody will ever ask you about all the knowledge that you gained, and if you do succeed they don't really care how you did it.

MR. BARTZ: I think the point to be emphasized, though, is the fact that what you do, in answer to a question "Do you go high pressure or not?", is strictly dependent on the question of do you go single-stage-to-orbit. Is that important? I think that has been skirted.

MR. MORRELL: This is not a job for a contractor.

MR. BARTZ: It has not been answered.

MR. MORRELL: That is our responsibility. Somebody has to get together within NASA and decide what they want to do.

MR. SLOOP: Hasn't another argument on pressure to do with vehicle design? You talk to a structures man and he doesn't like to see big things. He likes small things for reducing bending moments and things of that sort.

COMMENT: We agree that we are not going to be building transportation systems by 1970. But we say don't build more and more Peenemünde engines and try to amortize them.

MR. MORRELL: They work.

COMMENT: You don't keep going the same route. You try to do it a little better.

MR. MORRELL: There is nothing wrong with building something better if you can, and if you are sure that it really is better.

COMMENT: The way you did it at one time was because that was the only way you could do it. Right?

MR. MORRELL: Right now there is only one way I know to do it. You keep learning.

COMMENT: And the question is, what direction do you go to learn? You ask yourself the question, "If I had it, what would I do with it?"

MR. MORRELL: If you are just trying to learn basic information, you learn everything.

COMMENT: That is pure research.

MR. MORRELL: Somewhere as you go along you have to make a decision as to which of this you are going to discard.

COMMENT: The decision might be based on the question, "If we had this would we use it?"

MR. MORRELL: We know that by 1970 we won't have the vaguest idea of using air augmentation.

COMMENT: That will never work anyway.

MR. MORRELL: So throw it out right now. Forget about it. You are just cluttering up the works.

COMMENT: I agree. We have known that for 50 years.

MR. WEIDNER: Let me say one thing about the one-stage-to-orbit. I believe we are searching here, soul-searching, and we don't know whether it can be done. Very likely it cannot be done and we all agree with that. This doesn't mean we shouldn't at least look and make up our minds as to whether we are certain or uncertain about it.

Once we reach this point and we come to a real development thing, I think then the real soul-searching has to start since I believe that what we are about to do here, at least what we think we are about to do, would be so costly that we cannot afford to be wrong. As soon as that question comes up, I think we will all come much closer to the things we know, and the advancement will not be quite as advanced as it appears today. But I think this is all obvious to the people who, let's say, have been in this business for a long time. If we go into this undertaking with this in mind, we will never make progress and advancement. I think we should use this time right now to soul-search and somehow to think of things which might not be possible, but at least to look at them.

All we are saying here is, let's use this time to find out whether there are some areas where with safety we can make a small step ahead by the time we are ready to make that choice, or called upon to make that choice. I think that is all we are really looking for. This is at least my philosophy. I don't see these very big things myself, and I think nobody does. Not in our lifetime. But I think this shouldn't keep us from looking.

MR. SLOOP: Absolutely.

MR. BARTZ: The contractors are expecting a lot more though. They are proposing million-dollar programs.

MR. SLOOP: I think it is natural that the contractors propose million-dollar programs. I think they feel that the Nova is almost around the corner. I think our studies sometimes indicate this. But I think that we do have some time to do advanced technology, and I think it is very good that we do have this time and we can, as Mr. Weidner said, look at a number of possibilities here and bring ourselves as far along as we possibly can before we make the decision.

MR. MORRELL: There is one other thing we have to do, and that is we can't simply sit here and say that all we are after is information; certainly we are after that, but I think there was some truth in what was said here, that if you want something by 1975, or 1978, or whenever, you really cannot sit back for 5 years and accumulate information. You are going to have to start right now trying to decide what it is that you would do if you had to do it now; somehow we have got to solve this very difficult management problem of keeping that within bounds so that we can make improvements in it as we go along. We have to keep the program from freezing so that we are limited by it.

If we can keep it as a flexible program so that we can feed technology into it as it develops, and don't make the real final decisions until 1970, but hopefully have enough technology so that we don't have to have an 8-year lead time by then, we will have really accomplished something in this generation. But if we simply go about aimlessly accumulating information in the form of technology, we still won't have the information we need when we have to make the decisions because we will always find we have done it in the wrong pressure regime or the wrong velocity regime or the wrong propellants or something.

You have to do both things. There has to be a certain amount of flexibility in the technology program so that we don't put blinders on and only look at certain things. At the same time, those of you who have hardware responsibilities directly can't wait until 1970 to start thinking about what you are going to do, and if you start thinking now, as you say, you can't think of all the strange possibilities. You somehow have to dream up a flexible development program which will allow for the input of new developments until the very last minute, and that is something that has never been done before. As far as I know the development program usually gets frozen before you are ready.

MR. WEIDNER: That is true. We can't charge in all directions even in technology. We ought to get direction to this whole thing; on the other hand, if we think of what happened prior to the selection of the Saturn V mode and how to get to the Moon, there were battles in all factions. I am only thinking of the pressure of a national commitment.

Not having this power, this directing and guiding force here, how will we ever be able to make up our minds how to go?

MR. MORRELL: For a change we might get ahead of the game on propulsion. Usually the decision, no matter whether political or nonpolitical, or so-called technical decision, ultimately depends on how big an engine do I have and how many of them can I put together. If by some chance we should some day get ahead of the game so that we have something bigger than what we need, then that will be the limiting factor and that will determine what can be done. And so I think if I were starting out today, I wouldn't stop with the Mars mission. I would build an engine that was bigger than I needed for the Mars mission and I would start out thinking in terms of very simple concepts and doing the technology to check out those ideas as far as they go; then I would bring along this more advanced technology at an appropriate pace and keep that development program from freezing, because I don't need it now anyway.

MR. BARFIELD: I think you are going to have to find some way to bring things into focus, selecting certain areas of technology and going ahead and pursuing them and getting some of the design data on which you can base an engine development in the future.

MAJOR BOUVIER: I don't have anything to add except that in the Air Force we are trying to promote some projects in advance technology to fulfill the objective that Mr Weidner mentioned, and I do hope that you recognize some of the things that we are doing in planning your program. We have some high hopes for an advanced technology program which features the stage combustion, the high chamber pressure with storables, and self-compensating nozzles. That hasn't been approved in a hard fashion yet, but I think that we are in the right direction. I was struck with one thing, based on my own personal experience, that recovery and this very high payload, or high ΔV that you are talking about, just don't seem to be compatible. It goes back to this comment a little earlier about the number of missions one might expect to do in the coming years. If you have to put a lot of weight into orbit, maybe you want a smaller vehicle and do it more often. Then recovery would pay.

MR. THOMPSON: One thing that seemed to be hit pretty hard by several people was the system concept, which I feel is essential; that is, not looking at just one chunk but a system and the intereffects. Following logically from there one could say that certain representative systems could be picked and the technology required to look into these systems followed through on as one approach to a direction.

COMMENT: That is the point I was trying to make. The only way you will determine what you want is by system. Otherwise you will put more and more effort into all the bits and pieces and eventually end up with all shelves full of knowledge, very little of which will be useful to you. You have to decide which of the items you want to use.

COMMENT: You had better have a design study to get the selection process.

[redacted]

SESSION V

PUMPING AND FLOW SYSTEMS

Chairman: Ambrose Ginsburg
Lewis Research Center, NASA

[redacted]

18. INTRODUCTION

By Ambrose Ginsburg

Lewis Research Center, NASA

This session will concern turbopumps for the large-thrust engine. We believe that large thrust will mean large turbopumps for a reasonable modular arrangement. That brings up the question: What is a reasonable modular arrangement?

It is my personal belief that an arrangement of 30 pumps strung around like lights on a Christmas tree is not reasonable, and one pump is not reasonable. So for the purpose of this analysis we have picked a turbopump size for a 6-million thrust engine.

We are using hydrogen and oxygen as the propellants. We considered that high pressure was desirable, and we had to "pick a number" to make a design analysis. We picked 3,000 psi for the chamber pressure and an O/F of 7.

In this session, we will take a look at the turbopump problems that are involved in trying to satisfy an extreme situation of an old rocket engine problem. The severe situation is that we have a large high-pressure turbopump with severe cavitation at the pump inlet and extremely high-pressure loading at the pump exit.

This presentation is really in two parts. Part 1 covers some work by industry and comprises two papers. The first paper, which is mine, concerns the high-pressure pump experiments at Pratt & Whitney in Florida. The second paper, by Loren Gross of Marshall Space Flight Center, presents the results of the high-pressure turbopump studies at Rocketdyne.

Part 2 of this session covers an analysis of large high-pressure turbopumps by Lewis Research Center personnel. There the first paper, by Mel Hartman, head of the Pump Research Section, discusses pumping. The second paper, by Warner Stewart, head of the Turbodrive Section, discusses turbine drive systems and cycles. The final paper, by Herbert Scibbe, member of the Bearing Research Section, concerns bearings and seals.

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19. HIGH-PRESSURE PUMPING EXPERIMENTS

By Ambrose Ginsburg

Lewis Research Center, NASA

The Pratt & Whitney program was conducted under an Air Force Contract with the purpose of developing a pumping system to supply fluid for a high-pressure thrust chamber research program. The Pratt & Whitney turbopump experiments were really feasibility studies of high-pressure hydrogen and oxygen pumping. The turbopump program at Pratt & Whitney is practically concluded. It is very unfortunate that this program has not been extended, since interesting and successful results were achieved as far as this test program went.

The objective of this program was to develop a 5,000-psi hydrogen and oxygen pump system, with severe cavitation capability not a requirement. Also, the pumps were to be on separate shafts.

Table I shows the hydrogen-pump design point specifications. It is a two-stage centrifugal pump, back to back. The impeller blades are swept to 30°, which is somewhat unusual for a hydrogen pump. Also, when I first saw this pump I thought the tip speeds were quite unusual. The design tip speed of 2,200 feet per second is way up there in terms of the present state of the art.

There are other design features that might be of interest. The bearings of the system have a DN of 2 million; the cage material is a porous bronze coated with Teflon. The bearings are an angular groove type and are hydrogen lubricated. The impeller material for this high tip speed, 2,200 feet per second, is a titanium alloy, designated A-1-10 alloy with low oxygen content. The housings are aluminum; the turbine is titanium.

The thrust was balanced by a combination of back vanes on both the pumps which were back to back, a balance piston, and of course the angular grooved bearings.

Figure 1 shows a pump assembly; one portion is enlarged at the lower left of the figure, for greater clarity. A one-stage turbine is shown.

Note the passage shape of the pump, which is somewhat typical of the low specific speed of the design. The very small inlet is certainly not good for cavitation. The fairly small blade heights helped alleviate the stress problem at the high tip speeds. It is certainly not a flight-weight design, as can be seen by the heft of the shroud and casing structures.

Figure 2 shows the performance of this hydrogen pump. The head rise in feet is plotted against flow. The solid lines represent theoretical efficiency contours and speeds. The data points represent the actual data taken on this pump.

The data achieved are near optimum efficiency over the speed range; they represent the high-pressure data.

Test data were not obtained above 40,000 rpm because of bearing and piston problems. However, 4,000 psi at 2,000 gpm was achieved at this speed.

The oxygen pump is presently under test. There are a couple of characteristics that might be of interest. The bearings for the lox pump are oxygen lubricated. The drive turbine bearing on the shaft is either hydrogen or nitrogen lubricated. It uses RL-10 type carbon seals that have to function at surface speeds of up to 225 feet per second. It is a single-stage pump, not very different in appearance from the hydrogen pump.

The performance of the oxygen pump is shown in figure 3. Pressure is plotted against flow, shown with dashed theoretical speed lines. The pump was operated at about 10,000 rpm, represented by the solid line approximately parallel to the dashed lines, with data points covering the flow range on the lox pump. Then a speed ramp from 10,000 to 19,000 rpm was made and achieved 4,000 psi and a bucketful of parts.

This speed ramp was really supposed to be at non-stalled flow, but owing to faulty instrumentation it was in the stall zone. As soon as a speed of about 19,000 rpm was reached, a pump vibration probably occurred with a resulting pump failure. The failure was probably due to rubbing between the impeller and the casing.

This is an important point: in high-pressure machinery, stall flow and pump vibrations are problems.

In their experiment, Pratt & Whitney achieved 4,000 psi, with reasonable mechanical success. The program proved that methods for calculating back vane flow were not accurate enough for thrust calculations. They did not obtain any off-design data in terms of head and flow. They obtained no cavitation performance or effects of cavitation. They did not establish the stall line. And, certainly, they did not obtain any stability data. The program certainly pointed up what everybody has been expecting: problems in bearing loads and balanced pistons.

In going through the mechanical design procedure for such a program Pratt & Whitney did come up with other factors which they believe, and I agree, are quite important.

They believe that static seals for large high-pressure machinery are going to be an exceedingly difficult problem. They believe that work really should be done on flange design for static seals. Critical speed is a very touchy problem.

I recommend that this program be supported to complete the experiments, and this completion would include obtaining a complete performance map of both pumps, pump stability data over the range of flow and head, and also cavitation performance.

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TABLE I. PUMP DESIGN POINT SPECIFICATIONS

Flow – gpm	2,400
Pressure Rise – psi	5,000
Efficiency – percent	58
Speed – rpm	45,000
SSS (Inlet $P_o = 40$ psia)	14,300
Specific Speed	458
No. of Stages	2
Impeller Type	Backswept to 30°
Impeller Tip Speed – ft/sec	2,200
Impeller Diameter – in.	11.2
Horsepower	12,200

HIGH PRESSURE PUMP ASSEMBLY

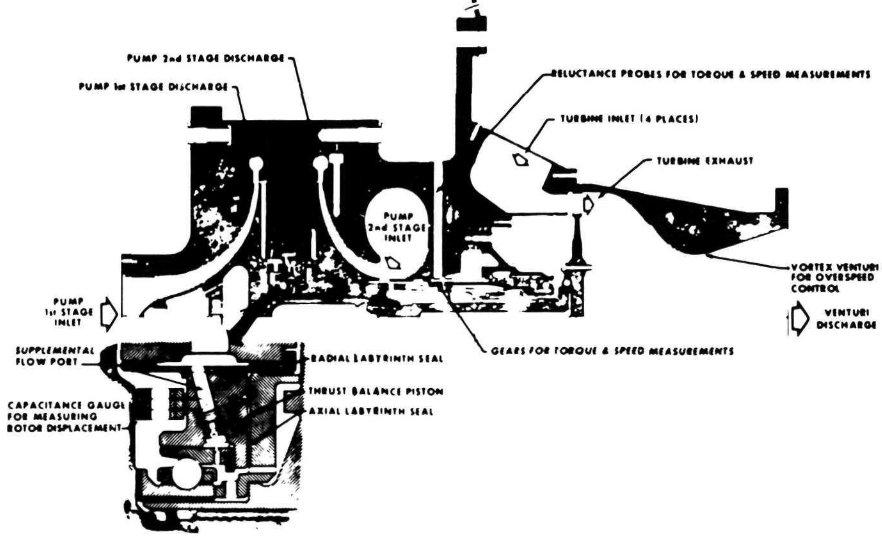

Figure 1

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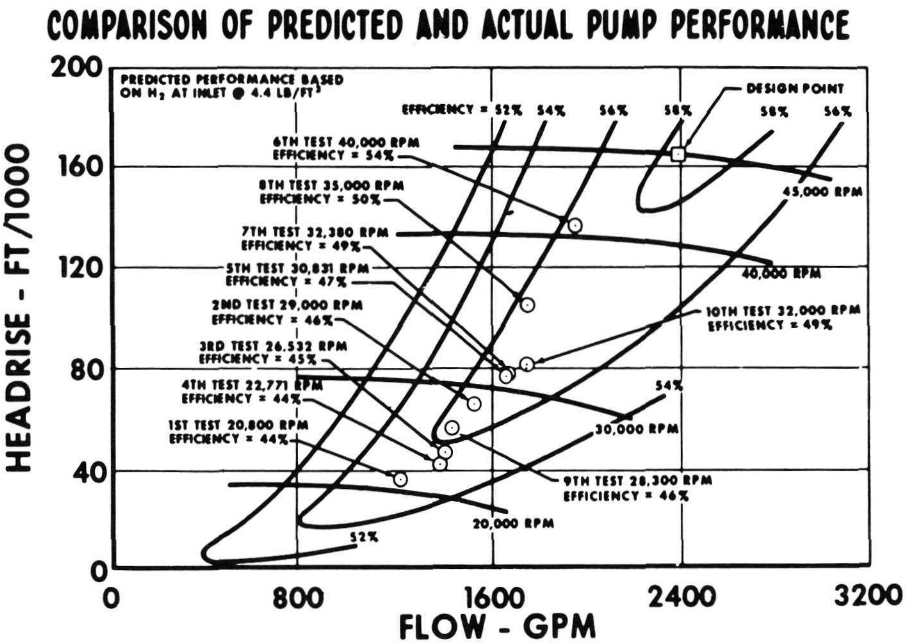

Figure 2

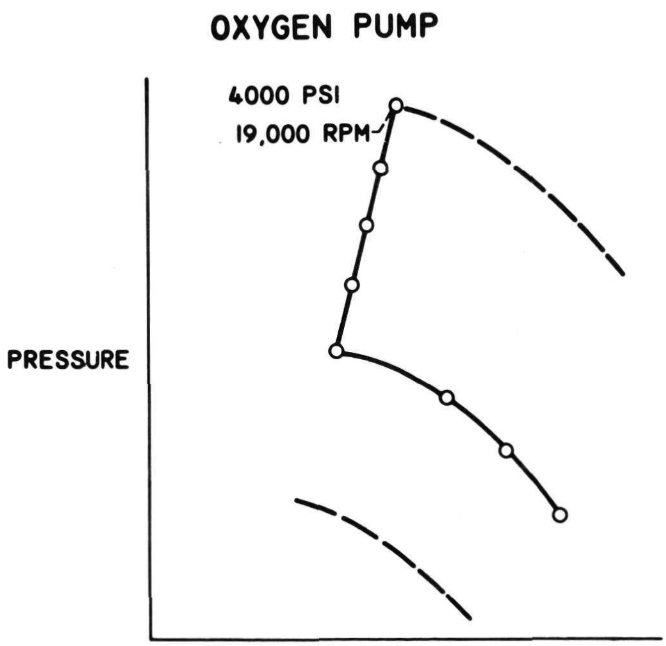

Figure 3

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20. HIGH-PRESSURE TURBOPUMP INVESTIGATIONS

By Loren Gross

Marshall Space Flight Center, NASA

This paper is a discussion of the high-pressure pumping program that Marshall Space Flight Center has funded at the Rocketdyne Division of North American Aviation Corporation for $300,000 in each fiscal year 1963 and 1964.

The first objective of this program was to define problems of high-pressure pumping systems. The ground rules used for the study were pumping systems for engines with chamber pressures from 1,000 to 5,000 psi and with thrust levels from 1 to 6 million pounds. The second objective was to initiate technology to solve problems as they were found. Both an analytical program and an experimental program were conducted.

In the analytical program the first area investigated was the definition of turbopump operating requirements for the large high-pressure engines. During this part of the program, work was closely coordinated with the high-temperature rocket-engine cooling which W. R. Wagner of Rocketdyne described in a previous paper. They used data from this study to determine pump discharge and flow-rate requirements for the very high-pressure high-flow-rate engines. Second, these data were used to analyze and lay out turbopump assemblies to fit a wide range of flows and head rises. The third effort was to extrapolate propellant properties to very high pressures. The propellants considered were liquid hydrogen and liquid oxygen. The final analytical work was on a cavitation scaling concept in which an acoustic theory of head breakdown was investigated.

The experimental program consisted first of an impeller burst investigation. In this phase, a radial-vaned shrouded impeller was procured, analyzed, and run to burst in air. Second, high-speed antifriction bearings were tested in liquid hydrogen. Third, an axial-flow impulse pump stage was designed and tested in liquid hydrogen.

In the definition of turbopump operating requirements, the effects of engine configuration were considered; that is, whether topping, GG, or other cycles were used. In the later detailed studies made during the past year, the GG cycle was considered almost exclusively. The type of combustion-chamber cooling, the effect of combustion-chamber material, and the effect of valve and line pressure drops were evaluated to obtain pump discharge pressures and flow rates.

The result of these studies was a series of graphs, an example of which is shown in figure 1. Here we have pump discharge pressure for a given cycle, in this case liquid-hydrogen pump discharge pressure, as a function of the engine combustion-chamber pressure. Note the continuous curve for estimated pump discharge pressures with combined regenerative and film cooling and the transition from complete regenerative cooling to the combined system. A series of these plots were prepared under the program for liquid oxygen-RP engines and liquid hydrogen-liquid oxygen engines.

The data from the study of turbopump operating requirements were used to design and lay out turbopump assemblies for the engine size range of interest. The purpose of this design and layout was to graphically find problem areas of large machinery. The turbopumps laid out were all for GG cycle systems. I might add that the liquid oxygen-RP turbopump systems were all single shaft systems, and except for the 5,000-psi-chamber-pressure machine were all single-stage pumps. The liquid oxygen-liquid hydrogen designs were dual-pump configurations.

These layouts were analyzed to determine engineering feasibility and problem areas that could be expected to be encountered from manufacturing, material, and testing considerations. Present-day state of the art was used in laying down these designs; that is, our present criteria for seal velocities, suction performance, and bearings were used.

From these layouts were obtained a series of representative design points over the range of thrust and chamber pressure of interest. The design points are graphically shown in table I. Liquid oxygen RP-1 engines were laid out at each point indicated by RP and a liquid oxygen-liquid hydrogen engine was laid out at each point indicated by LH_2.

TABLE I.- TURBOPUMP LAYOUT POINTS

Chamber pressure	Thrust level	
	1,500,000	6,000,000
1,000	RP(F-1) LH_2	RP LH_2
2,000	RP LH_2	RP LH_2
3,000	RP LH_2	RP LH_2
4,000	LH_2	LH_2
5,000		RP

The outcome of these studies was a series of analytical graphs in which critical turbopump parameters were plotted against chamber pressure and engine thrust. Figure 2 is an example in which turbopump weight was plotted against engine thrust for various values of chamber pressure, P_c. This was done for a number of other parameters, such as seal velocities and bearing speeds.

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The conclusions of this study were as follows:

1. All the turbopumps examined were within present engineering and component technologies. That is to say, if it were required to start a design today for one of the engines in the range of interest, a turbopump design could be laid down with some confidence. This is not to imply that such a design would be simple and easy. It was just North American's belief that machines could be designed and built.

2. There was a great deal of state-of-art and component technology desirable for overall design improvements. A number of these state-of-art and component technology programs desired were included in the follow-on program. This state-of-art work would include programs with high-speed seals, high-speed bearings, and high-head pumps.

There are two experimental parts of this program of particular interest: the axial-flow impulse pump work and the high-speed bearing work.

The axial-flow impulse pump work is an attempt to get high head rise from a small package by using impulse techniques in the staging. The impulse stage was designed for a head coefficient of 1.3 at a flow coefficient of 0.735 with a design efficiency of 75 percent. The stage was designed around the Mark 9 hardware. This is the nuclear feed system hardware. The stage consists of a single rotor and a three-stage stator. Liquid hydrogen was the test fluid.

The advantages of such a stage are that, hopefully, it would give a very high head rise in a single stage. The stage would also be expected to have essentially no axial shaft thrust. The biggest disadvantage of such a stage is the extremely high net positive suction head requirements for the stage. The net positive suction head requirements for the stage approach the total head rise of the stage. This would limit applicability to the final stages of high-pressure pumps.

Figure 3 shows the rotor used in the experiments, and figure 4 shows a partial assembly of the rotor and a section of the three-stage stator system.

The machine was tested and performed with a head coefficient of 0.9 at a flow coefficient of 0.5, with an efficiency of about 75 percent. The machine was reasonably stable over a wide operating range. Note that there was quite a discrepancy between the predicted and test performance for the machine.

The characteristic curves for the machine are shown in figure 5. The extremely large stall hysteresis loop should be noted.

The performance discrepancy was found to have been caused by a blade-setting error in the manufacture of the rotor. The rotor used did not fit any particular design criteria. A revised performance analysis was made on this rotor, and it seemed to fit the test data reasonably well. I consider the concept quite promising.

Under the second part of the experimental program, ten 150-millimeter antifriction bearings (fig. 6) were procured and tested. Liquid-hydrogen

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lubrication was used. The bearings were an extension of the J-2 liquid-hydrogen bearing criteria. The best performance obtained in a maximum-speed test was 26,000 rpm at a 2,500-pound thrust for 7-minute duration. In the maximum-load test, a 5,000-pound thrust load at 20,000 rpm for 2-minute duration was attained. All failures were due to overheating in the bearing.

The follow-on program for this work again will consist of an analytical program and an experimental program.

Under the analytical program Rocketdyne plans to do component design studies, that is, design and study of individual components that might be expected to give trouble in the design of a large high-pressure turbopump. They plan to do a turbopump design analysis. This will examine the economics, the logistics, the lead times, and the development times for the very large turbopump assemblies. They plan to do a turbopump performance analysis which would include control of a number of turbopump modules feeding single or multiple chambers such as might be found on a Nova vehicle. Such items as transient and throttling performance of these machines will be investigated. They also, if time allows, plan to do preliminary design on selected turbopump assemblies.

In the experimental studies Rocketdyne plans to test large antifriction bearings; they plan to test 200-millimeter bearings, again hydrogen cooled. They plan to investigate liquid-hydrogen hydrostatic bearings. They also will investigate a hydrostatic impeller seal which is a limited-leakage seal configuration such as might be used on a radial-flow impeller in place of a wear ring assembly. They plan to design and test a high-pressure volute such as might be found on large high-pressure engines. Work on the impulse pump will be continued. A rotor to fit last year's design parameters will be designed and tested, and the machine will be run with a highly loaded two-stage stator assembly. They plan to do some study of a pump inlet fluid-injection system, such as might be used with a throttleable engine. And finally, they plan to continue the cavitation scaling work.

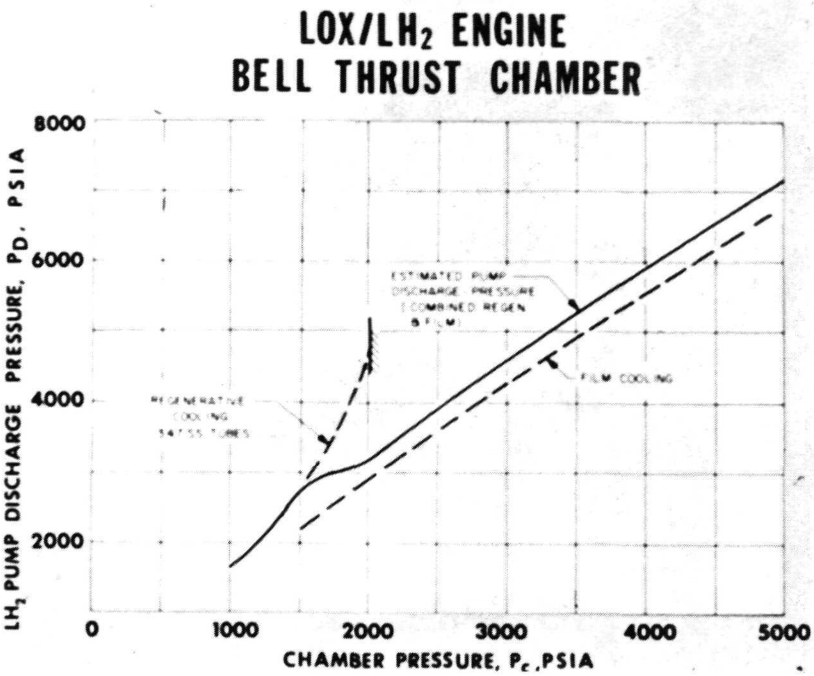

Figure 1

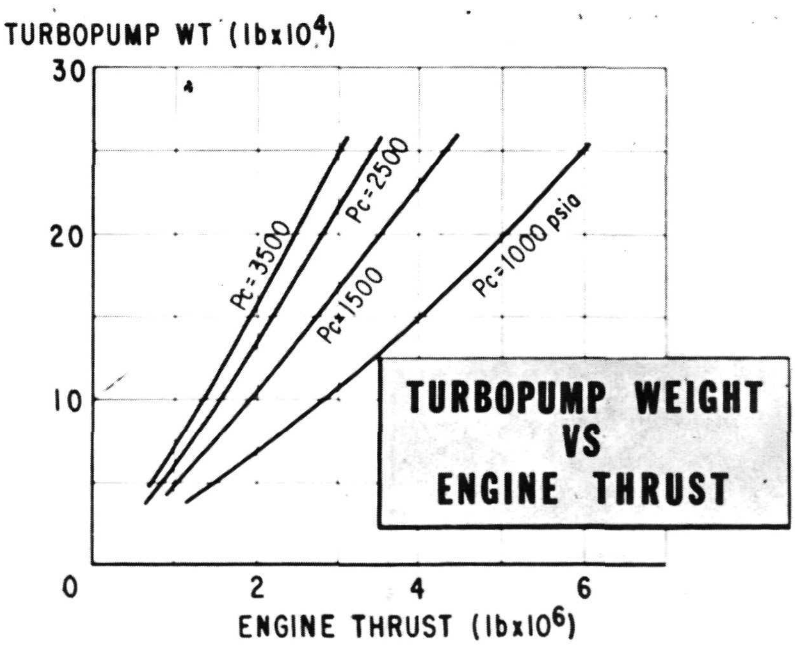

Figure 2

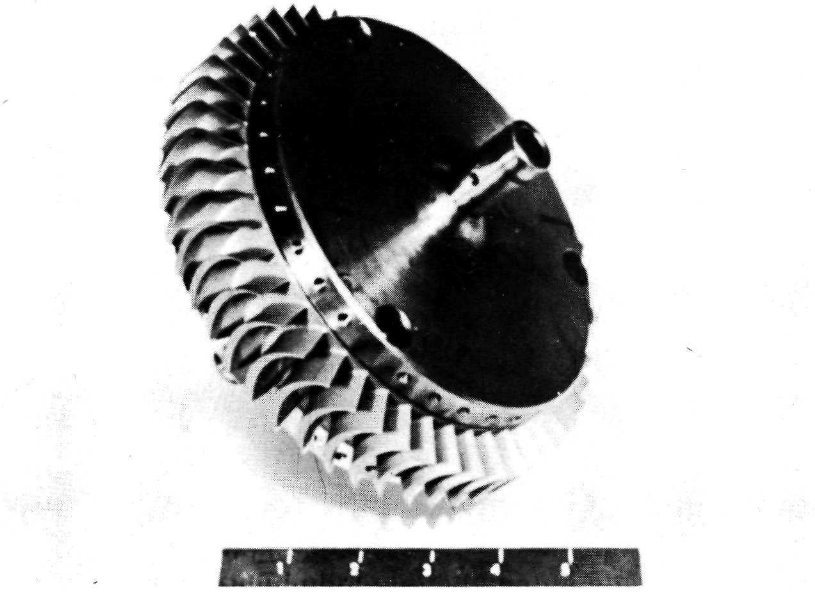

Figure 3.- Experimental rotor.

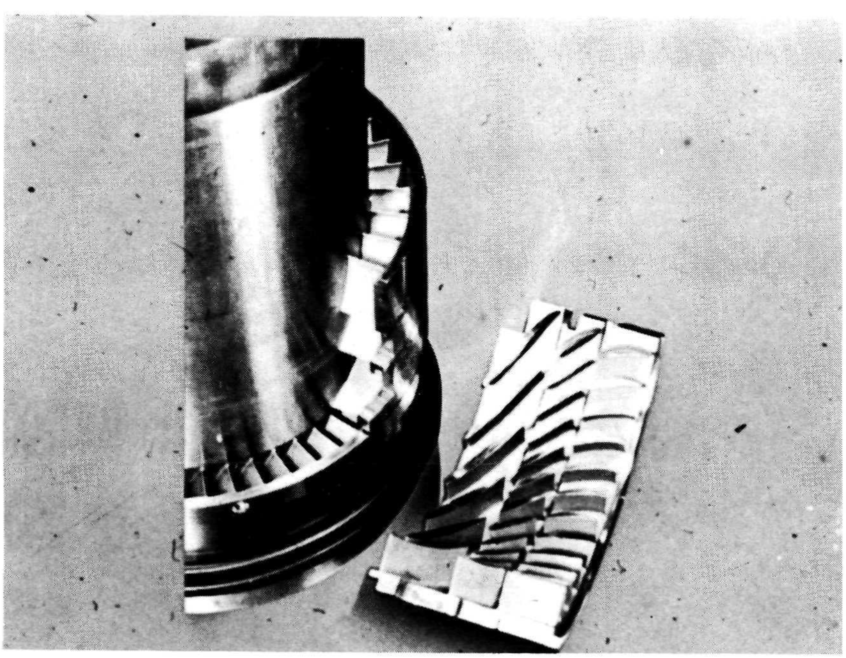

Figure 4.- Partial assembly of rotor.

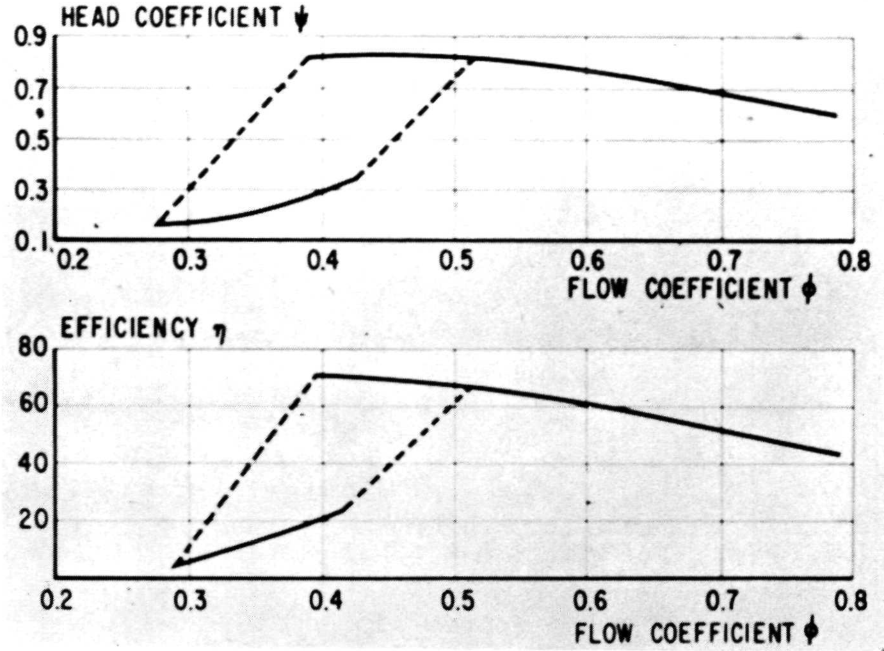

Figure 5.- Characteristic curves.

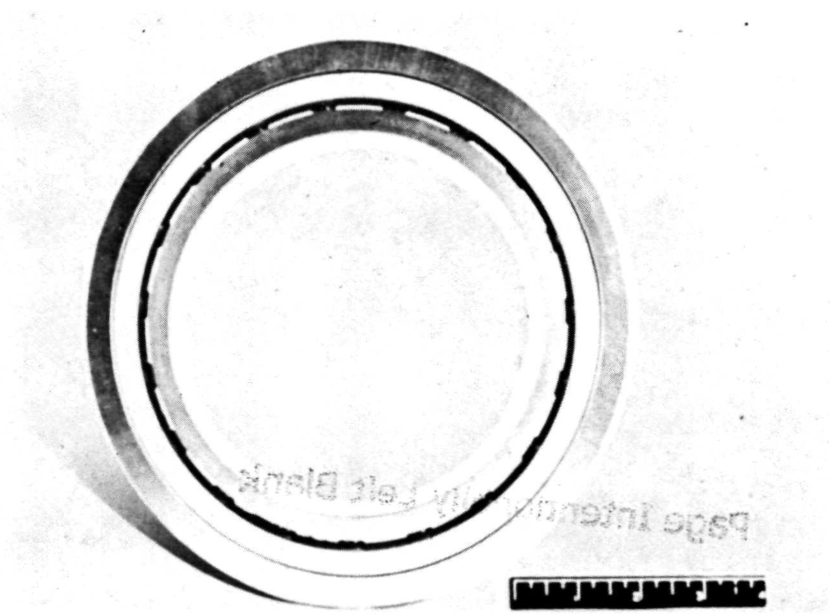

Figure 6.- Antifriction bearings.

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21. PUMPING WITH TURBOPUMPS FOR LARGE ROCKET ENGINES

By Melvin Hartmann

Lewis Research Center, NASA

The kinds of ideas that have been explored in these large, high-pressure turbopump studies are quite disappointing. It seems that no one is using any originality or ingenuity; they are only grinding over the same ideas that have been tried before. Even though packaging is an important aspect, it is disappointing that the number of pipelines might be the only consideration for selecting the number of turbopumps in a large engine. A plug-in turbopump, though it is a fine idea, is not a great advancement over other innovations that have been tried, and the monstrosity of a turbopump buried in the center of an engine presents some interesting problems, too.

In this paper I intend to discuss some of the concepts to be dealt with in going to high-pressure high-flow turbomachinery. To do this I shall discuss some of the things that we have been doing at Lewis in our pump research group and hope that these will contribute to the solution of the problems that we are facing now.

In figure 1 is shown a sketch of the turbopumpman's view of the large engine. Although in the past we have thought that some concepts are overly complex, I want to present them for your consideration once more.

This is the hydrogen tank, with a boost pump mounted in the tank, as in Centaur. It is followed by the high-pressure main pump feeding hydrogen to the engine. This tank has been purposely shown as not having a pressurization system because I would like to discuss the kind of pump that can pump from a tank that does not have a pressurized system, a locked boiling tank. I hope to substantiate this with some experimental results we have obtained at Lewis. I also want to discuss how we can use the boost pump to optimize the main pump, and some of the factors that make the main a good pump rather than just taking some pump, such as M-1, and adding a boost pump to make up for lack of tank pressurization.

I have shown the same kind of a system on the lox side, since there are now some indications that we might be able to use locked tank pumping suction in lox. I am not proposing that we use these systems but am merely offering these concepts for consideration.

Figure 2 shows some of the types of machinery that might be used for the high-pressure pump or what was referred to in figure 1 as the main pump. Figure 2 shows centrifugal, two-stage centrifugal, axial, and axial impulse machines. When high pressure is needed, it is obvious that it is necessary to try to get as high pressure per stage as possible. Then as high speed as possible must be attempted within the stress limits or cavitation limits.

The comment was made that the Pratt & Whitney machine was running at 2200 feet per second tip speed; I consider this rather high. Lewis started out in the M-1 with a centrifugal hydrogen pump at some 1600 feet per second but switched to two stage, and even then the machinery was so large that we finally switched to axial flow. I would like to point out to you that if the Pratt & Whitney-type, two-stage back-to-back device is scaled up to 6 million pounds of thrust, the pump diameter approaches 4-feet which looks rather large.

The normal axial flow pump in figure 2 has a set of inducers in the front and sets of stators between each blade roll. I would like to draw attention to the axial impulse pump because I intend to quit talking about it. Figure 2 shows the inducer and a number of stators of this type to build an impulse stage. When an impulse stage is working properly, it must ideally have a positive head flow curve. Experimentally it does not because, as the flow is increased, the velocity and diffusion problems in the stators increase so fast and the losses build up so fast, that the result is in reality a negative-shaped curve. The irony is that if we could make it work, the impulse-type pump may be stable in a system.

Figure 3 is a hydrogen pump that Lewis is trying for getting high pressures per stage from centrifugal pumps. This is radial blading, the type of blading used to get high pressure. However, it must be admitted that this pump was not designed for good cavitation conditions - the height of these blades at the inlet is very low. This is the type machine in which a boost pump or some other device would be used to take care of the cavitation problem.

The measured performance is shown in figure 4. It has a very narrow useful range, from some point where the curve is essentially flat to some higher flow where the head is dropping off very rapidly. These data were selected for the head coefficient curves at some inlet pressure with the suction head not too high, in which the high flow is limited by the cavitation cut-off. If a boost pump is used in front of this pump, this curve would continue in the high flow direction before falling off.

The two sets of data shown in figure 4 are for water and hydrogen; the normalized net positive suction head is given. The main point is that when the curves are the same, the cavitation effects are the same; the normalized net positive suction head in hydrogen is some 20 feet below that which would be required in water. This fact will be stressed later in discussing cavitating pumps. The pressure stability in this device is very impressive. The flow must be reduced to the very-low-flow region before appreciable pressure perturbations occur in the system. It is quite a stable pump.

The head coefficients that can be obtained for these high-pressure hydrogen pumps are very important. This one was pushing 0.7. It is doubtful in radial centrifugal machinery that it will go much higher than this. Besides being the head multiplied by the constant over the tip speed squared, this is also the slip factor times efficiency, so probably the value indicated in this figure is near the limit.

An estimate of 0.65 as the head coefficient of any existing centrifugal machinery would be a generous estimate. In figure 5, the pressure rise is plotted against tip speed for a centrifugal stage with a head coefficient of 0.65.

Again, the Pratt & Whitney people were running at 2200 feet per second where they should be able to get something in excess of 3,000 pounds pressure. However, they have used two stages and slightly lower head coefficient. When I objected to using tip speeds of 1600 feet per second for the M-1 engine, my point was the type of blading that they were using. Even though very careful stress analysis indicated bursting speeds of over 2500 feet per second, with the type of blading they were using, the vibration stresses would be very high, and so that it would be difficult to make 1600 feet per second.

Figure 6 indicates the type of blading required for axial flow. The three axial flow rotors shown are three different types of rotors that Lewis studied rather intensively in its research program. They might be considered the kind of building blocks used in building up an axial flow machine. First is shown an inducer with such characteristics as high blades to get good cavitation performance. The second stage might be an intermediate stage where the cavitation problem is less, so the axial velocity is increased. The third might be a high-pressure stage.

Tabulated in the figure are some of the design parameters that affect the head-producing capability of axial stages. The first stage which was optimized for cavitation has a very low flow coefficient, something in the order of 0.06. The blades are very flat; the axial velocities are very low. Low hub-tip ratios result in long blades, and it is difficult to get head coefficients of 0.2. Inducer stages of this type, with the long blade and its inherent stresses, are limited to rather low tip speeds. By "low" is meant speeds slightly over 1,000 feet per second.

The term usually used to express load capacity of the stage is the D factor; it is a lift coefficient. It relates the velocities on the blades and so on, and indicates whether an efficient stage will be obtained. In progressing through these stages, those stages that have been selected for discussion have approximately the same D factors, between 0.6 and 0.7. From blade-loading studies Lewis has found that this range of D factors usually produces high stage efficiencies.

Of the stages shown, the hub-tip ratio is increased progressively through the machine. To get the flow through the machine now, the through-flow velocity must be increased. The flow coefficients have gone from 0.06 to 0.42. At the same time, without increasing the D factor, the head coefficient is increased to something like 0.4. This is accomplished by judicious combinations of hub-tip ratio, D factor, and head coefficient values.

Therefore, this is the loaded stage, and these three stages in figure 6 are the kinds of stages depended upon to get high-pressure ratio per stage. Using 0.4 as a useful head coefficient, figure 7 shows the pressure rise plotted against tip speed for axial stages of the loaded stage type. If the tip speed were something like 1600 feet per second, with hydrogen a stage pressure rise of the order of 800 psi per stage could be obtained.

The Pratt & Whitney people think of high-strength alloys, of the kind they use in their machine, talking of numbers of 1800 to 2000 where we have exceeded

1000 pounds per stage. It should be pointed out that this is in the category of 125,000 horsepower per stage.

This horsepower is mentioned, however, to point out that there are very large fluid stresses on the blades, bending stresses. Even though this is high hub-tip ratio machinery, the blades are quite short. In this type of machine the bending stresses are something like three times the centrifugal stresses. Thus, if the designer is very careful, loaded stages may be possible at 1,400 or 1,500 feet per second in stainless steels, but super strength alloys are necessary at higher rotational speeds.

Figure 8 concerns the most popular problem in the turbopump business, stability. Lewis began having a little trouble with the F-1 although it has not always been this same trouble. The Titan people began to see that their turbopump was interacting with the structure in such a way as to cause very high vibrations. Others look at the Thor and Jupiter, and say that the turbopump was involved in large vibrations on each vehicle. The problem of how the pump contributes to the instability of a system is illustrated by this system (fig. 8) which is an idealized system of the F-1 engine. The upper left curve is a head rise versus flow curve; but in the plot of the head rise versus time, in the upper left curve the head rise varies and is not a discrete point for a given flow. I have chosen to call the variation the head from peak to peak, that is, the variation about some mean pressure. It seems reasonable to expect that that variation, peak to peak, when divided by the head rise $\left(H_{p-p}/\Delta H\right)$, might be a characteristic number or dimensional number. This is undesirable because it indicates increased head rise; even if that ratio stays the same, larger pressure fluctuations can be expected in the system.

To carry the story a little further, the lower left plot shows that departure from some operating point causes this $\left(H_{p-p}/\Delta H\right)$ value to vary with flow, increasing with decreasing flow values, and the lower right plot shows that it also increases in going to lower net positive suction heads (NPSH).

This is a very brief introduction to the pump instability problem.

Figure 9 is a sketch of what may have happened in one missile. The head rise characteristic curve when plotted against net positive suction head had a dip near low NPSH. At some time during the flight, as the tanks began to empty, operation was down in this NPSH region, where the slope of the line becomes negative. After Rocketdyne encountered this problem, a check with the analog verified that it is an unstable region because, as inlet pressure is lowered, the general pressure level of the system is lowered. Apparently, the inducer tends to increase its pressure rise, and it is going against the system, so the system becomes unstable. This problem, which was caused by operating with a dip in the characteristic curve, Rocketdyne solved by drilling holes in the inducer.

Figure 10 is a characteristic curve of an Aerojet pump from Titan II. Aerojet is clever at selecting designs which come up with this type of characteristic curve, plotting head against net positive suction head - with decrease inlet pressure, the pump pressure rise falls steadily. This should be fine except that when it is put on a vehicle, the vehicle starts shaking up and down, causing the pressure at the face of the pump to vary, sketched here as P_1. As a result, the pump operates within a range between two points and the head varies between these same two points. In other words, the pressure fluctuation that went into the pump, shown by the dotted line, has been amplified.

Figure 11 is a group of graphs of pressure fluctuations associated with pumps. These instabilities are all very much related, as will be shown later, but they manifest themselves in different ways. Here the system is one from the F-1, but probably represents the kind of problems that the Thor people begin to see. The head-Q curve shows that the head is not steady with time. Also shown is the head curve versus net positive suction head. This is well behaved and comes out flat before dropping. However, the peak pressures, or peak heads, and the minimum heads are also plotted here. As H_{sv} decreases, these limits diverge over the entire H_{sv} range. Blade wake frequencies can be seen in this fluctuation, and in this lower range a 10- to 20-cycle pressure fluctuation begins.

On the basis of the many variables plotted against the net positive suction head, the relationship to cavitation becomes apparent, but they are not necessarily entirely a result of cavitation.

In figure 12, without consideration of cavitation but based only on the streamlined flow through a series of pumps, are indicated regions of eddy flow which I believe are related to stability. The upper left diagram is the pump impeller, also represented in figure 3, investigated at Lewis. It seems to be quite stable at this stage. This impeller has a very narrow blade height, similar to what Pratt & Whitney uses in its high-pressure pump. The streamlines are "very well behaved"; it is not difficult to make the flow follow that passage.

The upper right sketch in figure 12 represents the flow into Nerva. The very large inlet was purposely made large to get high suction specific speed, low net positive suction head. However, other problems arise with that low net positive suction head. The axial velocities are very low. Any departure from design shape of the streamlines results in reverse flows, or eddies. There is an eddy shown in the tip region, which is rather typical of Aerojet machinery; an Aerojet pump usually bends rather sharply and has some eddies occurring in the region of the bend also. This is one of the disadvantages - a very strong tendency for instabilities - that had to be accepted in going to low net positive suction head, large diameters, and large passages.

The data on the inducer shown in the lower sketch are supposedly model data from the F-1 on which the streamlines were surmised by surveying inlet and outlet flow distribution. It was surmised that these eddies were quite large, both at the inducer tip and at the discharge of the inducer. So far in this discussion the relationship between the eddies and cavitation has not been covered. When vapor forms in this region, the vapor also contributes to the bending of the streamlines and may, or may not, cause the eddies to increase, or cause the instability to increase.

Figure 13 is one of the test facilities at Lewis; this figure is useful to open a discussion of pumping from a boiling locked tank. This is a large tank standing on end with a boost pump mounted in the bottom. It has an inducer driven from the bottom. It can be operated in two modes: (1) either circulating the fluid back into the tank for continuous operation, or (2) bringing another tank up on wheels and pumping out of the locked tank.

In figure 14 are graphs of some data taken from this facility. The time plotted is the time during the run. The tank is loaded with hydrogen. The pump starts pumping out of the locked tank; this is represented as zero for all curves. The flow rate and pump speed increase together. The pressure comes up and then is constant with time. The important pump parameters all seem to relate rather smoothly with time until the tank is empty. In figure 14, the apparent change of the net positive suction head of the pump to negative and the curve of the fluid vapor pressure in the tank are of interest. At the start of pumping, there is a slightly positive net positive suction head. As the fluid is pumped out, the vapor above the liquid expands for this short period of time. Then the vapor pressure of the fluid drops, and during the rest of the test the pump operates with what appears to be a negative net positive suction head. The amount of negative net positive suction head - about a half-pound - is the same as the pressure necessary to evaporate fluid in the tank at the same rate it is being pumped out, forming gas to replace the liquid that is being pumped out of the tank. During the pumpout time, the tank pressure drops about 2 or 3 pounds. These studies are being continued with the idea that some of the large tanks may tolerate a few pounds change in pressure, so that they can be pumped without a pressurization system, with the fluid at or near the boiling condition, and to determine the problems of such pumping systems.

The equation for specific suction speed is

$$S_s = \frac{N\sqrt{Q}}{(\mathrm{NPSH})^{3/4}}$$

However, from experience in pumping cryogenic fluids, it has become apparent that the equation is not valid for work in cryogenics. For cryogenic calculations a term must be added to the denominator, so that the equation becomes

$$S_s = \frac{N\sqrt{Q}}{(\mathrm{NPSH} + \mathrm{TSH})^{3/4}}$$

(Unfortunately, that term has been labeled thermal dynamic suppression head.) In experiments that Lewis has been running - research people are always limited in the equipment they have - the values of $N\sqrt{Q}$ have been rather low - around 700,000. In reality, data where this number is higher would be more useful - J-2 and Nerva are in the 2-million category, I think.

These data were taken at rather low numbers; however, the thermodynamic suppression head values are about 30 or 40 feet. In the pump in figure 3, not optimized for cavitation, the TSH values are something of the order of 20.

In the Nerva and similar vehicles, the TSH values are somewhat higher, 70 to 90 feet. The implication of this is that with a TSH value of 70 or 90 feet, the NPSH term can be allowed to go to zero. That is the concept used when evaluating boost pumps for large engines. In reality, they become rather large low-speed pumps, but they do boost the pressure a bit.

The following figures then deal with a 6-million pound engine, utilizing some of these concepts, and attempts to determine how such concepts fit into this type of engine. It is by no means a comprehensive study.

The two graphs in figure 15 represent data on a boost pump for a 6-million pound thrust engine designed to have zero net positive suction head. It was found that a single stage, turning over at 2300 rpm, is about 43 inches in diameter and produces about 40 psi. If two stages are used in the boost pump, a pressure rise of a little more than 90 psi is obtained, and three stages produce about 150 psi. These figures are based on a thermodynamic suppression head of 100 feet. Since that time, single-stage pressure rises of more than 50 have been obtained, and, theoretically, 100 or more seems quite feasible.

Figure 16 shows the effect of thermodynamic suppression head on boost pump diameter. The figures just quoted were for a boost pump diameter of about 43 inches and for 100 feet of thermodynamic suppression head. However, this is the way the diameter of such a boost pump would vary, depending on the thermodynamic suppression head that can be achieved by good design, or more careful installation. The pump pressure rise increases with increasing amounts of thermodynamic suppression head, even though the diameter does not decrease appreciably.

Although the previous discussions in this paper have dealt impartially with different kinds of pumps, in discussing pumps that might produce higher thermodynamic suppression head, only a few axial flow machines will be discussed. The curves in figure 17 represent data for the main hydrogen pump with the number of boost pump stages shown. This indicates the effect on the main pump as the vacuum boost pump pressure rise levels off. The required diameter of the main pump decreases as the boost pump pressure rise increases.

Again for a 6-million pound thrust engine a main pump of less than 30 inches is possible. In fact, the scale on the bottom graph is in error, and each of these numbers is displaced by one, so that actually the pump diameter ranges slightly below 28, with a two-stage boost pump. It is a fairly small size for the quantity of flows handled for the 6-million pound engine.

Instead of the 20 stages that might have been obtained with a single-stage boost pump, in a 2-stage boost pump it was possible to drop the number of stages here to about 10 for a topping cycle, for which a pressure rise of 5,700 psi was assumed, and something like 8 for 4500 psi for a bleed cycle.

Figure 18 is a continuation of figure 17 in which are shown more of the characteristics of the main stage pump when it follows a boost pump. At this point I think you can see some of my disappointment. We are pushing up the rotational speed toward 15,000 rpm, but the tip speeds that can be obtained, even with the two-stage boost pump, are only about 1,250 feet per second. To get up to this tip speed it was necessary to raise the hub-tip ratio to 0.9, using very short blades and larger diameter machinery, and the rotational speed would be suitable for turbine drive. Bearing DN numbers shown are only in the 2.5 million category. Original studies involving hub-tip ratios of 0.8 indicated bearing DN numbers of the order of 4 million. Actually, even with the boost pump, it is difficult to utilize very high tip speeds.

From this discussion and some of the trends shown, it should be apparent how important a boost pump might be, and that boost pump complexity may be worth studying, particularly in this large machinery.

Such high rotational tip speeds would be useful; it might be possible to split the main pump into two machines, the second one running faster and the first one supercharging the second stage. It may be that such two-stage main pumps are worth studying.

Figure 19 indicates what such a system might look like. The pump following the two-stage boost pump is called a supercharger pump. The second stage now is like the diameters previously discussed, 27 inches. The figure indicates what happens as the pressure rise for the topping cycle (5700 psi) is split in various amounts between the supercharger and main pump. Obviously, at zero supercharging there is only the main pump and for that particular pump probably 10 stages for 5700 psi which is a point to consider when something like 40 percent of the pressure rise is taken in the supercharger. A supercharger pump and a main-stage pump, each of some four stages, are needed. The weight numbers are only relative; the supercharger has four stages, and the main pump requires about four stages running at 1400 and 1500 fps tip speeds. As a result, the weight factor seems to minimize at some point not so far from the 40 percent split.

Probably this could be carried further, but of primary importance is whether a lox boost pump could be built to pump from a locked tank. There is little information to use as the basis for speculation, but in the F-1, the difference between the water and the hydrogen gives something like 10 feet of fluid property effect. It would be difficult to identify this effect as thermodynamic suppression head because theoretically, that effect must be very small. On the other hand, in lox, the tension effects seem to be very large.

Applying 10 feet of thermodynamic suppression head or fluid property effect - and you see now I begin to change terminology - into a lox boost pump for the 6-million pound thrust level would result in a lox boost pump diameter on the order of 50 inches. The pressure rise over such a boost pump is at least 10 psi. Thus, it may be that for cryogenic propellants, pumping from a locked tank may be considered.

This paper has covered some of the things we must consider and briefly summarizes the activities at Lewis that might be applicable to the development of high-pressure pumps and large turbomachinery. Perhaps some concept in boost pumps, some data, and some analyses being conducted in the boost pump tank situation, such as pumping from a locked tank, might be useful in future applications. Although the machines we have discussed are rather large, I have been quite conservative as to pump stability effects. In all these cases, the maximum of suction specific speed used was 30,000 which means a fairly high flow coefficient can be used and a fairly stable pump should result.

[redacted]

BOOST PUMP SYSTEM

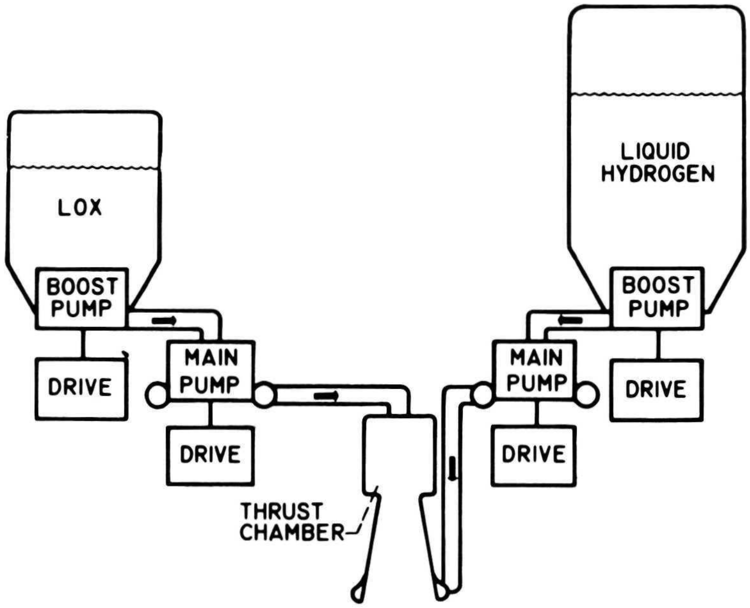

Figure 1

CENTRIFUGAL AND AXIAL FLOW PUMP CONFIGURATIONS

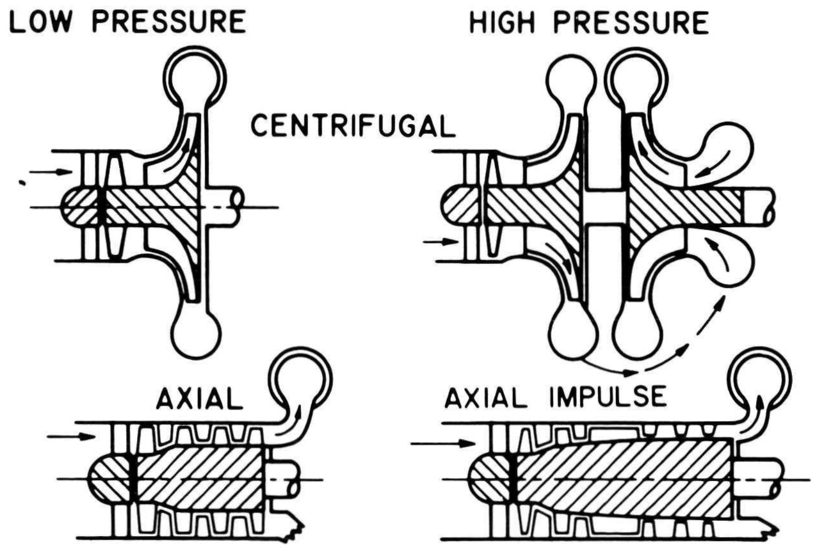

Figure 2

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LARGE FLOW HYDROGEN PUMP ROTOR

Figure 3

CENTRIFUGAL PUMP PERFORMANCE IN HYDROGEN AND WATER

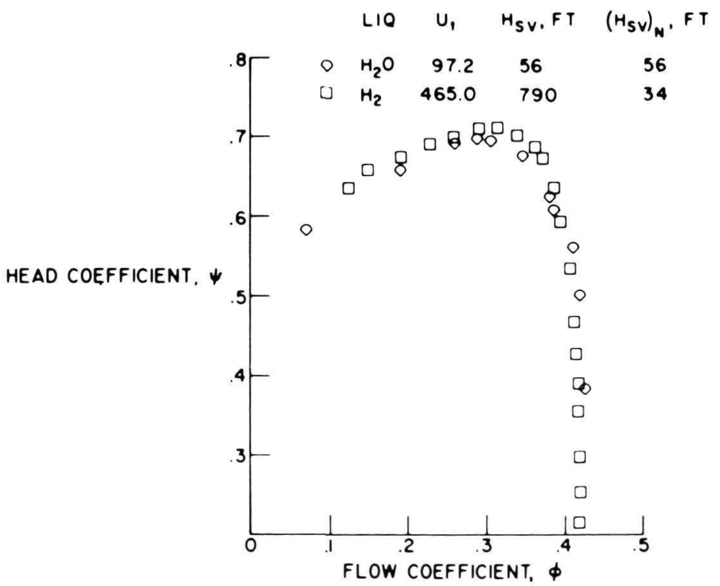


Figure 4

[redacted]

HYDROGEN PUMP PRESSURE

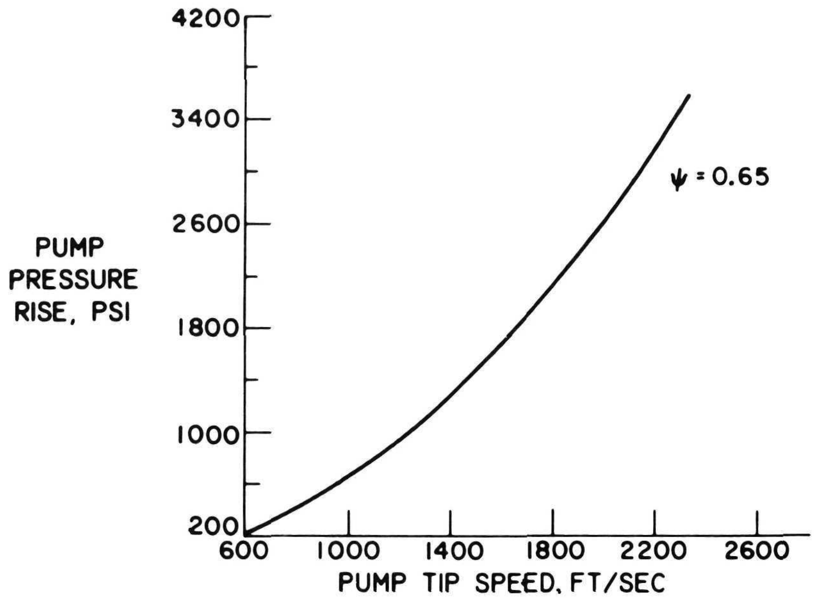
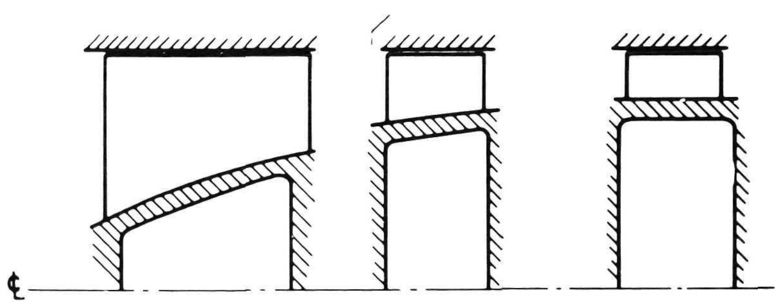


Figure 5

DESIGN FACTORS EFFECTING AXIAL PUMP STAGES



H/T ~	0.3	0.7	0.8
ψ ~	.2	.27	.42
"D"	.6	.7	.65
ϕ	.06 -.29	.29	.42

Figure 6

[redacted]

HYDROGEN PUMP PRESSURE RISE AND POWER REQUIREMENT V.S. TIP SPEED FOR A 0.4 HEAD COEFFICIENT

THRUST, 6000 K

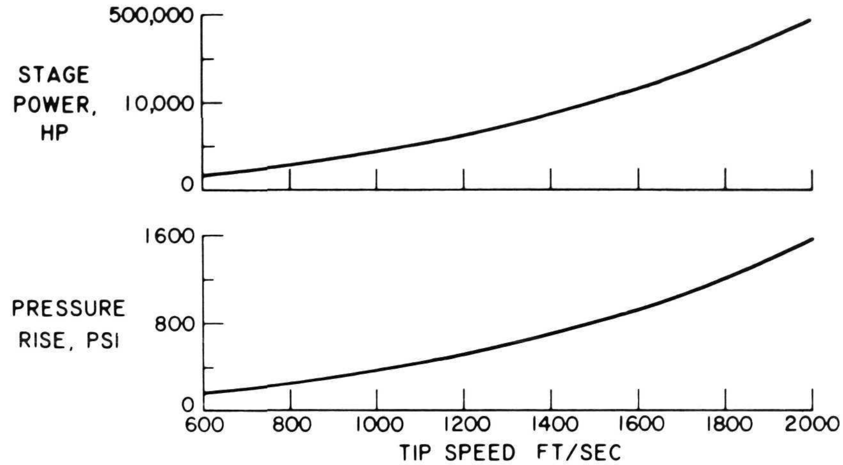


Figure 7

PUMP DISCHARGE PRESSURE FLUCTUATIONS DUE TO UNSTEADY FLOW CONDITIONS

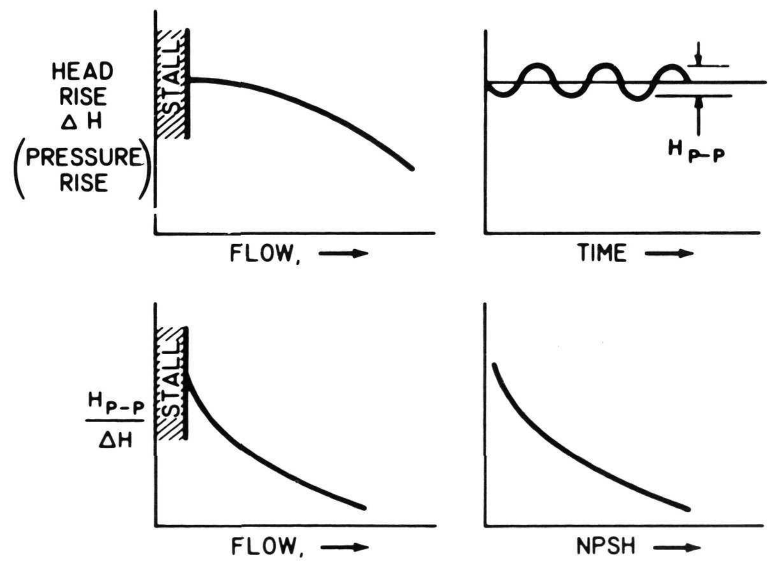


Figure 8

[redacted]

INSTABILITY WITH NEGATIVE PUMP SUCTION CHARACTERISTIC

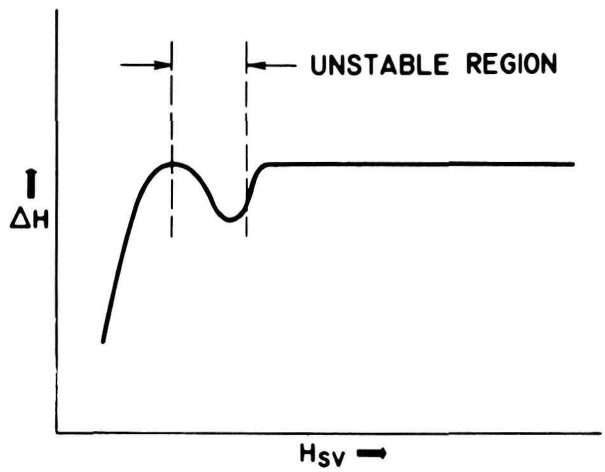


Figure 9

AMPLIFICATION EFFECT OF POSITIVE PUMP SUCTION CHARACTERISTIC

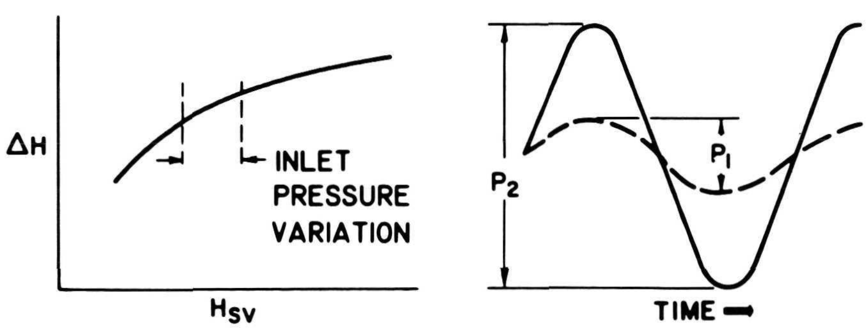


Figure 10

[redacted]

PRESSURE FLUCTUATION ASSOCIATED WITH PUMPS

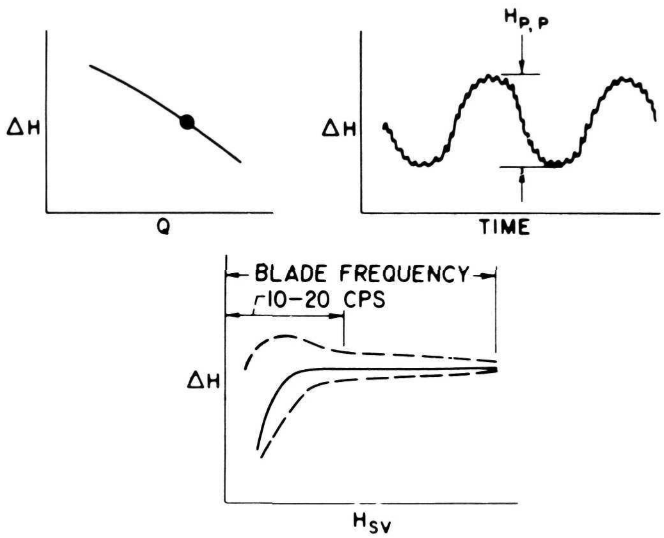


Figure 11

EFFECT OF DESIGNING CENTRIFUGAL PUMPS FOR HIGH-SUCTION PERFORMANCE

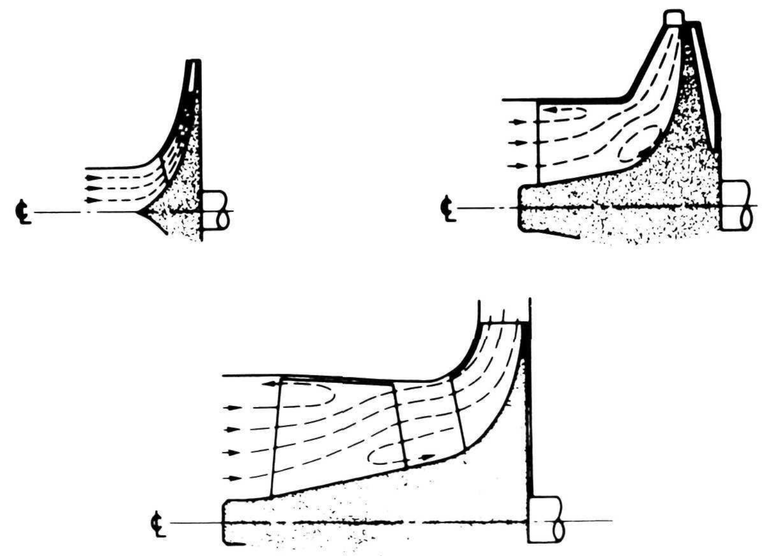


Figure 12

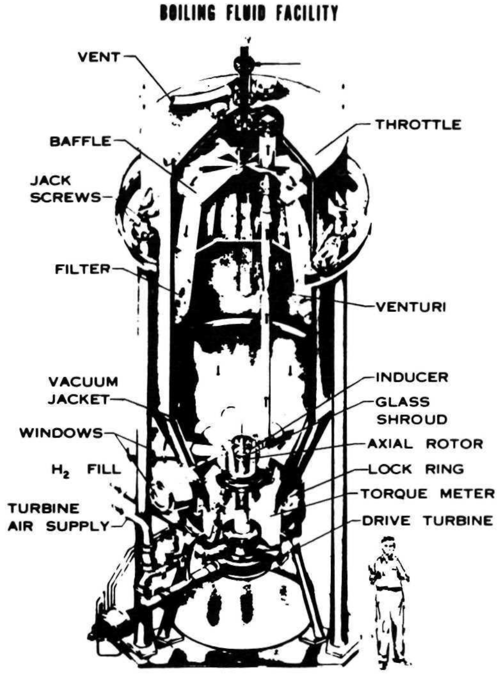


Figure 13

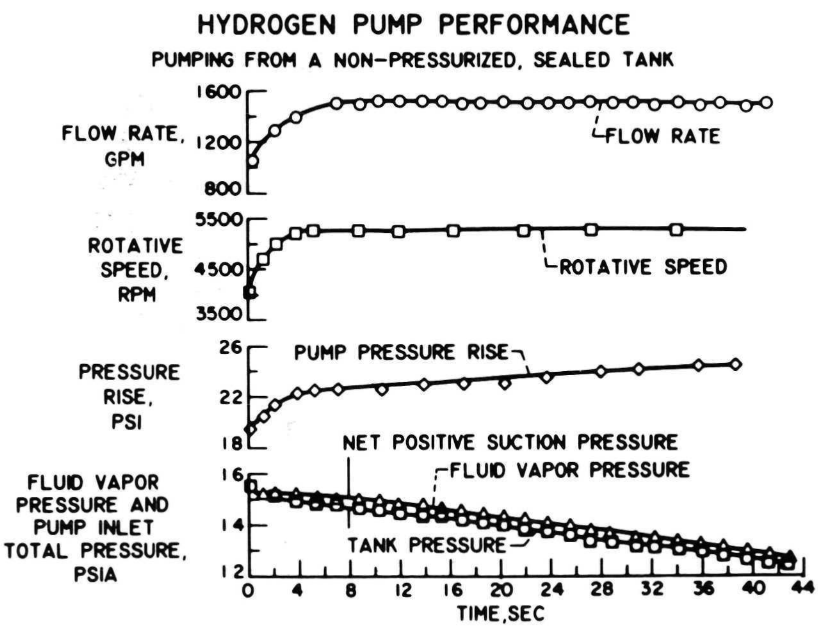


Figure 14

HYDROGEN BOOST PUMP FOR ZERO NPSH

THRUST, 6000 K

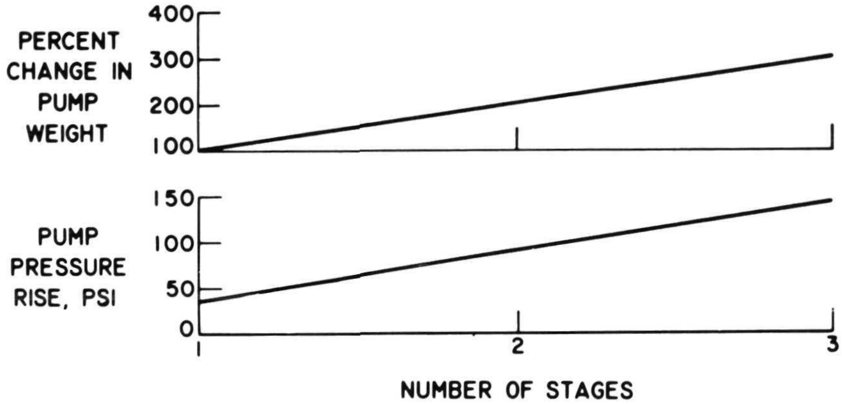


Figure 15

TWO-STAGE HYDROGEN BOOST PUMP

THRUST, 6000 K

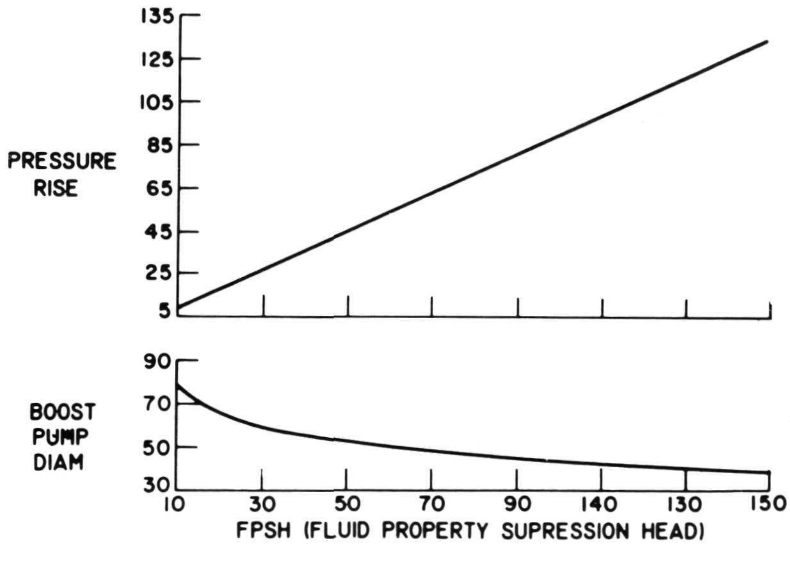


Figure 16

MAIN HYDROGEN PUMP CHARACTERISTIC

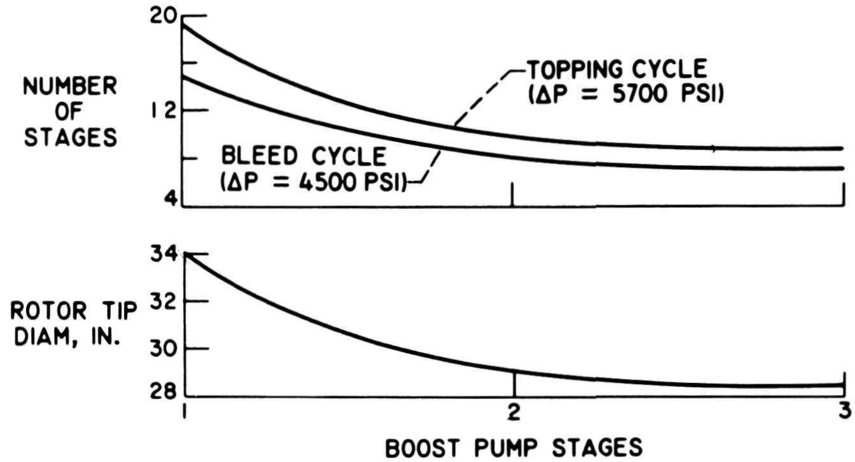


Figure 17

MAIN HYDROGEN PUMP CHARACTERISTICS

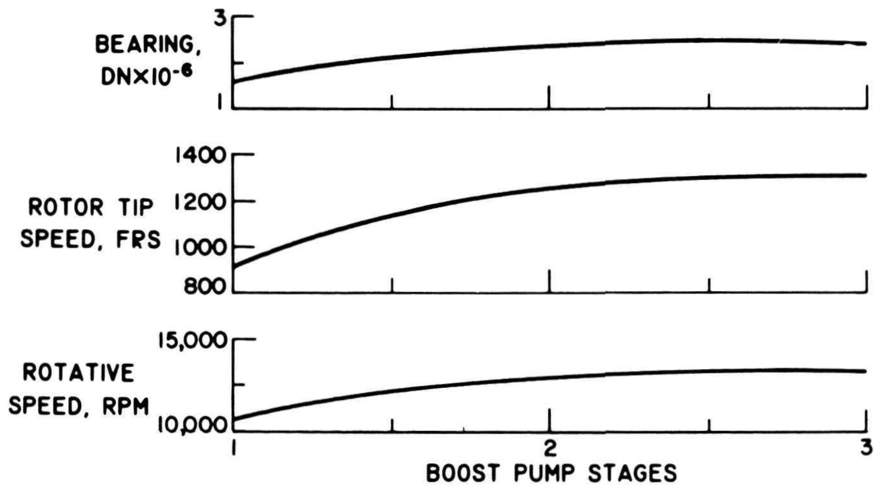


Figure 18

[redacted]

PERCENT TOTAL WEIGHT AND NUMBER OF STAGES VS PRESSURE SPLIT

TWO-STAGE HYDROGEN BOOST PUMP

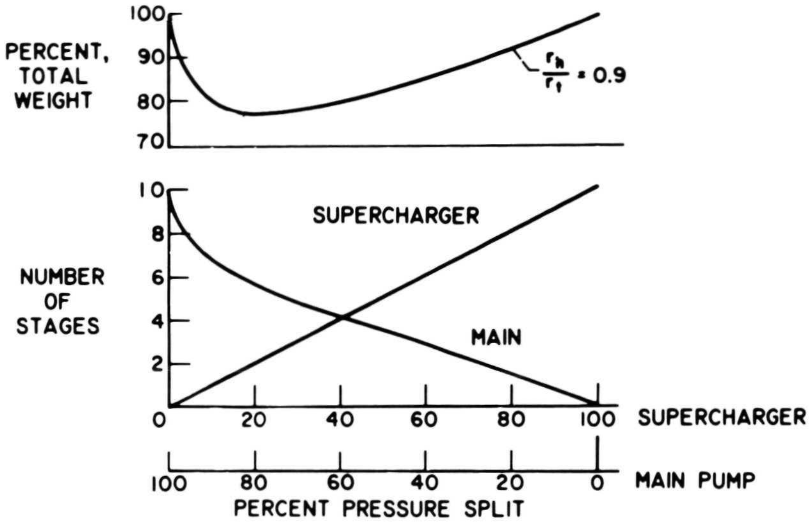


Figure 19

TWO-STAGE LOX BOOST PUMP

THRUST, 6000 K

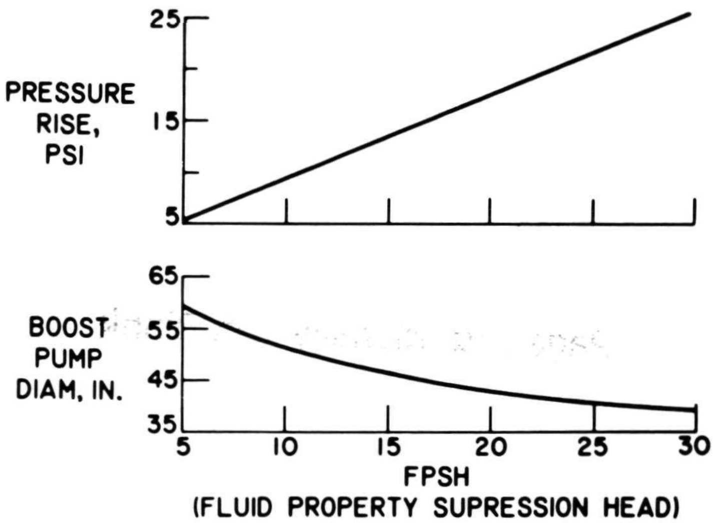


Figure 20

[redacted]

[redacted]

22. TURBINE DRIVE SYSTEMS FOR TURBOPUMPS FOR LARGE ROCKET ENGINES

By Warner Stewart

Lewis Research Center, NASA

The previous speakers have already indicated that, for these large engines, high pressure levels are needed in the chamber. This results in a requirement of pumping systems capable of developing pressure levels two to three times that currently being used. It also means that the associated pumps must be much larger with higher head rises.

Mr. Hartmann also indicated that there is a desire to use boost pumps to reduce the required tank pressure, as well as to make the main pumps smaller and lighter.

All these pumps must be driven by turbines, and these turbines must have appropriate feed systems. So the objective of this discussion is to describe some of the aspects of the turbine drives, particularly for higher system pressure levels.

The emphasis will again be on hydrogen/oxygen systems as they have been indicated as being of most interest. In particular when we describe some of the turbopump configurations, we will dwell principally on the hydrogen turbopump since this is the most critical, has the highest head rise, and yields the greatest geometry problem.

The scope of this discussion will be first to dwell a bit on the feed system to indicate the effect of increased pressure level on the choice of the system that might be used. We will then show some representative turbopump configurations that might be associated with the various types of feed systems. Finally, we will briefly describe some of the aspects of the choice of drive turbine for a boost pump and give a brief description of some representative boost pump and drive configurations.

Let me first dwell on the feed system selection. Yesterday contractors discussed the pros and cons of the various feed systems. Apparently the topping and gas generator systems have been compared and analyzed for years, and different contractors take divergent views as to which is best. Let us briefly consider these two systems.

The gas generator system will be called the bleed system. There are in general two types of bleed systems that have been considered, the so-called parallel flow system and the series flow system. These have been proposed as various engines have been evolved.

Figure 1 is the parallel flow system. In a general arrangement the hydrogen flows in through a pump, into the nozzle coolant jacket, and then into the thrust chamber; and the oxygen comes from the tank, being pumped and injected.

[redacted]

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A small fraction of the propellants are bled off and burned in a gas generator. In the parallel system, the flow is split into two parts to drive the turbines which are required to power the associated pumps.

Figure 2 shows the series system, which is being used in the J-2 and M-1 engines. Basically, it is the same system except that after the gas generator, practically all the flow goes through the hydrogen turbine and then the oxygen turbine. An advantage of this system is that a multistaging effect is obtained with two turbines rather than one, permitting each turbine to require fewer stages than that for the parallel system.

There are problems with the bleed system. The principal one is a thrust reduction. The turbine flow is essentially by-passed around the main chamber resulting in an impulse loss of some magnitude. In addition, the turbine flow is hydrogen-rich, having an oxygen/fuel (O/F) ratio of approximately 1 compared to the main chamber ratio of 5:7. This means that a large percentage of the most critical fluid, hydrogen, is passing through the turbine. For these reasons we like to minimize the amount of bleed rate.

The main question is the effect of increasing chamber pressure level on this bleed rate. Such an effect is illustrated in figure 3 in which are shown three curves of the bleed rate, or the ratio of the flow diverted to drive the turbines over the total pump flow, as a function of chamber pressure. The lower curves were drawn for two temperature levels, 1500° R and 2000° R. There is a very small effect of temperature on bleed rate, but as the temperature level is varied there is a molecular weight compensation due to the required change in O/F ratio with the subsequent little effect of temperature on flow.

These curves were based on an assumption of an O/F ratio of 7 to 1, and turbine efficiency of 70 percent, which is rather good. With increasing pressure, the bleed rates become larger varying linearly. In the 3,000 psi area, under optimum conditions the bleed rate is 4 percent or greater which can hurt the cycle quite a bit.

In particular, the required bleed rate can hurt the system when the O/F ratio or turbine efficiency deviates somewhat from the assumed 7:1. For instance, the M-1 O/F ratio is nominally 5, and the overall turbine efficiency is less than 50 percent. These changes in conditions from those previously assumed have a drastic effect on the bleed rate as evidenced by the upper curve in the figure. At M-1 condition, 1,000 psi, about 3 percent bleed rate is required. Going up to 3,000 psi or better, the bleed rate goes up to 10 percent. This also means that about a third of the hydrogen flow that is being pumped is by-passing the turbine to drive the pumps. Obviously the bleed system is not advantageous at high pressure levels.

In a highly efficient turbine drive system, a large number of turbine stages is required. Figure 4 shows bleed rate as a function of number of turbine stages. Because of the high energy content of the hydrogen drive gas and the desirability of high efficiency, a large number of stages are needed to achieve the proper velocity. Of course, the number of stages can be reduced a bit from optimum without too large a penalty in bleed rate. Down to perhaps three stages the penalty is not large, but going down to 2 or 1, the bleed

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rate starts to increase quickly. The use of two-stage turbines in the M-1 in series is effective because the effect is more like three or four and, therefore, dips below the region where the bleed rate starts to really accelerate.

A schematic of the topping system is shown in figure 5 in which the hydrogen and oxygen turbines are driven in the topping fashion. The principle is that the hydrogen is pressurized and a preburner increases the temperature and adds energy to the gas. The gas drives the turbines and then goes into the main chamber where final combustion takes place.

This cycle, of course, has the advantage that by-pass flow is not required but at higher chamber pressures there is the problem of having to superimpose a substantial additional pressure requirement across the pump in order to have sufficient pressure ratio across the turbine.

This effect of chamber pressure level on the pump pressure is illustrated in figure 6. In the lower portion of the figure, turbine pressure ratio is shown as a function of chamber pressure for various turbine inlet temperatures. The turbine inlet temperature has a marked effect; as the temperature level is increased the same hydrogen flow is maintained, but oxygen is added. This increases the total turbine weight flow, so that even though the specific energy (C_p times temperature) does not change greatly, the additional weight flow increases the total energy which gives the spread of the temperature curves.

With increased chamber pressure, the pressure ratio across the turbine increases rapidly which indicates the advantages of high turbine inlet temperatures for such a system. However, there are definite temperature limitations.

Also, the system requires higher pumping pressures. For example, at 3,000 psi, and 1800° R, the turbine pressure ratio is in the order of 1.5, which requires greatly increased pressure rise across the pumps in order to drive the turbine.

The upper part of figure 6, in which hydrogen pump pressure rise is plotted as a function of chamber pressure, illustrates this point. At 3,000 psi chamber pressure, the pump pressure rise is in the area of 6,000 psi. As the chamber pressure moves up from that, or as the turbine inlet temperature drops off a bit, the required pump pressure rise really starts to sky rocket.

The conclusion, then, is that not only is a straight topping system a mechanically complex system, but also the pressure rise across the pump becomes extremely severe. The pressure rises are severe with conventional engines, but with the topping system the rise is tremendous.

The problem is quite complex because a feed system must be provided with adequate power for the pumps, and both the bleed and topping systems become unreasonable as the required chamber pressure is increased.

Mr. Hartmann suggested, in his discussion, the possibility of splitting the hydrogen pump into two parts.

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For such a split pump arrangement the feed systems to be used could include (a) a straight bleed system, where both pumps are driven in a bleed fashion, still having the problem of a large amount of flow, and (b) what might be called the supercharged topping which utilizes the good ingredients of both the topping and bleed systems.

The supercharged topping presents the more attractive solution. However, in discussing turbopump configurations later, the possibilities of what can be evolved with the straight bleed system will also be discussed.

Figure 7 shows a schematic of the supercharged topping with a low-pressure hydrogen pump which might create about 2,000-psi pressure rise, driven by a bleed system. This is then followed by a high-pressure hydrogen pump which could be in the dome of the engine, driven by a topping turbine. The oxygen pump could also be driven by a bleed-type turbine (parallel bleed system is shown for illustration.)

To illustrate the application of such a system, the M-1, with the desire to increase the chamber pressure level to 2500 or 3000 psi, can be used as an example. The oxygen pump is not stress limited. The speeds are down, and it is not a very difficult mechanical problem. Higher pressure levels can be reached in the oxygen pump without too much difficulty.

However, there are limitations in the hydrogen pump. These two pumps might be kept as they are, and the topping system used in the dome of the engine. The advantage of this arrangement is that it maintains the bleed rate at acceptable values and reduces the pressure ratio across the topping turbine to a low value.

These effects are illustrated in figure 8 for an example case of 3,000-psi chamber pressure and turbine inlet temperature of 1800° R. In this figure, the bleed rate and hydrogen pump pressure rise are plotted as a function of the ratio of the pressure rise for the pump that is driven by the bleed turbine to the total fuel pump pressure rise. Zero would represent a complete topping system with only one pump. There is a certain bleed rate shown at this point because the oxygen pump is still driven by a gas generator system. The value of unity would be an all gas generator system.

For example, for a 40-percent pressure split the bleed rate is about 2 percent, still rather small. In addition, the pressure ratio across the topping turbine drops drastically, because the high-pressure pump contributes only a little over half of the total pressure rise. This results in a large reduction in the total hydrogen pump pressure rise and represents a compromise between the two extremes.

From the foregoing considerations it is evident that at higher pressure levels, conventional feed systems become unacceptable so that more sophisticated feed systems may be needed to keep within reason the penalties that grow with pressure level.

In considering some of the turbopump configurations that might evolve at higher pressure levels, for comparative purposes the straight systems, which include the straight topping and straight bleed system, and the combined system will be considered.

Figure 9 shows representative hydrogen turbopumps that were evolved for the straight systems for 3000-psi chamber pressure at 6000° K thrust level. Three-stage turbines can be used for the bleed systems as a compromise between efficiency and mechanical simplicity. Two types of gas generator systems are considered, the parallel and the series. For the parallel flow bleed system, the three-stage turbine is shown driving an 8-stage pump at 11,000 rpm, which is much lower than minimum capacity of the pump. The pump hub-tip ratio is 0.9 in order to match the turbine which was stress-limiting. The diameters are on the order of 30 inches. Of course all of these are highly loaded pump stages at the limit of or somewhat beyond present technology.

For the series flow bleed system there is the possibility of a much smaller pump at a higher speed. The diameter is only about 23 inches resulting in a much lighter system. A lower hub tip ratio can be used on the pump (0.85), and a higher speed will permit using fewer stages. This is possible because with the parallel flow system, the full pressure ratio is taken across the turbine and only a fraction of the total turbine flow, whereas with the series system only a fraction of the pressure ratio is taken across the turbine but all the flow goes through it (except for control by-pass). In the pressure ratio range encountered, the increase in weight flow is more than compensated for by the reduction in pressure ratio across the turbine, so that a smaller annulus area is required at the exit. This permits, within stress limitations, going to higher speeds and represents an advantage for the series system. Another associated advantage for it is that it results in a much lighter turbopump.

For the topping system, a two-stage turbine is shown driving a 10-stage high hub tip ratio pump (0.9) at a low speed of 11,000 rpm. The main problem here is severe stress limitations in the turbine. Another major problem is the very large bending stresses in the rotor blading which must be compensated for by tilting the blade or by some other technique.

Figure 10 shows some comparisons for the split system. The split system may be of two types: either a complete gas generator system or a split gas generator and topping system. In either event, it is assumed that the lower pressure pump would be a bleed system.

The upper part of figure 10 shows the low-pressure turbopump. With a 2,000-psi pressure rise across this pump only about four stages, with a three-stage turbine, are required, and the pump is about 20 inches in diameter. It can be speeded up to 15,000 rpm which permits fewer pump stages.

The high pressure turbopumps are shown in the lower part of the figure. On the left is that for the bleed system, in which the turbopump is driven by gas generator flow (in series with the low pressure turbopump). The pump pressure rise was 2500 psi in this case. A three-stage pump is shown operating at 17,000 rpm. In this case, because of the high turbine pressure, the pump became the stress limited component.

The high pressure turbopump for the topping system is shown on the right. This is a rather large diameter unit, 6 stages, driven by a single-stage turbine. This design uses about 20 percent by-pass around the topping turbine for control. Our studies thus far indicated that it would be better, perhaps, to go to a higher by-pass because even though a slightly higher pressure rise across the pump is required, the reduction in annulus area at the exit of the turbine permits going to a higher speed which results in permitting substantially reduced unit size.

The following discussion of boost pump drives is divided into two parts (1) description of some of the boost pump drive systems and (2) showing some representative configurations.

As shown in figure 11, there are two basic types of drives that might be considered: (1) a gas drive in which the boost pump drive turbine is supplied by gas, either from a separate gas generator or from the main turbine exhaust; and (2) a hydraulic system in which a hydraulic turbine is fed by high-pressure liquid hydrogen, bled off the pump.

In the representative gas system, on the left of figure 11, the two turbines are located together, and it is assumed that a certain fraction of the main turbine gas could be ducted to drive the boost pump drive turbine. This, of course, would mean that the boost pump and the main pump would have to be located rather close together. Another way of doing it would be to have a separate gas generator supplying the gas for this turbine.

In the hydraulic turbine system, shown schematically on the right of the figure, the turbine flow comes back into the main system prior to going into the main pump. When the boost pump must be located remotely from the main pump, a hydraulic turbine system probably would be more desirable.

The effect of these two drive types on the system is shown in figure 12. The amount of additional bleed rate required to drive the boost pump is shown as a function of the ratio of the boost pump pressure rise to total hydrogen pressure rise. No effect is shown for the gas turbine because it is assumed that a certain amount of flow from the turbine exit is used to drive the boost pump resulting in no penalty. If a gas generator system were used, there would be a small penalty which is not shown in the figure. For the hydraulic turbine system considered additional power is required for the main turbine because it must pressurize more liquid hydrogen than it would otherwise. This means that there is a required increase in bleed rate, but such an increase is not very large. Thus, the penalty due to the inefficiencies of the pump and boost pump drive turbine is not large enough to result in any significant increase in the bleed rate.

There is, however, a major effect on the pump design if the hydraulic system is used. This effect is illustrated in figure 13 where, for example, hydrogen is bled from the pump at the 2000-psi point. If a split system is used, the hydrogen might be drawn off at the exit of the supercharger pump.

The ratio of the hydraulic turbine flow to fuel flow to the chamber is shown as a function of the ratio of boost pump pressure rise to total hydrogen pressure rise. In the 0.02 range, which would correspond to a 100-psi boost pump pressure rise and a 5000-psi overall pressure rise, 20 percent of the flow going through the supercharger pump has to be diverted to drive the hydraulic turbine. Such an increase in required flow has a major effect on the pump design.

Typical boost turbopump configurations that were evolved are shown, again at the 6,000 K level and 3,000-psi chamber pressure, in figure 14. The upper part of the figure shows the gas turbine-driven boost pump.

The pump is a two-stage unit having a pressure rise of 90 psi; the tip diameter is $3\frac{1}{2}$ feet; the speed is only a little over 2,000 rpm. The turbine utilizes about half of the flow that would come out of the main hydrogen drive turbine. It is a full admission, two-stage unit, about 3 feet in diameter.

The lower part of the figure shows the hydraulic turbine-driven boost pump. The two types of hydraulic turbines shown differ in the direction of discharge. The turbine is shown located at the tip of the second-stage rotor and is about 4 feet in diameter. It has about 10 percent arc of admission.

The reason the turbine is shown in two flow directions is to illustrate the problem of what happens to the hydrogen after exiting the turbine. If it is discharged at the pump outlet, it intensifies the cavitation problem of the main pump. The other approach, exhausting the flow back into the tank, has certain advantages, particularly if boiling fluid is being pumped. A certain amount of it would flash into vapor and could help pressurize the tank. In addition, it would tend to fill the void created by the hydrogen as it leaves the tank.

In summary, we have tried to give some ideas about the problems of the feed systems and the associated turbine drives for these hydrogen-fueled engines, particularly as the thrust chamber pressure is increased. We have shown that with increasing pressure level, the straight bleed system requires a prohibitive amount of by-pass. With the topping system, we are already going up in pressure and the topping system just adds that much more to it. More sophisticated advance arrangements should be considered. One such system suggested could be the supercharged topping. Admittedly it is more complex, and it does have additional shafting. However, it does have the advantages of bleed rates and pressure rises that stay reasonable. In addition, the associated components are more state-of-the-art.

Either the gas or the hydraulic boost pump drive system can do the job. The hydraulic drive has the advantage that it can be remotely located and is cold. However, a major problem requiring study is where to put the turbine exhaust flow. In addition, it presents a dynamics problem, particularly on start-up, that must be thoroughly studied.

[redacted]

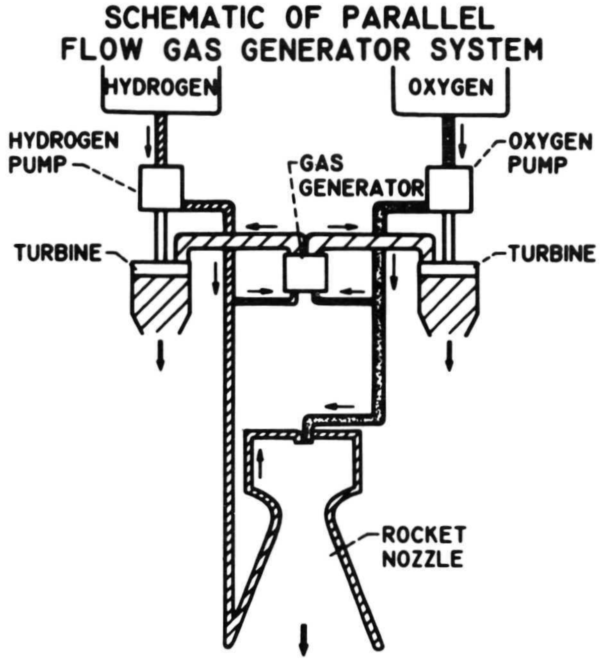


Figure 1

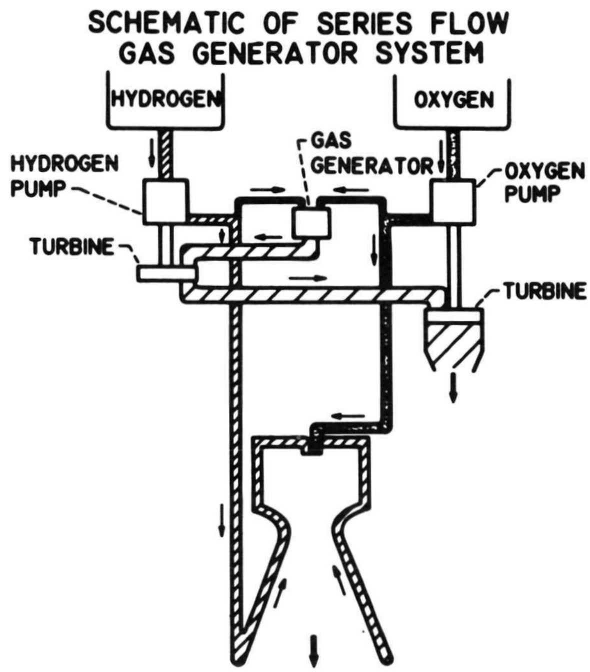


Figure 2

[redacted]

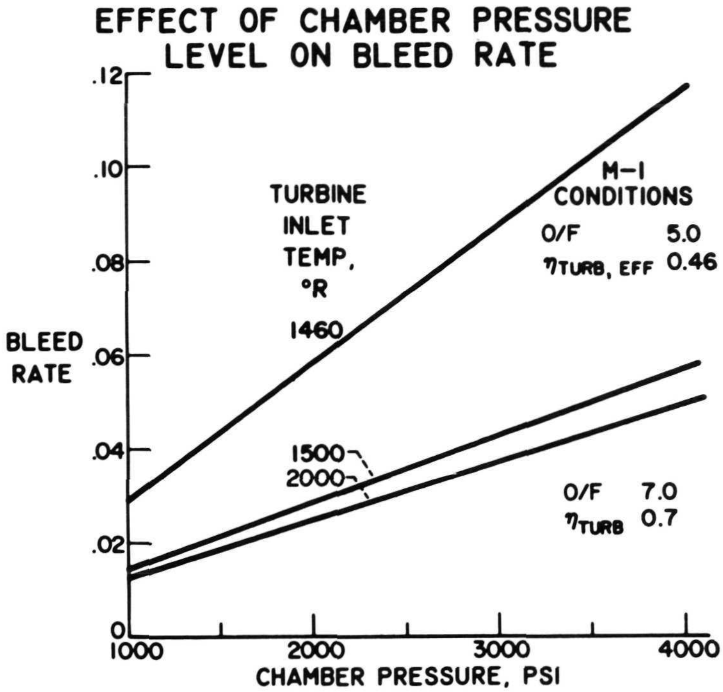


Figure 3

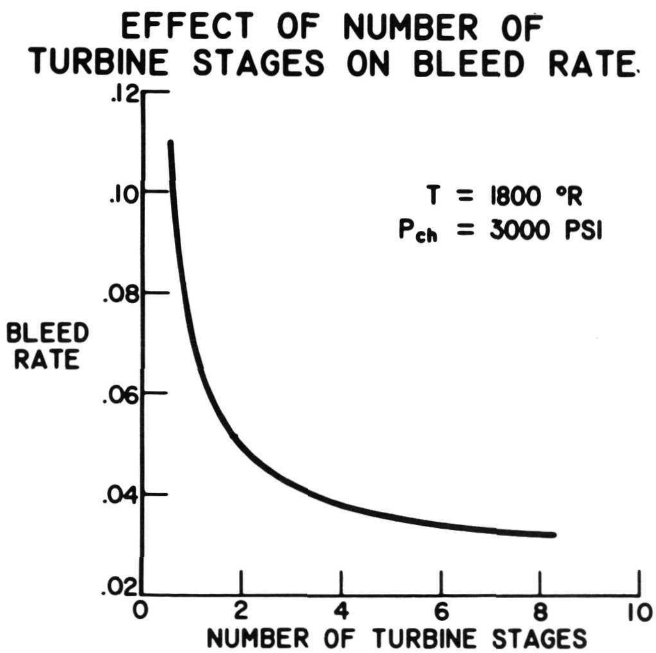


Figure 4

SCHEMATIC OF TOPPING SYSTEM

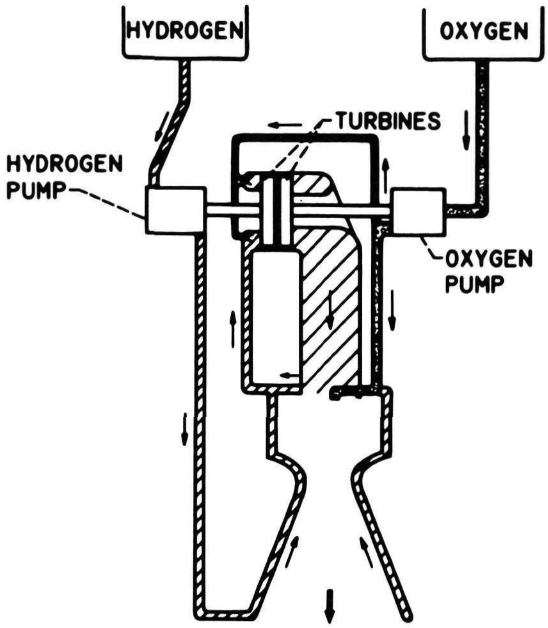


Figure 5

EFFECT OF CHAMBER PRESSURE LEVEL ON FUEL PUMP PRESSURE RISE FOR TOPPING SYSTEM

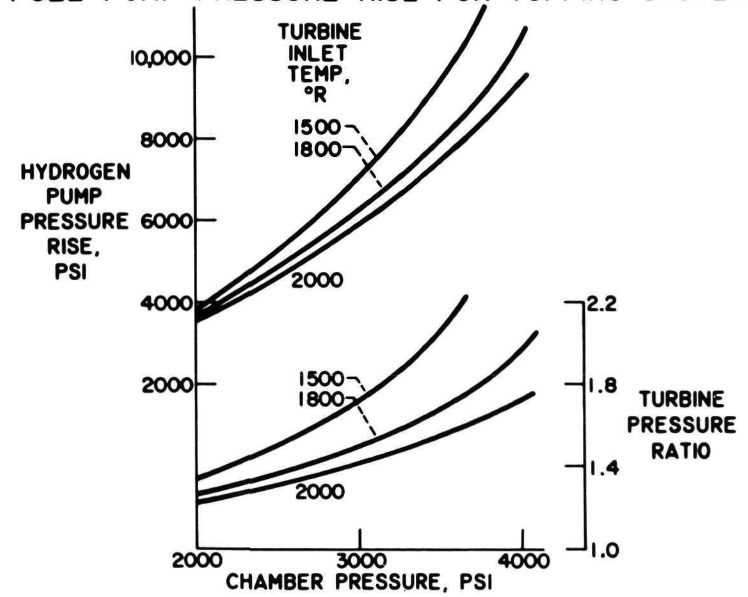


Figure 6

SCHEMATIC OF SUPERCHARGED TOPPING SYSTEM

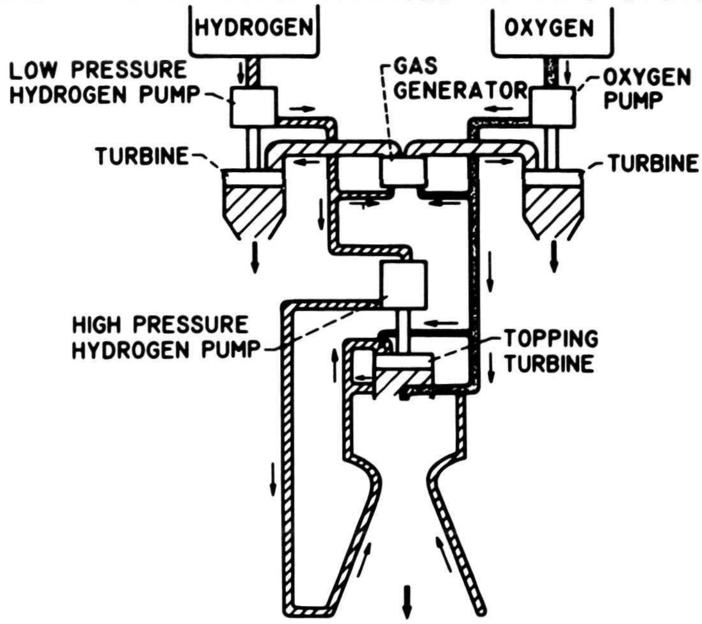


Figure 7

CHARACTERISTICS OF SUPERCHARGED TOPPING SYSTEM

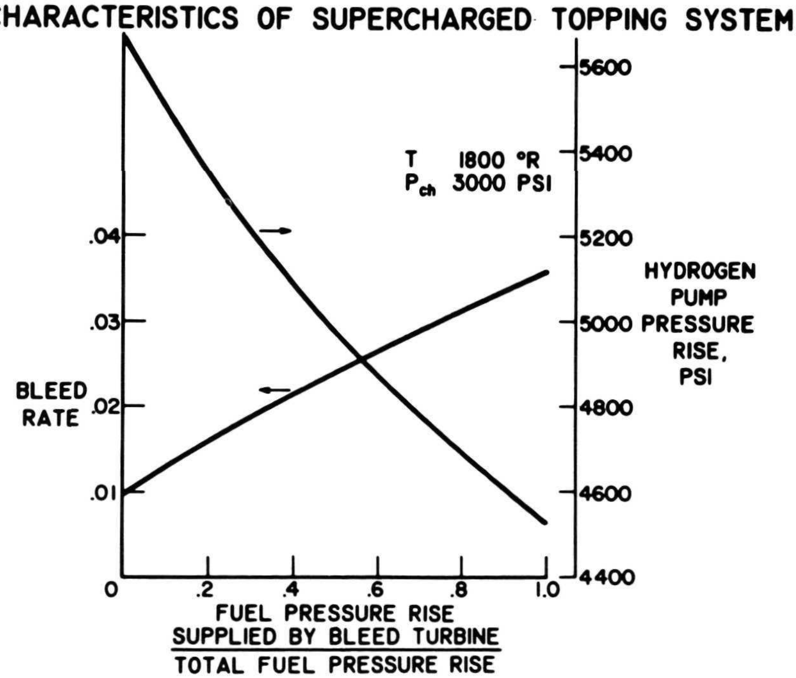
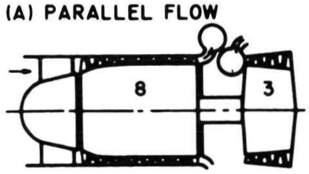


Figure 8

HYDROGEN TURBOMACHINERY CONFIGURATIONS
6000 K ENGINE; 3000 PSI CHAMBER PRESSURE

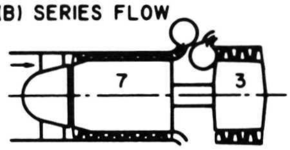
GAS GENERATOR SYSTEM

(A) PARALLEL FLOW



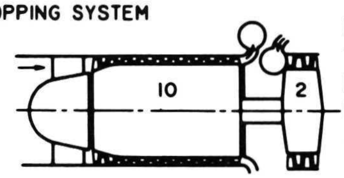
	PUMP	TURBINE
PRESSURE RISE, PSI	4500	-
TIP DIAM, IN	27	28.9*
RPM	11,000	
DN	2.3×10^6	

(B) SERIES FLOW



	PUMP	TURBINE
PRESSURE RISE, PSI	4500	-
TIP DIAM, IN	22.5	24.6*
RPM	13,000	
DN	2.55×10^6	

TOPPING SYSTEM

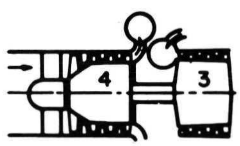


	PUMP	TURBINE
PRESSURE RISE, PSI	5700	-
TIP DIAM, IN	27	29.1
RPM	11,000	
DN	2.6×10^6	

*EXIT

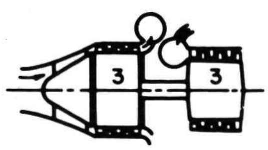
Figure 9

MULTIPLE HYDROGEN TURBOMACHINERY CONFIGURATIONS
6000 K ENGINE; 3000 PSI CHAMBER PRESSURE



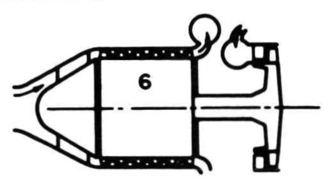
	PUMP	TURBINE
PRESSURE RISE, PSI	2000	-
TIP DIAM, IN	19.5	20.1
RPM	15,000	
DN	2.1×10^6	

LOW PRESSURE TURBOPUMP



	PUMP	TURBINE
PRESSURE RISE, PSI	2500	-
TIP DIAM, IN	21.5	18.7
RPM	17,000	
DN	2.6×10^6	

BLEED TURBINE DRIVE



	PUMP	TURBINE
PRESSURE RISE, PSI	3100	-
TIP DIAM, IN	27	29.1
RPM	11,000	
DN	2.0×10^6	

TOPPING TURBINE DRIVE

HIGH PRESSURE TURBOPUMPS

Figure 10

BOOST PUMP DRIVE TYPES

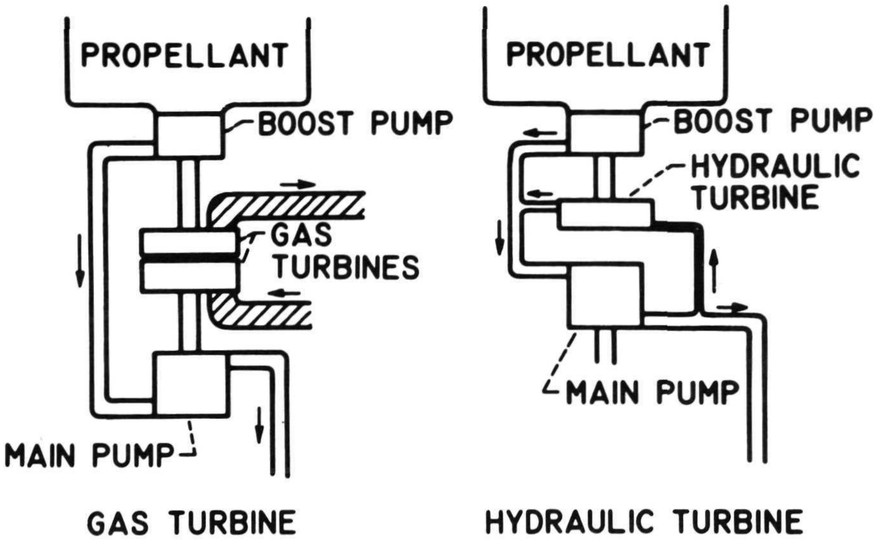


Figure 11

EFFECT OF GAS AND HYDRAULIC BOOST PUMP DRIVES ON BLEED RATE

6000 K ENGINE; 3000 PSI CHAMBER PRESSURE

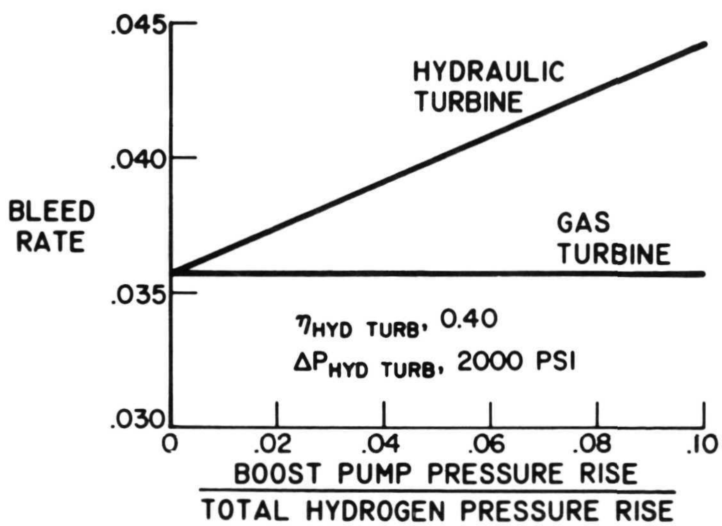


Figure 12

INCREASED FUEL PUMP CAPACITY REQUIRED TO SUPPLY HYDRAULIC TURBINE

6000 K ENGINE; 3000 PSI CHAMBER PRESSURE

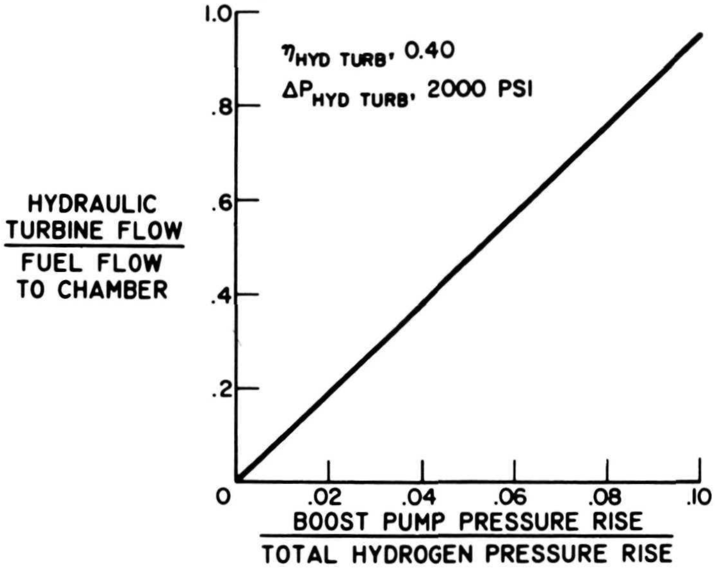
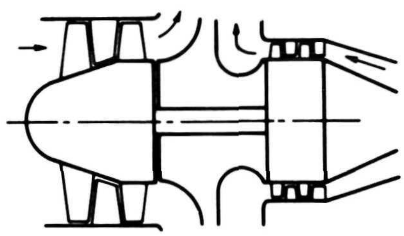


Figure 13

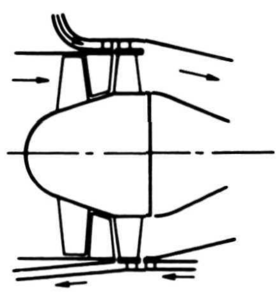
HYDROGEN BOOST PUMP CONFIGURATIONS

6000 K ENGINE; 3000 PSI CHAMBER PRESSURE



GAS TURBINE DRIVE

PUMP	
PRESSURE RISE, PSI	90
TIP DIAM, IN	43.8
RPM	2280
TURBINE	
TIP DIAM, IN	34
FULL ADMISSION	



HYDRAULIC TURBINE DRIVE

TURBINE	
TIP DIAM, IN	48.5
PARTIAL ADMISSION	

Figure 14

23. BEARINGS AND SEALS FOR TURBOPUMPS FOR LARGE ROCKET ENGINES

By Herbert Scibbe

Lewis Research Center, NASA

This paper will discuss

(1) the design and material factors necessary in rolling element bearings for operation in liquid hydrogen and other cryogenic fluids
(2) the present state of the art in current research and development programs
(3) some of the variables or factors that must be investigated both in rolling element bearings and fluid film bearings to run at the high DN (diameter of bearing in millimeters multiplied by the speed in revolutions per minute) values under consideration, approaching possibly 4 million
(4) state-of-the-art research and development in seal materials, and work that must be done in this area to run at the high surface speeds required for the turbopump systems of these large chemical rockets.

When high-speed turbomachinery can be successfully operated with the bearings lubricated and cooled by the cryogenic fluid, shaft seals and the lubricating systems can, of course, be greatly simplified. This, then, would result in a savings of weight, cost, and complexity of the turbopump.

Before a rolling element bearing can be successfully operated in a marginal fluid, such as liquid hydrogen or liquid oxygen, two principal problems must be considered. The first of these is the surface integrity in the ball-race contact which must be maintained. Therefore, the bearings should be made of materials that tend to exhibit minimum wear and maximum resistance to galling and surface welding in sliding.

The second is that the heat generated within the bearing must be removed to assure an equilibrium operating condition and to prevent loss of internal clearance that results in bearing seizure.

In a conventional lubricating system, using ball bearings, for instance, integrity in the ball-race contact is maintained and surface welding prevented by the presence of a contaminant film, or, in some cases, by a lubricant separating the surface asperities. The lubricating mechanism for bearings operating in fluids, such as liquid oxygen or liquid hydrogen, is quite different.

In liquid oxygen, surface oxides, or low shear films, are formed in the ball-race contacts. These surface oxides prevent the welding and galling between the metallic contacts of the bearings.

Operation in liquid hydrogen, however, is considerably more difficult since in a reducing environment such as liquid hydrogen these low-shear-surface films cannot be readily formed. To facilitate the formation of a transfer film on the balls and in the race grooves that would prevent the contacts of these clean, nascent metals, the bearings should be equipped with a retainer made of a self-lubricating material.

Such a mechanism is shown in figure 1. As the bearing rotates, the balls contact the retainer at these contact points. As the ball rolls, it transfers the retainer material to the outer and inner race in the form of thin transfer films. These thin transfer films prevent contact of these nascent metals and, therefore, prevent galling and surface welding.

To date, the best retainer material for liquid hydrogen service has been those made of Teflon-containing materials. These Teflon materials exhibit a low coefficient of sliding friction and have very low wear rates; both of these properties are desirable in this type of operation.

The second problem is one of removing the heat generated within the bearing. The major portion of heat generated within the bearing is due to spinning which occurs in the contacts between the balls and one race. Ball spinning occurs in all ball bearings operating under a thrust load.

In normal operation under thrust load, figure 2, the ball is spinning and the bearing operates with some given contact angle β, which is equal at the outer and inner race contact points. When the bearing is further operated at high speed the ball centrifugal force creates an additional load at the outer race contact. This changes the contact angle to one of unequal condition at the outer and inner race contacts. The magnitude of this difference in contact angle further increases ball spinning, and, consequently, more heat is generated in the bearing.

An analysis was made and programmed on a digital computer at Lewis to determine the effects certain factors have on the heat generation rates in a ball bearing. The same factors were then verified by experimental data from our bearing test program. Results indicate that to have low heat generation rates in a bearing with the ball spinning, there must be (1) open race curvatures (since the curvature is the ratio of the race groove radius to the ball diameter, and a value of 50 percent would be minimum, that would be the race contacting the ball at all points), (2) small ball diameter, (3) spinning of the ball at the race contact with the more open curvature (for instance, in the outer race), and (4) spinning of the ball at the race contact with the smaller angle (as illustrated in table I it would be the outer race) to keep spin velocities down.

The results of the computer program for two 40-millimeter bore bearings with different ball sizes utilizing two race curvature combinations are shown in table I. These are both 40-millimeter ball bearings; one has a 3/8-inch-diameter ball and the other a 1/4-inch-diameter ball. These are ball spin torques. The torque is obtained by converting from the heat generation rate by dividing through the shaft anular velocity. The values shown are for a thrust load of 100 pounds at a shaft speed of 40,000 rpm, or DN value of 1.6 million, at a contact angle of 10°.

In the 108 series bearings, spinning torques are higher in both cases, and the large value for the 108 series bearings with the 51 percent inner race curvature is much greater than that of the 1908 series bearing with the 1/4-inch ball for a value for 52 percent inner race curvature. This difference is the effect of both open race curvature and small ball diameter; these values are for ball spin torques at the inner race contact.

In the turbopump system for the Nerva engine, 50-millimeter bearings run at 24,000 rpm. Torque data (210 series) are plotted in figure 3. These bearings are to run at a rated thrust load of 2,000 pounds. A similar computer program was run to determine the effect of shaft speed on ball spin torques with various race curvature combinations. The 210 series, or the 50-millimeter bearing has a ball diameter of 7/16 inch, for a thrust load of 2,000 pounds at a contact angle of 20 degrees. The coefficient of sliding friction of 0.56, used is an arbitrary figure, so these are only comparative results.

These values are for the ball spin torques at the outer race contact. By increasing the outer race curvature to 54 percent, the ball spin torque has been decreased approximately 40 percent over the entire speed range. By further increasing the outer race curvature to 58 percent - that is, going to more open race curvatures - the spin torques are decreased a total of about 70 percent from the values in the upper curve.

At this point there is a change in control, or rather, a change from the ball spinning at the outer race to spinning at the inner race contact. This is due to ball centrifugal force, and because of the lower curvature more heat is created.

Several bearings of the 52-outer, 52-inner type have been run, and several of the 54 outer, 58 inner have been operated at 2,000 pounds thrust, at a shaft speed of 16,000 rpm for running times of approximately 90 minutes in the Nerva bearing test program. These were run with force-through feed of hydrogen flow.

Another major source of heat generation within the bearing is between the retainer and its race locating surface. In figure 4 is shown the cross section of a conventionally designed retainer with this heavier mass creating more heat at the locating surface here at the outer race as the bearing rotates. The new design is with an open thin-line retainer of lower mass. This design also permits better flow of liquid hydrogen through the bearing and, thus, improves the cooling efficiency of this type of retainer to more than that of the conventional design.

The roller bearings of the Nerva turbopump must support a radial load of 2,000 pounds. Some development work has been directed toward producing a cageless roller bearing. In the experimental design shown in figure 5, the load rollers are spaced and evenly separated by spacer rollers which are guided by flanges on the inner race. These flanges rotate with the inner race. The almost absolute absence of sliding here has eliminated quite a bit of heat that would be generated since all the components are of a rolling nature. This bearing is reported to have been run successfully for the Nerva bearing test program.

Table II is a report of data on the effect of engine size and rotative speed on ball-race loading. In the first three types of bearings listed - those for the Lewis ball bearing test program, the Nerva engine, and the M-1 engine - the DN value approaches 2 million, and as the ball diameter is increased, ball centrifugal force increases correspondingly.

The fourth engine listed in table II is a 6-million pound engine having a shaft size of approximately 10 inches. Running at a shaft speed of 16,000 rpm, a 4-million DN value would be approached. If this bearing were of conventional design, it would require, or would have, balls $2\frac{3}{8}$ inches in diameter, and at this speed they would produce a centrifugal force of 17,300 pounds. Of course, it is not feasible to design a bearing like this.

However, Mr. Gross has suggested that, disregarding the corresponding thrust loads, the 4-million DN value can be approached by decreasing the ball diameter and increasing the number of balls. For instance, in the table, the Rocketdyne engine, with a small 150 millimeter bearing, has 19 balls of 5/8-inch diameter, but the DN values are listed as 3.80×10^6 with a ball centrifugal force of 474 pounds. Perhaps for the 6-million pound engine, the same effect could be achieved by reducing the ball size to about 3/4 inch and using two dozen balls.

If the Nerva bearing and M-1 bearing were run at this shaft speed of 50,000 rpm, since centrifugal force is proportional to the square of the velocity, at 2:1 velocity ratio there would be a 4:1 increase in centrifugal force; and, if these were run at the same speed, there would be a 9:1 increase in centrifugal force. Table II illustrates that for approximately the same range of DN values different bearing configurations have correspondingly different values of ball centrifugal force.

It is important to note that there is little or no experience with rolling element bearings at DN values above two million. Therefore, any bearing applications which involve DN values at this figure and above are beyond the present state-of-the-art. Such applications will require research and development programs before reliable rolling bearings can be developed or designed.

It is not entirely certain that reliable rolling bearings can be designed for these extreme DN values. As can be seen in table II, the thrust load of the Rocketdyne engine was 2000 pounds. (This was some earlier work than that described by Mr. Gross.) The bearing failed and caused extreme failure of their test machine in the test immediately after the one for which these data are given. However, the Rocketdyne engine was a move in the right direction to get a rolling element to operate at such high DN values, if roller bearings can be designed to do the job. However, it may be necessary to consider, instead, fluid film bearings such as hydrostatic, or extremely pressurized, bearings.

The bearing test facility at Lewis, listed in the first line of table II, has been redesigned to test 40-mm bearings at a shaft speed of 50,000 rpm. This should result in the desired two million DN value; forced-through liquid hydrogen flow will be used.

There is a need for research and development work with ball and roller and hydrostatic fluid film bearings, in the size range contemplated for the large rocket engine. The variables that should be studied are listed in table III.

In rolling element bearings there are four factors to be considered. In the internal bearing design with ball bearings, ball size, race curvatures (which have been discussed somewhat previously), contact angles, the number of balls, and cageless bearings are important.

In the roller bearings, roller size, number of rollers, the internal radial clearance, and cageless bearings, which have been mentioned in connection with the Nerva program, must be considered.

The second major factor, retainer or cage design, has also been discussed somewhat.

The third factor is rolling element and race materials. To date, the only two adequate ball and race materials are either the old standby, 52100, or the 440C stainless steel. As mentioned before, the sliding coefficient of friction is around 0.56 for 440C stainless steel sliding on itself. Later tests in liquid hydrogen have shown that this in some cases approaches as high as a coefficient of 1.00. Therefore, new ball and race materials must be sought, and the factors to be considered are hardenability, good dimensional stability, especially at the lower cryogenic temperature, and resistance to wear and surface damage at liquid hydrogen temperatures.

The fourth item, new retainer materials, has been discussed somewhat in considering the reinforced, filled Teflon. A new type of material coming into view is the sintered metallics that are porous and contain either Teflon or other solid lubricants as, for instance, molybdenum disulfide.

In hydrostatic fluid bearings or pressurized bearings, an important item is the method of compensation, either orifice-compensated or capillary tube-compensated.

The flow requirements are critical because for a comparable size rolling bearing, the flows in the hydrostatic bearings, using liquid hydrogen, are much greater. However, the power consumption would be much less because the only requirement is support of the bearing on a fluid film, and the only power required would be to shear the film, considering only the viscosity effects. Also, because of the low viscosity, very large flows would be required through the bearing to support any type of load.

The effect of system misalignments, or the effect of system thermal distortion, would be a critical design factor, since these bearings are made with fairly close clearances. For L/D ratios of a 10-inch-diameter shaft, the L/D ratio would have to be limited to a value of 1. However, edge loading, as in journal type bearings, is a possibility.

The large chemical rocket engine under consideration is well beyond all existing knowledge as pertains to dynamic seals. Figure 6 shows a shaft-riding seal by which a dynamic sealing effect is obtained.

[redacted]

The secondary seal is on the axial face, and the axial load pressure or the static load is provided by a wave spring or load spring. The seal-riding member is tensioned by the garter springs.

In cryogenic applications, where low viscosity fluids or vapor change is present, the use of this seal probably would be prohibited unless it could tolerate some leakage across the seal face.

Figure 7 shows two types of face contact seals. The seal on the right has a bellows seal. The bellows acts as a loading device for the primary seal surface and also as a secondary seal. In this seal, the primary seal is loaded with a wave washer or some type of load spring, and the backup secondary seal is some type of O-ring.

Both of these seals, although providing the most positive sealing, have operating limitations that must be considered carefully for turbopump applications. Some of the problems with this type of seal, or any dynamic seal, would be warpage of the nose piece in the face contact seal; instability of the materials at the cryogenic temperatures; pressure balancing; selection of mating materials; and, finally, machinery stability. These problems can be expected to multiply and to become more severe as the size of the turbopumps is increased.

The size factor, aside from fabrication, may well be the stumbling block when bellows are used as the secondary seal. Bellows are very susceptible to vibration instability and wobble, and this instability can be initiated in several ways. Primarily, the vibration of machinery, itself, is enough to excite the bellows to the extent of fatigue or, perhaps, shattering of the nose piece.

It would appear, therefore, that a positive secondary seal with the capability of dampening vibrations would be required for these types of face contact seal to have successful operations at the high surface speeds. The selection of slider materials at these surface speeds will require further research. Existing information indicates that most materials have a limiting surface speed, after which wear progresses at prohibitive rates.

Furthermore, the heat generated at these high surface speeds is sufficient to initiate destructive exothermic chemical reactions, and such reactions have been experienced with carbon seals operating at surface speeds in excess of 350 feet per second.

For example, in a lox pump at Rocketdyne the dynamic seal was a carbon member impregnated with organic material which is not compatible with lox. Destructive exothermic chemical reactions resulted in the parts flying about 400 feet.

Figure 8 shows that material wear increases with surface sliding speed. With the three combinations of materials shown, increased surface speed causes an optimum in material wear to be reached after which any further increase in the speed increases the wear excessively. This might be called a characteristic wear curve for dynamic seal materials.

[redacted]

It is possible that the short duration that the seals must operate will dictate the type of materials to be used, and that existing information will provide a basis for the material selection. It should also be noted that wear particles cause separation of the sealing surfaces; hence, leakage of seals increases with wear rate.

In summary, the bearing and dynamic seal requirements of the high-speed turbopump systems for large chemical rocket engines are well beyond all existing knowledge. The speed requirements alone of a possible 4-million DN value for bearings and 700 feet per second sliding velocity for dynamic seals are approximately twice as great as the stage to which research and development programs have advanced the present state-of-the-art. The problems of maintaining surface integrity, heat generation, and material wear are more severe in liquid hydrogen for both bearings and seals, because of the nature of this fluid and because of the higher rotative speeds required by the hydrogen turbomachinery.

Even with the optimized design and development of new materials, reliable operation of rolling bearings cannot be assured at these high DN values; therefore, advanced designs of fluid film or hydrostatic bearings will have to be considered for the larger engines. Similarly, existing information on dynamic seals will provide only a basis for the seal design and selection of future slider materials. It is not certain that the surface temperature and wear rates can be tolerated at these high surface speeds, even for the short duration of time over which the seals must operate.

[redacted]

TABLE I

TORQUE ABOUT BEARING AXIS

T = 100 LB, N = 40,000 RPM, β' = 10°, f = 0.56

RACE CURVATURES	$\rho_1 = 0.51,\ \rho_o = 0.58$	$\rho_1 = 0.52,\ \rho_o = 0.54$
108 **SERIES** $d = 0.375$ **IN.**	2.095	1.653
1908 **SERIES** $d = 0.250$ **IN.**	1.831	1.412

TABLE II

THRUST BEARING CONFIGURATION FOR LIQUID HYDROGEN ENGINES

ENGINE	SHAFT DIAMETER, MM	SHAFT, RPM	THRUST LOAD, LB	DN VALUE	BALL DIAMETER, IN.	NUMBER OF BALLS	BALL CENTRIFUGAL FORCE, LB
LEWIS BALL-BEARING RIG	40	50,000	100	2.00×10^{6}	0.250	14	32
NERVA	50	24,000	2000	1.20×10^{6}	0.438	16	56
M-1	110	16,000	13,000	1.76×10^{6}	0.500	27	72
6000 K	250 (9.8425 IN.)	16,000	?	4.00×10^{6}	2.375	17	17,300
ROCKETDYNE	150	26,000	1000	3.90×10^{6}	0.625	19	474

[redacted]

TABLE III.- VARIABLES TO BE STUDIED IN DESIGNING IMPROVED ROLLING ELEMENT AND HYDROSTATIC FLUID-FILM BEARINGS

A. - ROLLING ELEMENT BEARINGS

1. INTERNAL BEARING DESIGN

(a) BALL BEARINGS: BALL SIZE; RACE CURVATURES; CONTACT ANGLES; BALL COMPLEMENT; CAGELESS BEARINGS

(b) ROLLER BEARINGS: ROLLER SIZE; ROLLER COMPLEMENT; INTERNAL CLEARANCE; CAGELESS BEARINGS

2. RETAINER OR CAGE DESIGNS

3. ROLLING ELEMENT AND RACE MATERIALS: HARDENABILITY; GOOD DIMENSIONAL STABILITY; RESISTANCE TO WEAR AND SURFACE DAMAGE AT LIQUID HYDROGEN TEMPERATURES

4. RETAINER MATERIALS: REINFORCED FILLED TEFLON; SINTERED METALLICS CONTAINING TEFLON AND OTHER SOLID LUBRICANTS

B. - HYDROSTATIC FLUID-FILM BEARINGS

1. METHOD OF COMPENSATION

2. FLOW REQUIREMENTS

3. POWER CONSUMPTION

4. EFFECT OF SYSTEM MISALIGNMENTS

5. EFFECT OF SYSTEM THERMAL DISTORTION

TRANSFER FILM FORMATION MECHANISM

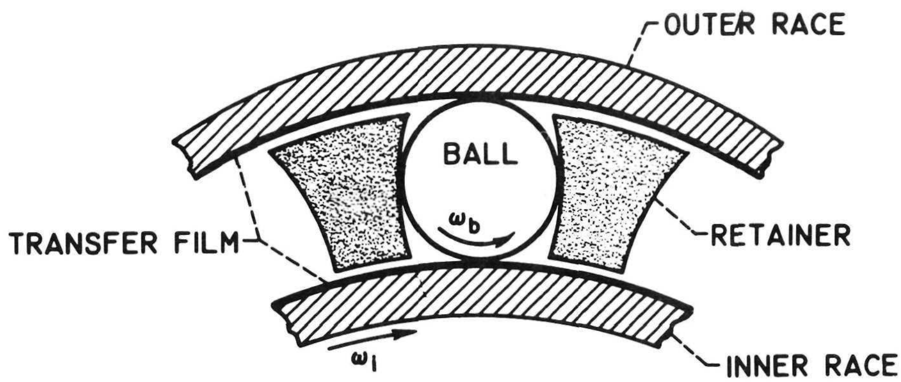


Figure 1

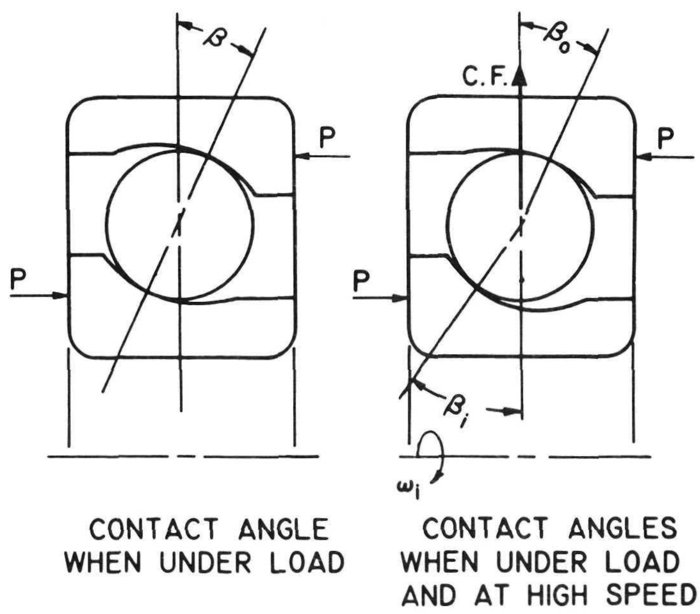


Figure 2

TORQUE DUE TO BALL SPINNING

210 SERIES BEARINGS (NERVA)

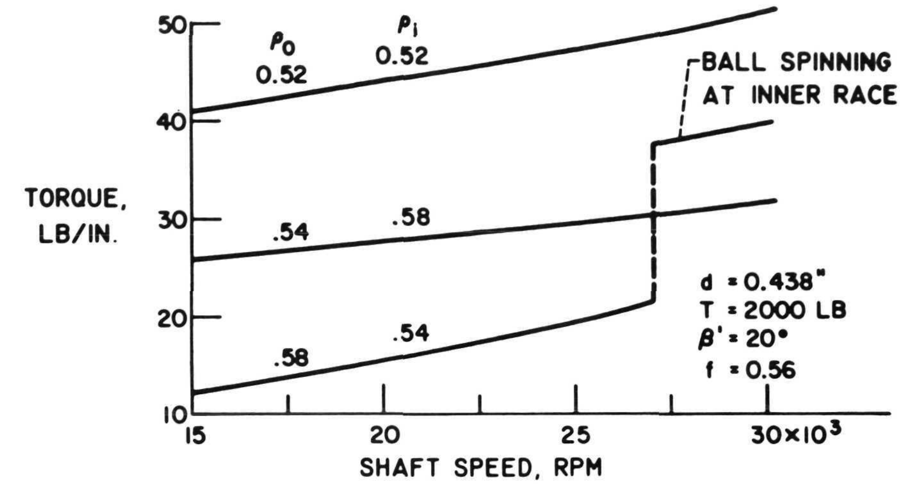


Figure 3

BALL BEARING RETAINER TYPES

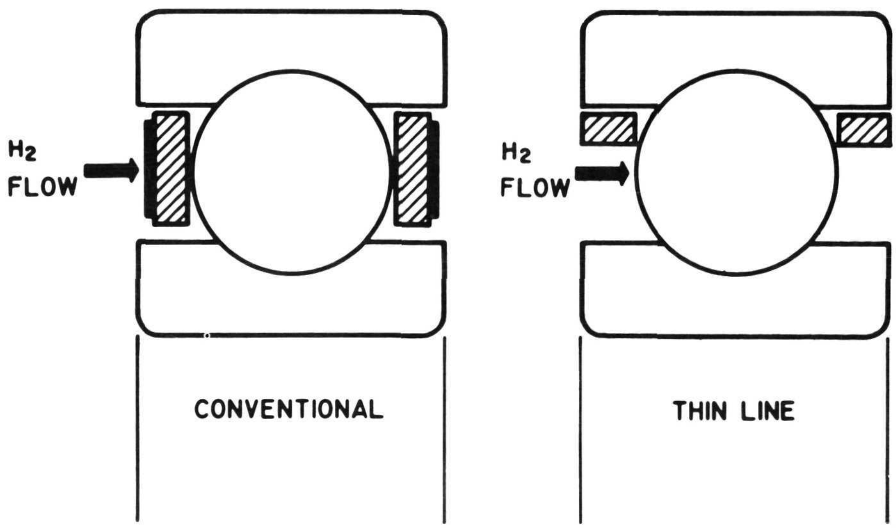


Figure 4

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EXPERIMENTAL CAGELESS CYLINDRICAL ROLLER BEARING

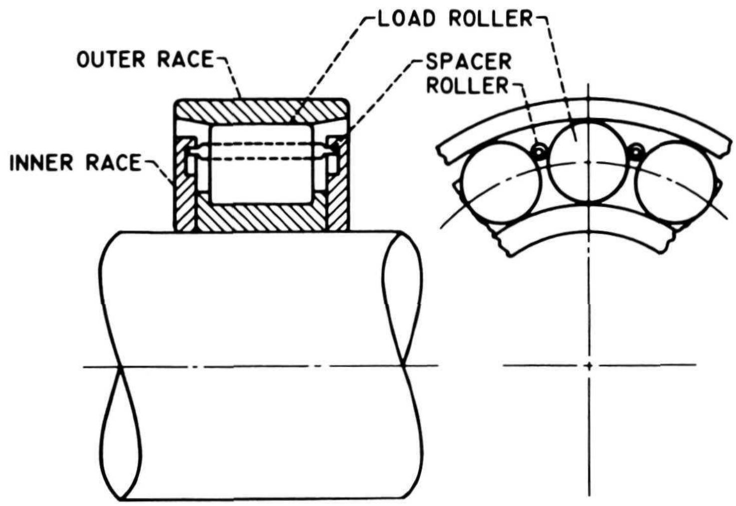
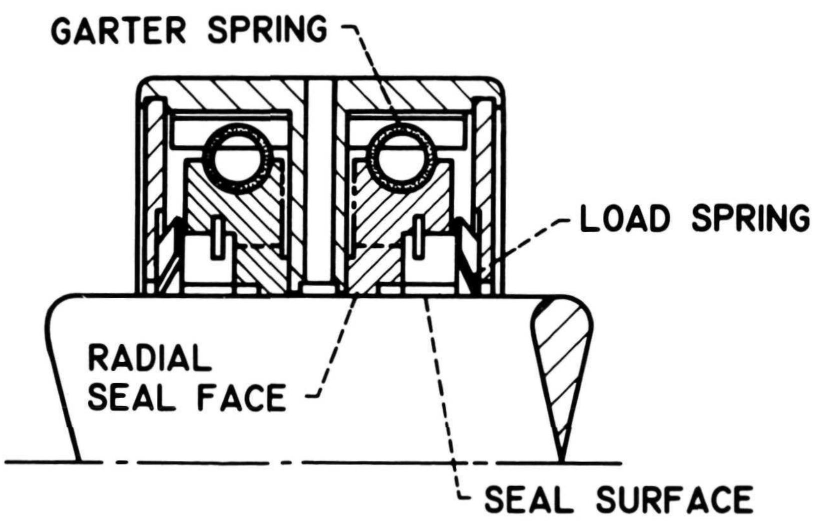


Figure 5

SHAFT RIDING SEAL



SEGMENTED

Figure 6

[redacted]

FACE CONTACT SEALS

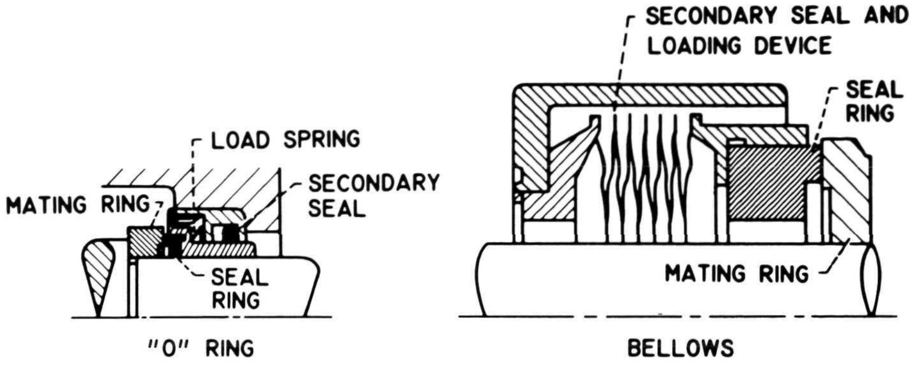


Figure 7

EFFECT OF SURFACE SPEED ON MATERIAL WEAR IN CRYOGENIC FUELS AND OXIDIZERS

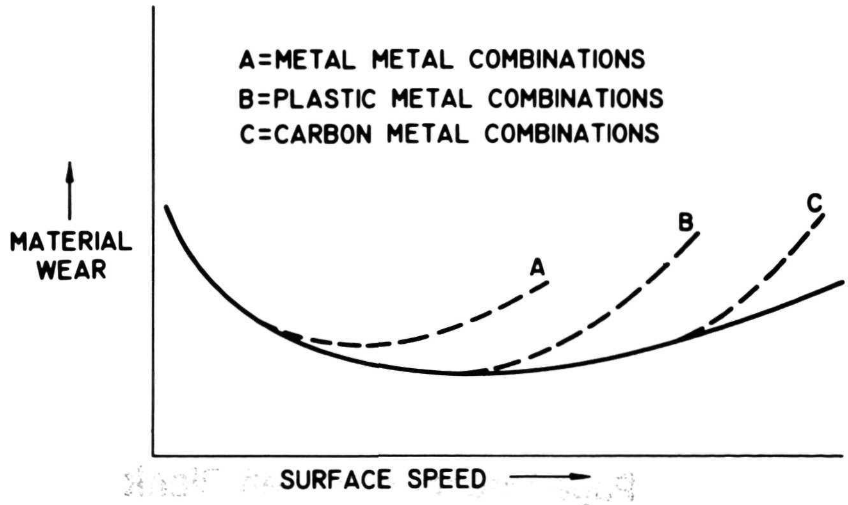


Figure 8

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24. SUMMARY DISCUSSION

MR. GINSBURG: This completes the session on pumping and flow systems. Important turbopump research and development problems can be summarized as follows:

I. Boost pump systems
- A. Multistage boost pumps - boiling fluid
- B. Drive system for high-pressure boost pump
- C. Boost pump dynamics

II. High-pressure turbopump
- A. Increased stage pressure rise
- B. Pump flow stability
  1. Design flow
  2. Off-design flow
  3. Cavitation
  4. Stall range
- C. Mechanical considerations
  1. Matching turbine to pump - high stresses
  2. Shaft thrust load - design and off-design flow
  3. Thrust balance devices - balance piston
  4. Bearings - ball, roller, and hydrostatic
  5. Dynamic seals - surface speed and materials

III. General - performance, reliability, and development
- A. Choice of turbine drive cycle
  1. Pump pressure requirements
  2. Turbine feed system flexibility
  3. Engine performance
- B. Turbopump system
  1. Size - development
  2. Single or multiple shaft - pressure and flow
  3. System complexity in relation to component state of the art

MR. HALL: I would like to ask you, did Pratt & Whitney report on initial study conditions during their high-pressure pump experiments? I am interested in how long it took to cool down and what the initial conditions were. Do we have any information on that?

MR. HARTMAN: Didn't they avoid the problem? Somehow they by-passed.

MR. GINSBURG: I am not sure of enough detail to answer that. I would expect that they avoided the problem by whatever was required to achieve cool down.

MR. TISCHLER: I have a very simple-minded question in the hope of getting a simple answer.

We have discussed a great number of possible put-togethers of turbines and pumps. Can you point the direction to a particular choice of those combinations that can be made at this time for a pump system for a large engine which will end up in the neighborhood of 2,000 or 3,000 psi chamber pressure?

MR. GINSBURG: A tough question. My personal opinion is that the general direction should be to achieve turbopump component reliability first, and if you can meet that, then you have the problem of evaluating the reliability of any system complexity that results.

It is our opinion that even with present flying engines and development engines that are trying to package cavitation and pressure rise together, the things are not particularly reliable.

Our concept is, as I see it, to separate characteristics so that we can arrive at a stable pump, and then do something else to meet the cavitation requirements which we have presented as a boost pump system.

MR. TISCHLER: That doesn't answer the question. I wasn't too optimistic about getting an answer at this time. Nevertheless, I think it points out the necessity for hitting this thing both analytically and experimentally to see if we can push the technology toward making decisions in this area. Otherwise, I am afraid we are going to be left with the problem of examining so many different configurations and systems that we can't get the problem really nailed down in the relatively short time that I would like to think is available to us.

MR. HARTMANN: Could I have comments on this question - does everyone else get the opinion that our contractors, the people we depend on for machinery, just aren't pushing very much? They are not coming up with new ideas or really grinding these into studies. They are sterile.

MR. TISCHLER: I agree.

MR. SLOOP: My question is somewhat related to what Mel Hartmann mentioned. It seems to me that you started off saying that 24 or 32 pumps were too many, and one was not enough. Yet you showed some configurations of booster pumps in the 6-million pound thrust stage. You showed a booster pump at the bottom of a tank. Have you thought about configurations where you have one booster pump for 24 million, and then the second stage divided into four parts of 6 million each? Or would you have four booster pumps in the bottom of a single -

MR. GINSBURG: I believe that our booster pump designs for this study were on the basis of 6 million.

MR. SLOOP: So, it was a one to one match?

MR. GINSBURG: Yes.

MR. SLOOP: You didn't think of an initial and four intermediate 6-million pound booster stages?

MR. GINSBURG: I think we did, but they got too big to think about. Seriously, the size of the boost pump is large even for 6 million. It was 40-odd inches. This size was entirely dependent on the state of knowledge of the

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thermodynamic effects of hydrogen. If in a reasonable period of time we can home in on the absolute maximum thermodynamic effect that we can safely design for, it is possible that this size can come down quite a bit.

For example, the thermodynamic properties that we were using were equivalent to 100 feet. It is possible that maybe 200 feet might be achieved which would reduce the size from 40- to 20-odd inches. In that case we could then think, I am sure, of the possibility of a one-boost pump system.

MR. SLOOP: The other comment had to do with Mr. Stewart's discussion of trying to reduce the hydrogen bleed by dividing his turbine into several parts. Isn't this swapping the devil for the witch, because the last speaker told us how difficult it is to provide bearings and seals. Every time you separate your turbine, you increase your bearing and seal problem don't you?

MR. GINSBURG: If you notice the curves on the charts, I believe you would have found in most cases the DN number dropped. Is this correct, Mr. Stewart?

MR. STEWART: Yes, this is correct. Actually, one of the points that I attempted to make in the summary was the philosophy of staying close to the state of the art and having more parts, rather than moving way beyond the state of the art and having less parts. The point here is that if you split the system, you do have more pieces, perhaps, but all of the components move back closer to the state of the art.

MR. SLOOP: This afternoon we will hear another use for bleed gas in the nozzle for thrust vector control, I believe, which you didn't cover.

MR. WEIDNER: I have one short point. I think somebody, rather critically said that our contractors are sterile in this area, and it appears to be that way from what we heard yesterday. I agree. We also have the duty of not just sitting here saying this and then going out and impressing our thoughts on them What are we going to do with whatever new ideas we have here to make them follow them and adopt them, possibly, and still to trigger, to initate more thinking by the contractors?

MR. HARTMANN: I was going to follow that up but someone else got the floor. We have to think about how we can do this. I think a part of the problem is that certain of our hydrogen pump information, for example, that we consider confidential, since it is design type information, also contains certain ideas and concepts. The working level people in the engine companies have this information; however, it is doubtful that their management accepts these ideas when presented by their own people. Somehow we have to bat these ideas back and forth similar to the way people have with different kinds of thrust-generating devices so that they are willing to get in and work at them.

MR. SLOOP: I fail to see the correspondence between classification and getting management to accept it. Would a conference in which we were fairly critical of present techniques be one way of doing this?

MR. GINSBURG: No.

MR. TISCHLER: Maybe I can answer that question. I think we had yesterday, in particular, some considerable exposure to the sales treatment, which is advocated by the company's management, of presenting information on a future system. It is never going to be possible to filter this out completely.

We have had this morning at least an approach to the technical problem and issues. These are, I think, quite separate. I believe that Mr. Hartmann is right in arguing that management, as it presents its ideas to us on future systems, has filtered the technical problem down to the point of obscurity. I think this is basically what he is trying to say in saying that he believes the companies are sterile.

I think he countered his argument, or even corrected his argument, by saying at the working level within some of these companies it isn't completely sterile. What we must do, I believe, is bring forth the ideas at the working level and develop some of these, at least the good ones. It is going to be our job to decide which are the ones that ought to be brought forth.

MR. WEIDNER: Then, we have to wave the green flag, not the red one.

MR. BEHEIM: I wonder if you would care to comment further on the tip turbine drive that was directed from the standpoint of whether you think this is a mere novelty, so to speak, or does it really show promise from the standpoint of leakage, efficiency, hub tip ratios, and such?

MR. STEWART: We at Lewis have not experimentally investigated turbines of this type. However, we have made in-house studies of diagram characteristics, determining arc of admission, and so forth. They do appear to be more than a novelty. They appear to have potential.

In particular, I might cite the Aerojet work which I mentioned before, which we looked at critically because we were interested in the efficiency levels that might be achieved from these tip-type units. Based on their experimental work, the efficiencies were reasonable.

We think that the main problem here is probably not so much one of leakage or performance of the turbine but one of putting such a turbine into the system. As I mentioned before, there are two basic problems: (1) where does the turbine flow go, and what is the subsequent effect on the adjacent part, either the next pump or the tank; and (2) what are the dynamic characteristics when you try to drive a turbine with this high pressure liquid that is fed back from the pump. It is very attractive, though. Its advantages of being cold and of being the possibility of remotely located are very strong.

MR. GINSBURG: Probably the biggest problem to get at for the boost pump and its drive will be the dynamic relation to the main pump. In order to get at the transient problem of the bootstrap turbopump with the rest of the main pump you really have to have a system.

You can approach it analytically, but the real proof of the pudding is the system. This gets to be a costly technology program.

MR. STEWART: I might add one more point in terms of the start-up problem. Under steady-state conditions you match, of course, the torque between the turbine and the boost pump. However, on start-up, since the turbine is being driven by liquid that comes from, say, the exit of the main pump, there is a pressure lag. This might become a real problem, particularly if you try to start up quickly. This lag might be such that you actually starve the boost pump before you get enough power to the turbine in order to speed it up. There are methods, I would think, for alleviating this problem. For example, you could actually pass more than the required flow through the turbine to begin with to create an excess torque. Such approaches could be evaluated in transient studies.

MR. WOODCOCK: I would like to make one small point, that elimination of a tank pressurization system is not categorically desirable. It may be that allowing the tank to fill with this vapor would be a serious penalty on the system. As a matter of fact, I think that is the case on Saturn 5. It would be possible in principle, at least, to operate the locks pump on the F-1 without a tank-pressurization system because of the enormous size of the vehicle. I am not advocating doing this. I don't think it is practical.

MR. GINSBURG: Your point really just eases the boost pump problem.

MR. WOODCOCK: Yes.

MR. GINSBURG: I think the boost pump serves two purposes: (1) it helps solve tank pressurization, and (2) it is designed to allow the high pressure pump to have freedom of design, to be, for example, stable.

MR. WOODCOCK: I realize that.

MR. HARTMANN: If the tank is filled with cold gases, I think the objective would be to reduce the pressure, and the density of the tank, to reduce the residuals. Whereas, if they are filled with warm gases, the pressure level is not so critical, the density is reduced by virtue of the temperature. It needs to be tried in the system and looked at rather than our arguing the pros and cons on the surface of it. We recognize the point.

MR. WILLIAMS: Even that can be carried one step farther. Tremendously high pressures in the tank are not desirable because they necessitate large wall thicknesses and this sort of thing. In the large diameters that we are getting into, in the Nova-type vehicle, whether multicell or cylindrical tank, actually the pressure helps from a structural standpoint.

Some pressure is necessary to help give a rigid structure. This pressure should not be too high or the resulting structure will be too big. There is one more iteration into the system where it really helps. I think I could say, without fear of being incorrect, that in any of the systems we looked at, zero pressurization requirement or zero pressure, so to speak, in the system, would cause weights definitely to go up in the over-all vehicle itself, because of just plain structural requirements of the tank.

MR. PAUL: If we operate gases at a saturation temperature, and we have sloshing motions, I think we are very much susceptible to pressure decay. It is probably advisable to raise the temperature to avoid these pressure decays, in addition to the weight savings.

For oxygen containers, we use temperatures I would say at 100° F. This is an optimum temperature, and for liquid hydrogen these temperatures are even lower. They are -200° or -300° F. But still roughly 100° or 150° above saturation temperature.

MR. GINSBURG: That completes the session. Thank you.

MR. SLOOP: Thank you, Mr. Ginsburg. I think you brought up a lot of interesting questions, a lot more than we have had time to deal with in our short discussion. I do think it does bring out the duty of all of us, when we generate information, to communicate what we are trying to influence. In this conference we are communicating among ourselves. We need to communicate this knowledge to the contractors as well. We do that sometimes in our written reports which often get pushed to the sidelines as we race along in this rapidly progressing technology. But, I do want to emphasize the need to write reports, to travel, to talk, and to communicate in any way that you can, putting these new ideas, this new information, out where it will do the most good.

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SESSION VI

NOZZLES AND NOZZLE CONCEPTS

Chairman: Milton A. Beheim
Lewis Research Center, NASA

[redacted]

25. INTRODUCTION

By Milton A. Beheim

Lewis Research Center, NASA

The preceding papers were interesting discussions on advanced nozzle concepts, but I am sure some of us feel they glossed over some of the existing problems.

They served satisfactorily, though, as an introduction to this session; they told why we are interested in advanced nozzle concepts. The subjects that we are covering in this session are the matter of altitude-compensated nozzles, air augmentation, the ability to predict performance of these nozzles, and means of optimizing nozzle contours.

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26. NASA ALTITUDE-COMPENSATED NOZZLES

By James F. Connors

Lewis Research Center, NASA

This paper will be a further discussion of advanced nozzle concepts. In the turbomachinery session there was some discussion of lack of ideas among the contractors. Such is not the case in the nozzle area. In fact, it is my concern that they are given an uncontrolled rein without being required to pay any substantial heed to the basic problem of nozzle aerodynamics. I fear that we, at NASA, are not exercising enough control over the contractors and that they are not receptive to NASA inputs.

However, a problem of communication exists more than just between NASA and the contractors. It exists within NASA, in that I have yet to meet any of our present project managers, and discuss the work that has been done at Lewis in the area of unconventional nozzles.

From the presentations yesterday it was evident that nozzle aerodynamics do not seem to be what they were in the turbojet days. Apparently altitude compensation is thought by the contractors to come by some exotic, sketchy, superficial treatment. However, I do not believe that to be the case. I think it is something that must be worked at, and we must consider the basic aerodynamics, the detailed flow processes that go on within the nozzle.

I was appalled by a statement by one of our contractors, Rocketdyne, who pointed out that venting is not the engine contractors' concern, and that they could simply assume that base pressure equals ambient pressure. Thus, they would assume altitude compensation, and that the vehicle people would then worry about getting the external air into the nozzle. This is not the case!

When we are talking about altitude compensation, we are talking about a complex flow mechanism, one that regulates the expansion process over a wide range of conditions. I think we have to look at the flow of the total configuration in considerable detail.

I cannot emphasize too strongly that there is a need for basic wind-tunnel evaluation of the various acceptable proposals. We need to come up with a consistent set of criteria, testing techniques, and modeling. I think it is important that we judge these configurations on a standard frame of reference.

Those of you who have done experimental work in nozzles know that it is very difficult to make good thrust measurements. Also, such testing must be done in the wind tunnel because, I think, stream effects are very important in determining true altitude-compensation capability.

Static-firing tests at sea level and altitude are not enough, because, when we impose the requirement of ventilation, we also have to be concerned with base pressure effects. We have to be concerned with the jet interaction with the free stream, or the entire process of venting.

Lewis has been doing some pioneering work in this area since 1958. In some of the current contractors' configurations I have not seen any reception of our viewpoints or any significantly good nozzle data. I hope Clark Hawk is about to correct me on this in his paper about some of the current work.

Instead of going to systematic cold-flow wind-tunnel testing and truly evaluating nozzle aerodynamics, we are too quick to build a 20,000- or 50,000-pound engine and fire it. When that is done, the whole problem is complicated, and it is difficult to come up with any significant evaluation of the nozzle. It is superimposing the whole combustion process on top of the nozzle aerodynamics.

I do not want to belabor the Lewis work that has been done. As I said, it was done some time ago, but, for introductory purposes, I will touch upon some of the viewpoints and conclusions that we have made in this area.

Currently there is interest in looking beyond the Apollo lunar mission and considering vehicle requirements for more ambitious manned interplanetary flights. Mission analyses have indicated that, for such flights, payloads in earth orbit in excess of 1,000,000 pounds will be required. This reflects back into boost-vehicle requirements of thrust in the range of 20 to 30 million pounds.

For this class of vehicle, a number of individual or segmented engines will undoubtedly be employed, rather than the alternative of a single huge chamber. With present engine technology, this number would probably be 16 or 18 as a minimum. This situation provides the vehicle designer an opportunity for considering unconventional approaches to the overall powerplant and booster configuration.

In order to review the various proposals with proper prospective, it would be appropriate at this time to review the Lewis studies. Figure 1 illustrates equivalent nozzle systems for a given large thrust application which really show the evolution of the Lewis annular nozzle concept. To achieve a given large thrust, three possible approaches may be considered. A single large nozzle may be built to produce this thrust or a random cluster of smaller engines. If geometric similarity in the nozzle is assumed, for example, the clustered configuration can be achieved in a much shorter geometry. There is also the third possibility of taking these individual nozzles in the cluster and arranging them in a circular pattern and, then, integrating them into one annular geometry, maintaining the short length advantage of the cluster configuration. However, with the annular geometry, the center portion may be vented to eliminate any base area and the consequent possibility of base heating or base fires. This is illustrated in the bottom sketch.

In the past, the Lewis Research Center has contributed heavily in the areas of base heating. Single and clustered engine configurations present the problem of the exhaust jet from the nozzle interacting with the external stream or with the jets from adjacent nozzles, causing hot gases to flow forward into the base area with the possibility of attendant base fires. Obviously, a vehicle must be designed for this condition and, to protect against it, a heavy heat shield must be provided, although this involves a consequent weight penalty. The advantage of going to the annular geometry thus eliminates the possibility of base heating, and, in effect, a refined base bleed configuration is achieved. In concept, of course, the combustion chamber could be a complete annular combustor, a segmented combustor, or a multitude of individual burners that would feed into one common nozzle. Unit combustors, of course, offer some potential developmental economies. The main objectives of the Lewis study were primarily vehicle and nozzle aerodynamics.

In figure 2, a further refinement of the annular nozzle is considered. That is the provision of utilizing external expansion and thus achieving altitude compensation. Through the use of a free jet boundary, the expanding supersonic stream automatically adjusts with altitude. In this figure are shown three conditions - below, at, and above design pressure ratio. Enough internal expansion was designed into the nozzle to achieve full expansion of the flow at the lift-off condition. External expansion was built in to accommodate the range of altitudes for a representative boost trajectory. As with the plug nozzle, the flow adheres to the surface and follows the contour at below-design pressure ratio. As the nozzle pressure ratio is increased, the flow continues to expand until, at design pressure ratio, the nozzle is flowing full and discharging in an axial stream. At pressure ratios above design, the flow continues to expand and actually flares out from the nozzle, as indicated in the lower sketch. Perhaps the most important aspect of this design is the recognition that it is the maintenance of near-ambient pressures at the external lip that provides the regulation of the expanding stream. Severe performance losses will be incurred by configurations wherein low base pressure is encountered at the cowl lip. This might occur, for example, with a conventional plug nozzle installed in a very large base area or in the case where the cowl lip is designed with a very blunt edge so that the flow overexpands with a consequent reduction in the base pressure. This would also occur on the inside-out version of the plug or the so-called E-D nozzle where there is no bleed in the base area and where the lip pressure is aspirated to some fraction of the ambient pressure.

To explore the performance potential of this type of annular nozzle, employing internal-external expansion, two configurations were designed and are illustrated in figure 3. These nozzles were investigated in the 10- by 10-foot supersonic wind tunnel, both in quiescent air and in supersonic streams at Mach numbers of 2 and 3. Cold air, pressurized to 600 psia, was used to simulate the jet flow. One of the nozzles was designed for an area ratio of 15 and was essentially in the style of the conventional plug nozzle with the center area vented to ambient pressure by means of an annular intake. In this case, of course, the purpose of the center bleed air was only to reduce the base drag and the regulation of the external expansion within the nozzle was controlled by presenting a minimum lip angle to the external stream as it flowed over the cowl. The annular entry of the bleed duct was practically flush with the simulated fuselage and was designed to admit low-energy boundary-layer air to minimize duct losses. The other nozzle had an area ratio of 25 and was essentially an inside-out version of the previous nozzle. In this case, the external expansion ramp was actually external to the simulated jet. Here, the annular inlet was considerably larger than the simulated fuselage and supersonic flow flowed through the center of the annular nozzle.

Some of the rationale, or philosophy, that was incorporated into the design of these bleed ducts included such considerations as an air-breathing center stage or air augmentation (as in a ram rocket) wherein fuel could be injected into this center bleed air so that ramjet-type impulses would actually be attained. Another possibility, particularly true with larger annular configurations, was that the annular nozzle could be designed as a ring wing or Busemann-type airfoil and an essentially zero drag configuration attained. The experimental setup in the wind tunnel was unique in that it provided three-component force measurements and had cold high-pressure air to emulate the jet. This facility is available for future investigations.

In figure 4, are shown the essential results of the investigation. Here, we have compared the $\epsilon = 25$ annular nozzle performance with an equivalent-area-ratio C-D nozzle and an $\epsilon = 8$ C-D nozzle. We have assumed a representative altitude schedule with a 600 psia chamber pressure and converted these parameters and the actual test data into thrust coefficients. The $\epsilon = 25$ annular nozzle showed true altitude compensation and maintained a high thrust level - 98 percent of ideal - all the way up to design altitude. At takeoff, of course, with an equivalent-area-ratio C-D nozzle, there was a 20-percent reduction in thrust due to overexpansion losses in the C-D nozzle. The shaded areas represent the margin of performance gains to be had with the annular altitude-compensating nozzle. Compared to the $\epsilon = 8$ C-D nozzle, its nearest competitor, the annular nozzle showed a 4-percent integrated-thrust improvement for the booster altitude schedule illustrated on the figure. A further translation of this performance data into terms of overall payload lifting capability will be given in a subsequent table.

Figure 5 is a sketch of a more practical configuration - one that is not subject to the expediency of the test as were the configurations viewed at this point. Three flush bleed inlets to admit air into the center portion of the nozzle are illustrated. This might be representative of a structurally, more attractive configuration. Also shown is the jet canard which has been under study here at the Lewis Research Center where an attempt was made at combining reaction and aerodynamic control. The "penshape" nozzle, which was investigated for this application, lends itself to unusual packaging arrangements and incorporates the advantages of external expansion, and thus altitude compensation.

A sketch of a large-area-ratio annular booster engine - one that utilizes the center bleed as the mechanism of providing altitude compensation - is shown in figure 6. This configuration is representative of an experimental program that was underway at the Lewis Research Center but was subsequently put aside. This program was aimed at determining the possibility of using a very large area ratio (something approaching 150 to 1) and using minimum amounts of center bleed. One of the main parameters of the investigation would be to determine this minimum center-core bleed flow. This configuration, of course, would be highly attractive in the concept of single stage to orbit. As shown here, the center would have to be vented and the pressures at the internal lip maintained at near ambient pressure. In this configuration, it is proposed to have small external scoops draw in boundary-layer ambient-pressure air and admit it to the cavity - again maintaining conditions near ambient. Should the concept of single stage to orbit prove to have some merit, a program such as indicated here would certainly be necessary and the Lewis experimental facility would be particularly attractive for this purpose.

Some of the factors presented in favor of going to an unconventional engine approach are:

1. Altitude compensation
2. Elimination of base heating
3. Developmental economics
4. Nozzle length (and weight) reduction
5. Vehicle stability and control
6. Vehicle drag reduction.

The first one has been dealt with in rather great detail and is that of altitude compensation - a significant increase in specific impulse through improved nozzle performance. Also discussed was how venting the base eliminates the problem of jet interactions. Thereby, the base heating problem can be eliminated and the consequent penalties of a heavy heat shield avoided. Another apparently major point in the argument is that of developmental economy - one that favors working with a small component of the combustor and developing that to its fullest without having to work with the complete powerplant system. This apparently can reduce the facility and developmental cost by some very significant factor. By going to the annular nozzle configuration, geometric and packaging advantages are achieved in that length is reduced significantly and presumably this can be reflected as an overall weight reduction. There is also a structural advantage in going to the annular geometries in that the thrust can be directly transmitted from the powerplant to the outer fibers of the vehicle. Presumably, this offers increased vehicle structural stability and control. The last point, probably a minor one, is that the vehicle component base drag can be eliminated through the use of base bleed.

Table I summarizes the performance potential that exists through the use of compensating nozzles. In this table, are identified the altitude-compensating nozzle as an advanced nozzle; in this table are considered only the impulse gains to be had through changes in chamber pressure and nozzle area ratio, and through the use of an advanced nozzle as opposed to a conventional C-D nozzle. Two types of vehicles on different boost missions have been considered. The first three cases are that of a multistage vehicle to escape velocity at an orbit of 150 miles. A single stage to orbit at 150 miles was also considered. In the multistage vehicle to escape, increasing the chamber pressure from the contemporary level of 1,200 psi to 3,000 psi at comparable area ratios increases the payload by 6 to 8 percent. Now, if the nozzle configuration is simply changed from a conventional to an advanced design, the payload gains may be on the order of 4 to 5 percent. Again, with a multistage configuration, the most significant improvements in payload lifting capability (13 to 25 percent) are to be found by going to large area ratios in the upper stages. (These are nozzle area ratios that are limited by the vehicle envelope and having values as high as 200 to 1.)

Another mission that was examined is the single stage to orbit. Here, it is quite evident that as the difficulty of the mission is increased, i.e., the velocity increment that will be required for the particular stage is increased, the gains to be had by increasing chamber pressure or going to an advanced nozzle are far more significant. For example, in going to an advanced nozzle at a chamber pressure of 1,200 psi, payloads can be increased by 68 percent. However, if the chamber pressure is at 3,000 psi, going to an advanced nozzle will yield a payload gain of about 17 percent. The gain in simply changing the chamber pressure from 1,200 to 3,000 psi amounts to about 83 percent. These figures represent only the payload gains as a result of engine performance and do not take into account the seven arguments listed favoring the unconventional approach. Such considerations as logistics, reliability, handling, and the costs of developing such a system, of course, will carry a lot of weight in the argument as to which direction we should go for manned interplanetary missions.

When the required engine development times and costs are taken into consideration, the most appropriate direction for the large boosters would seem to be the utilization of the existing (F-1, M-1) engine technology in combination with an advanced nozzle. This, in effect, means specifying the engines up to the circular throat and providing design freedom in shaping the divergent portion of the nozzle. Then, these internal-expansion nozzles would discharge into a common altitude-compensating high-area-ratio nozzle shroud. By configuring the vehicle for venting, the advantages of altitude compensation and the elimination of heavy base heat shields can yield significant payload gains. Chamber pressures near 3,000 psi, will require a new cooling technology (other than regenerative) and correspondingly a much longer development time.

Since the completion of the Lewis study, many varieties of unconventional nozzles (fig. 7) have been proposed and investigated. Notably, Dr. Rao of the North American Rocketdyne Corporation presented the concept of the E-D nozzle and, without supporting data, indicated that base bleed would not be required to achieve the altitude compensation that was claimed for their particular nozzle design. Since that time, much effort and money have been invested in proving what should have been learned from the Lewis studies, namely, that altitude compensation for this type of configuration can come only through the use of center bleed, or venting.

There have also been the advocates of the reverse flow (RF) and the horizontal flow (HF) nozzles which are really varying degrees of folding the combustion chamber back into the nozzle as illustrated by the lower sketches. These particular configurations look mighty attractive on paper in that they reduce nozzle length considerably; however, in reality, the mechanism of providing the ambient pressure environment at the internal lip (or the point of external expansion) appears virtually impossible - and thus would not achieve altitude compensation. Altitude compensation is not important for upper-stage application, in that all the nozzles are presumably underexpanded over their whole range of operation. Therefore, some packaging advantages may exist in going to annular geometries - either annular C-D or any of the folded configurations that we have indicated on this figure.

Utilizing the available 8- by 6- and 10- by 10-foot wind tunnels, the Lewis Research Center has planned a rather extensive program for the investigation of plug, annular, and ejector nozzles. Test conditions simulating sea level static operation and altitudes to 160,000 feet with Mach numbers up to 3.7 can be obtained. The exhaust jet will be simulated by compressed air up to

2,000 psi and either by combustion of JP-lox at $P_c = 1{,}000$ psi and of H_2-O_2 or by hydrogen peroxide decomposition at $P_c = 1{,}200$ psi$\longrightarrow$3,000 psi. Variables in the test program will be plug length and contour (i.e., effect of truncation) and secondary bleed or venting for the annular configurations. The ejector nozzle, of course, is aimed at air augmentation in the manner of the ram rocket. The variables in this phase of the program will include mixing length and primary-to-secondary flow ratio. The adaptability and versatility of the above facilities to these research areas covering the full booster trajectory are unique.

In summary, then, for very large boosters, the advanced concept that presently appears to be of most promise is one wherein a multitude of individual combustors are clustered so as to feed into one common nozzle which is vented to near ambient pressure to achieve altitude compensation. The provision of a near ambient pressure environment at the point where the external expansion begins is a necessary ingredient in altitude-compensation nozzles. Venting of annular nozzles has been pioneered and successfully demonstrated in studies conducted at the Lewis Research Center; however, much work yet remains in the area of vehicle, nozzle, and engine aerodynamics. Much wind-tunnel work must be done on specific configurations!

In a multistage vehicle, the use of an advanced nozzle can lead to payload increases in the order of 4 to 5 percent. In a single-stage-to-orbit application, the payload gain, due to altitude compensation in the nozzle, may vary from 17 to 68 percent, depending on the particular chamber pressure. If single stage to orbit is a concept of merit, much is to be gained through the use of the altitude-compensating nozzle and to the use of the very high chamber pressures (approaching 3,000 psia).

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TABLE I.- PERFORMANCE POTENTIALS

Booster	Variable	Payload gain, percent
Three stages to escape at 150 miles	P_c from 1200 to 3000 with comparable ϵ	6 to 8
	Advanced nozzle in first stage	4 to 5
	Large ϵ in upper stages	13 to 25
Single stage to orbit at 150 miles	Advanced nozzle at $P_c = 1200$	68
	Advanced nozzle at $P_c = 3000$	17

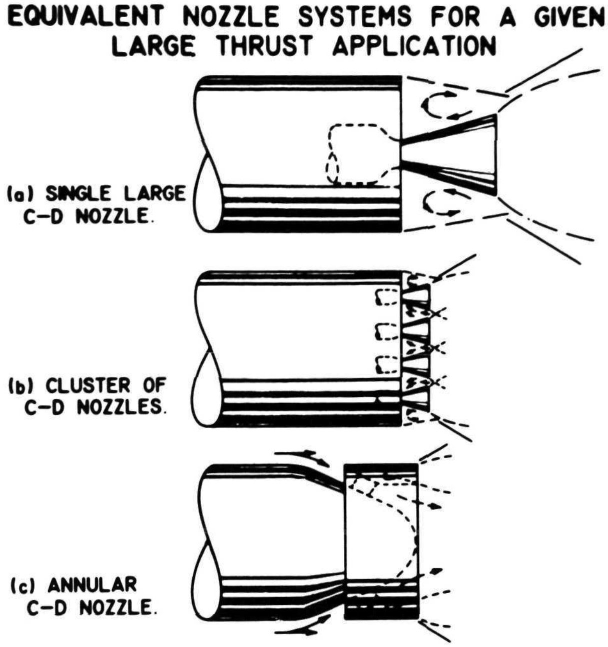

Figure 1

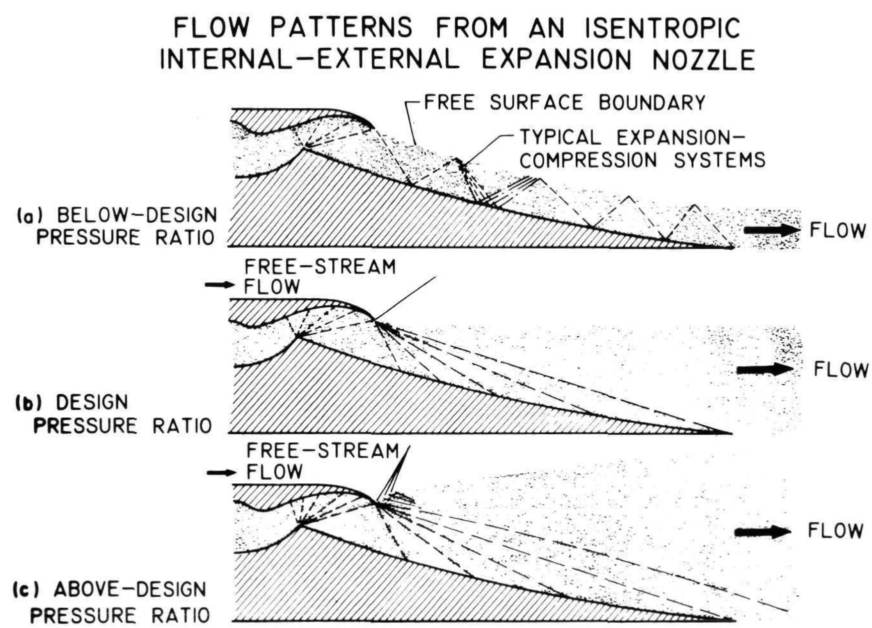

Figure 2

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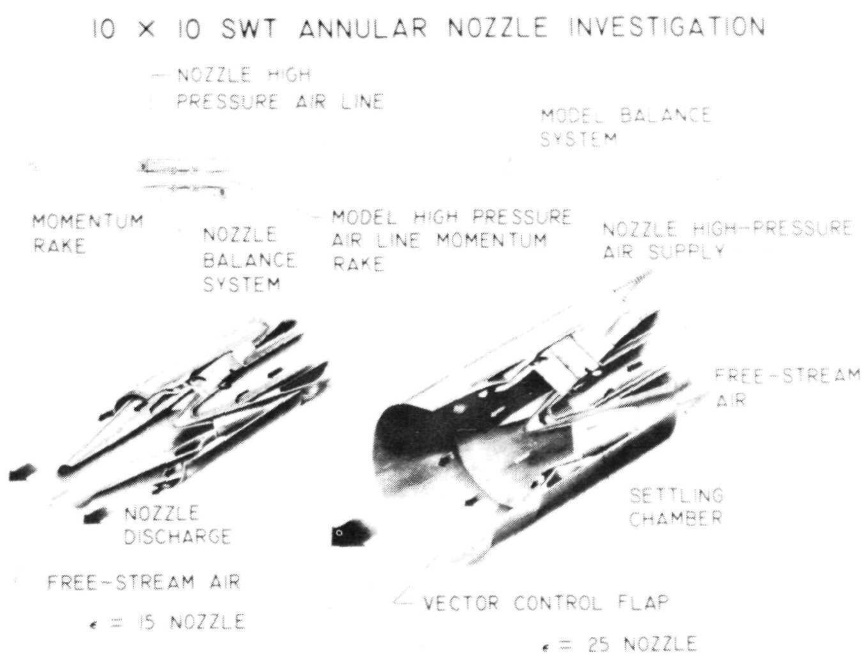

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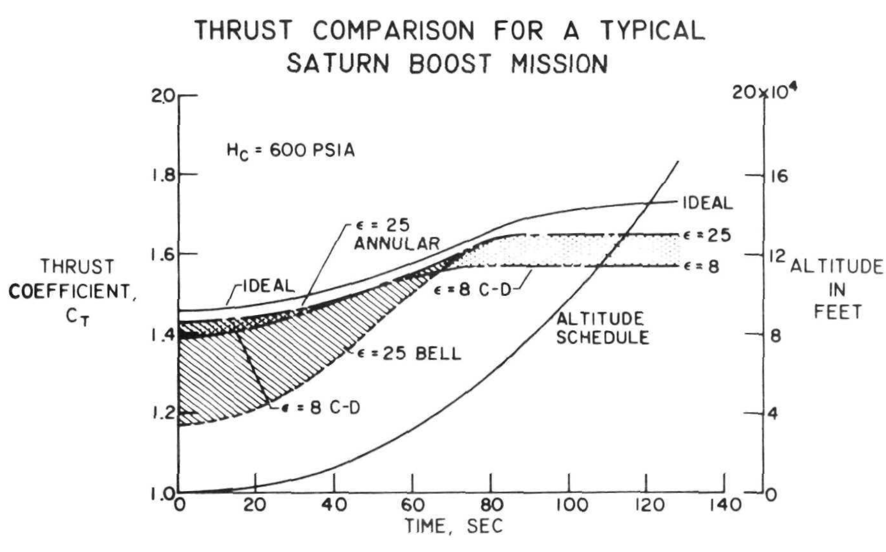

Figure 4

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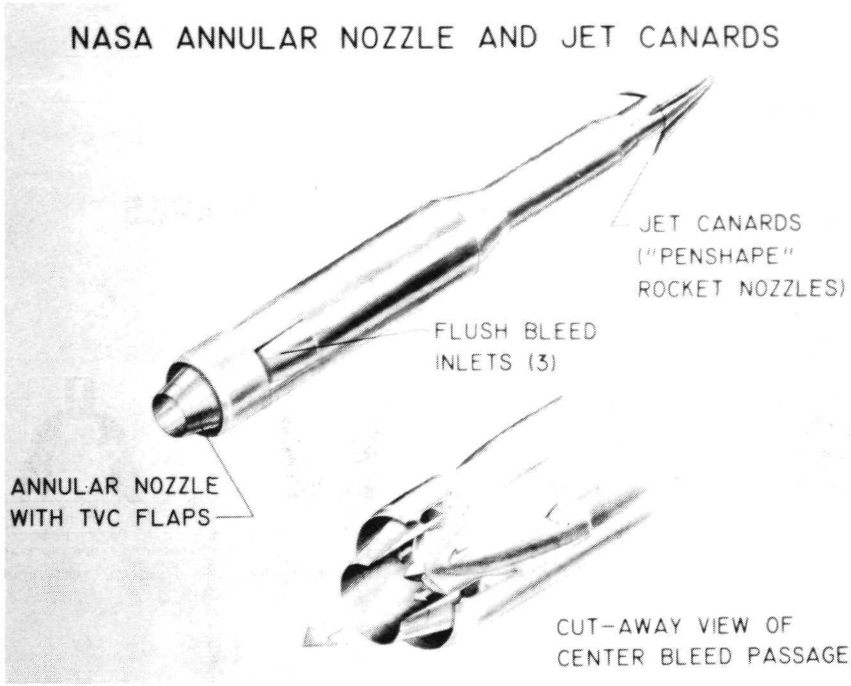

Figure 5

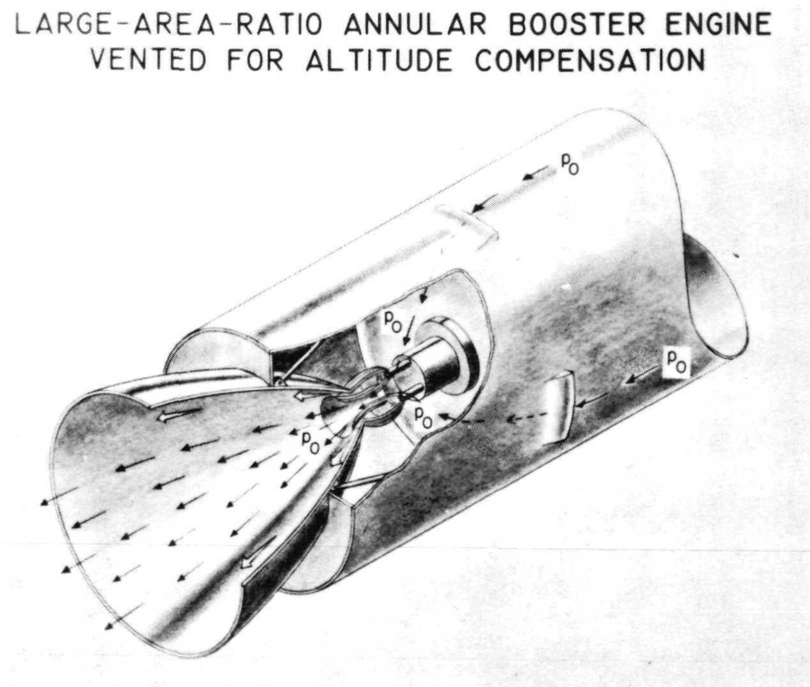

Figure 6

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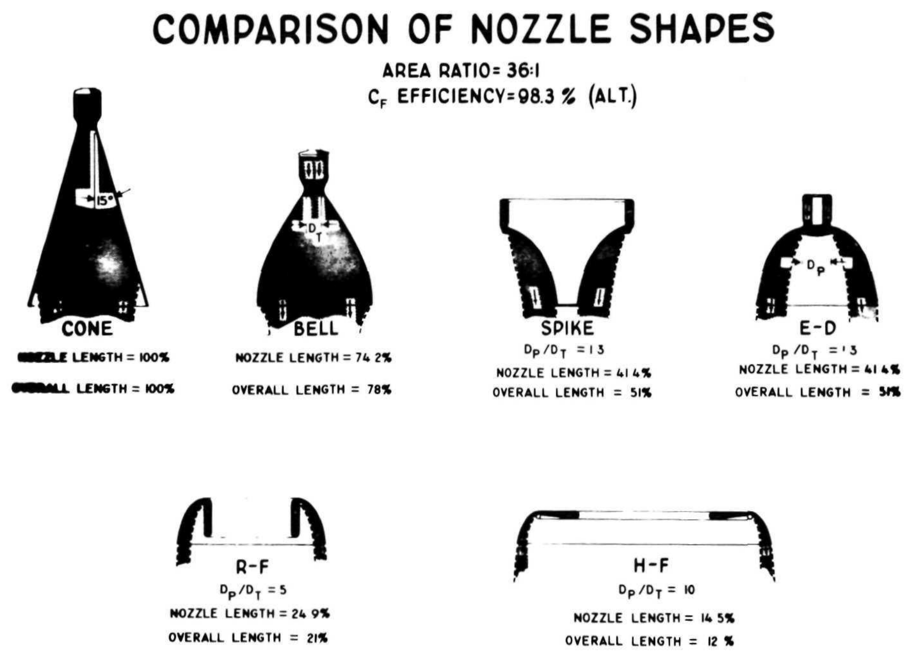

Figure 7

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27. AIR FORCE ALTITUDE-COMPENSATED NOZZLE

By Clark W. Hawk

Edward Flight Test Center, USAF

This paper is a discussion of the Air Force Rocket Propulsion Laboratory work on the advanced nozzles. This discussion will cover some of the background on why the Air Force became interested in advanced nozzles, where its interest lies, the results of some of the programs conducted over the past three years, the present status of advanced nozzle work, and what should be done next.

The Air Force nozzle programs have included development of

expansion-deflection nozzle
reverse-flow nozzle
discrete throat forced-deflection nozzle

and the approaches used have been by

- cold flow
  - performance correlation
  - TVC
- hot firings
  - performance correlation
  - heat transfer

Free expansion or altitude compensating nozzles can be considered as being of two classes, one in which the flow is directed radially inward (or to the center line) as in the spike configuration, and the other concept in which the flow is directed radially outward, exemplified by the ED and RF configurations.

The expansion process is dependent upon the pressure realized on the base of these plugs or at the base of the lip on the radial inward flow configuration.

If the base pressure is lower than ambient pressure, there are two losses: (1) a drag loss on this base because the pressure is lower than ambient, and (2) overexpansion of the rocket jet gas to this base pressure, so that the altitude compensation feature is then degraded somewhat.

The term, geometric area is self-explanatory; it is the total exit area divided by the throat area. For years it has been the practice to compare all performance and weight on the basis of the aerodynamic ratio, which is represented by the ratio of total exit area minus base area divided by the throat area.

The nozzle area ratio essentially defines the design point of the nozzle as that condition where the flow is discharged axially from the nozzle.

When the Air Force first started its advanced nozzle program, the goal was a single-stage 8500-nautical-mile ICBM, using the storable propellant combination N_2O_4 and a 50:50 blend of UDMH and hydrazine. The sea-level thrust was two million pounds, stage mass fraction was 0.96 (scaling up of Titan II), and the chamber pressure was 2770 psi.

For these conditions if a 100 percent altitude compensation can be obtained, a payload of 43,000 pounds can be put on target, whereas with a bell or conventional nozzle technology only 31,600 pounds can be put on target. This is based on the assumption of a bell area ratio of 50:1 and advanced nozzle area ratio of approximately 100:1. This is a 36 percent payload gain which is rather significant.

In deciding on which of the nozzle families efforts should be concentrated, some analytical work was done by Douglas Aircraft as well as some in-house work. Here, briefly, in table I are some of the general trends indicated by these studies which led to concentration on the radially outward flow family of nozzles.

TABLE I.- ANALYTICAL COMPARISONS OF NOZZLES

Nozzle	Thrust chamber weight, lb	Performance	Heat flux
Bell	1600	Reference	Lowest
RIF	4000	Best	High
ROF	1792	Better	High

First, some thrust chamber weights for a million-pound oxygen/hydrogen system operating at 1500 psi chamber pressure were generated, with an aerodynamic area ratio of 25:1. With the bell nozzle as the reference weight of 1600 pounds, the radially inward flow or spike nozzle was two and one-half times as heavy; and the radially outward flow family of nozzles had roughly 12 percent greater weight.

From a performance standpoint, the bell nozzle was again used as a reference. The radially outward flow nozzles offered some advantage in performance, but the spiked nozzle was the best. However, it did not appear that at that time the advantage was sufficient to overcome the weight disadvantage.

Finally, from a heat-transfer standpoint, the radially inward flow and radially outward flow families both exhibited higher heat-transfer flux in the throat and to approximately the same extent.

Since 1960 the Air Force has conducted three programs (fig. 1): the expansion-deflection nozzle and the reverse-flow nozzle concepts at Rocketdyne in 1960 and 1961 (some of this work has been carried on until this year) and, more recently, the discrete throat forced-deflection nozzle concept of Aerojet in conjunction with the so-called integrated components program which encompasses many other things besides the advanced nozzle work. In all cases, the approach has been generally the same in that the Air Force has conducted the initial analysis work, the cold-flow experimental programs to evaluate the performance of these nozzles, the evaluations of thrust vector control techniques with particular emphasis on secondary injection, and, finally, the use of hot firing programs at a reasonable scale to (1) get performance correlation with cold flow data, and (2) to get heat-transfer information with these advanced nozzles.

Since the Aerojet forced-deflection work is the more recent program and more or less encompasses all advancements to date, the majority of this discussion will be directed toward the Aerojet program with a comparison summarizing the advanced nozzle work.

The Aerojet concept has the eight individual circular combustors and throat with a supersonic transition section employing contoured sides and having somewhat of a hot dog shape so that an annular jet is produced. There is provision to bleed in air to the center body and attempt to bring up base pressure between the individual combustors.

The four areas considered major problems that had to be defined were:

(1) Area ratio definition. The aerodynamic area ratio was not considered a satisfactory basis for comparison, so tests were conducted at pressure ratios beyond the design pressure ratio based on aerodynamic area ratio to determine whether maximum nozzle efficiency corresponded to aerodynamic or geometric area ratio.

(2) The engine segmentation approach to construction of large engines. It seemed important to determine the effect of engine segmentation on engine performance, whether performance loss resulted from the privileges of building in segments.

(3) The effect of external flow on performance.

(4) The base bleed or attempts to raise base pressure in the center to get altitude compensation.

In figure 3, C_T (the ratio of actual ideal thrust coefficient) is plotted against pressure ratio. The advanced nozzle concept (the one having eight discrete throats, an overall area ratio of 73, and an aerodynamic area ratio of 59) is compared to bell nozzle performance with a geometric area ratio of 59.

The performance reached its peak at a pressure ratio of 1840 for the bell nozzle, whereas with the advanced nozzle the peak was at about 2550, corresponding to the design point for the geometric area ratio of 73.

Figure 4 shows the effect of segmentation; the C_T ratio is plotted against pressure ratio for an annular throat model (No. 9), and for the model with eight circular throats. Sea-level operation would occur at a pressure ratio of about 190; there is something less than 1/2 percent loss in performance associated with segmentation at sea level, and roughly 1/2 percent at the design point.

All the work, including that shown in figure 5, has been conducted for Aerojet by Fluidyne in Minneapolis. Attempts to get some external flow information in their transonic tunnel were quite limited and resulted in rather meager data at Mach numbers of 0.95 and 1.35 over a range of pressure ratios. However, it established that the still air performance curve, shown in figure 5, is rapidly approached; this is verified by a program conducted by Rocketdyne for the Navy, in which they did considerable testing at Mach numbers 2 to 4, with very small nozzles, using a spike, the reverse flow family, and the bell nozzles. But the trends are exactly the same in that still air performance intersection occurs at very low pressure ratios.

It should be emphasized that because of the high takeoff pressure ratio (190) the results from the Air Force program showed that performance penalties are essentially nil at sea level and at higher pressure ratios.

Figure 6 is a chart of altitude plotted against Mach number, again for an 8500-nautical-mile single-stage ICBM. This figure shows that Mach 1 is reached at an altitude of 23,000 feet, or pressure ratio of approximately 450. Hence, fairly high pressure ratios are reached before the external flow will offer a significant effect, and from the trends exhibited on the previous curve it appears that these effects will be small.

Figure 7 shows attempts to improve altitude compensation through base ventilation. The technique of bringing air onboard was poor at best. There was considerable total pressure loss in the system for bleeding air into the center body. However, we did realize, in comparing the effects of bleed on the performance of the annular throat model that we had $1\frac{1}{2}$ percent to 2 percent performance gain at the sea-level pressure ratio of 190, and this gain tails off until performance is essentially unchanged at a pressure ratio of 500.

Figures 8 and 9 show where this performance gain comes from. In comparing the secondary bleed flow with the primary flow at various pressure ratios in figure 8, it appears that at sea-level conditions approximately $3\frac{1}{2}$ to 4 percent bleed air is being brought in.

Figure 9 shows the effect of this on base pressure; the base-pressure-to-ambient-pressure ratio is plotted against the chamber-pressure-to-ambient-pressure ratio. Ideally, the base pressure should be equal to ambient pressure to get altitude compensation. However, with the radially outward flow family nozzles, this curve is typical of all models tested in that base pressures are on the order of eight-tenths of ambient at low pressure ratios, and they drop off with increasing altitude or pressure ratio to minimum levels. In this particular case, the base pressure was about 0.5. However, in the Rocketdyne programs with ED and RF, they dropped down in some cases to as low as 0.1.

There is a negligible effect on base pressure resulting from the bleed. Hence, it can be stated that essentially some performance improvement is obtained through mass addition but not through increased base pressure.

To compare all the nozzle work that the Air Force has done is a rather formidable task because the work has been done with different area ratios and different gammas. Figure 10 is an attempt to find a way to put all these data on a comparable plot.

The C_T was plotted against pressure ratio for all advanced Air Force nozzles, this being a typical curve. The performance of a nonseparating bell nozzle of the same geometric area ratio, and with the same gamma as the advanced nozzle, has been used as the basis for comparison. For each nozzle test, there is a C_T versus pressure ratio curve, and there is also a nonseparating bell curve for the same gamma and same area ratio. Therefore, at a given pressure ratio the C_T ratio of the advanced nozzle (its maximum C_T achieved during the test divided by its design C_T) can be compared to the same C_T ratio of the nonseparating nozzle for the given conditions.

This is illustrated in figure 11. The no-compensation condition (the nonseparating bell) is a 45° line on this figure.

The performance of the 80 percent bell that was tested in the Aerojet program has been plotted to show that with separation there is some degree of altitude compensation. There are better ways to design separating bell nozzles; hence, the compensation illustrated is a little lower than might be expected if good design practice were used.

An interesting point is that the data plotted in figure 11 are for the Aerojet forced-deflection work with annular throats and the Rocketdyne ED configuration with discrete throats, and some data from a company-sponsored program performed by Pratt & Whitney, yet all these data from different tests rigs and with different area ratios, fell with a percent of one another.

The forced-deflection nozzle with bleed again shows an increase in altitude compensation.

The X data points in figure 11 which represent hot-firing data with the ED nozzle at Rocketdyne show very good agreement with the cold-flow program. The forced-deflection hot-flow results from Aerojet show slightly higher performance but again the same trends as exhibited by the cold flow data, and this difference could very well be just scatter of instrumentation.

It is unusual that the results from the RF nozzle cold-flow data do not exhibit the same characteristics as the other radially outward flow family of nozzles. These data show lower compensation at the low-pressure ratios near sea-level condition, and the curve crosses over all other lines as it nears the design point. There is no obvious explanation for this other than that there may have been some inaccuracies in the measuring system or in the data from Rocketdyne.

Data from the hot flow test program that was conducted at NASA-Lewis in their PSL-2 facility are plotted in figure 11. These results exhibited the same characteristic shape as our other plots from cold flow tests. The performance did range about 1.5 to 2 percent higher than indicated from our cold flow data, but yet it was in the same magnitude.

Also plotted in this figure are data from a radially inward flow or spiked nozzle tested on a company-sponsored program by Aerojet. These data are included for strictly comparative purposes because there was very little data available as to the manner in which the tests were conducted. The nozzle was tested in cold air with an area ratio of about 18 to 1 and shows a gain of about 5 percent over the other family of forced-deflection nozzles.

As stated previously, our mission analysis involved a single-stage 8500-nautical-mile range with sea-level thrust, mass ratio, and chamber pressure assumed. The payload of the conventional bell nozzle is 31,000 pounds, and that of a 100 percent compensated nozzle, 43,000 pounds.

With the results of our cold flow program from Aerojet, not considering any bleed whatsoever, we calculated a payload for this mission of 39,200 pounds, representing a 24 percent increase over conventional technology. This was somewhat surprising to me in that I did not think our altitude compensation work had been so successful, but the results here are rather significant, I believe. This is not a contractor-conducted study. This was done by our own Air Force mission analysis people. All the input was from my reduction of these data as I have just shown you, and with our own trajectory analysis, et cetera. Hence, we believe strongly that this 39,200 pounds is a realistic number.

Table II is a summary of the present status of the program, as far as altitude compensation is concerned. With our radially outward flow family of nozzles, we have demonstrated altitude compensation in excess of 54 percent, with, consequently, a significant payload increase.

TABLE II.- SUMMARY

Problem	Results	Effects
Altitude compensation	>54-percent compensation	Significant pay load gain
Definition	Geometric	Weight
Discrete throats	<1-percent loss	No segmentation penalty
External flow	No loss at high pressure ratios	Design for still air
Base bleed	Mass addition gain	Undefined

The single-stage application really points up the difference or the added performance of these advanced nozzles because each slight increase is magnified in performance. In multiple stages, two- and three-stage vehicles, this 54 percent compensation looks more like 5 percent, rather than 24 percent.

As far as area ratio definition is concerned, we are now convinced that geometric area ratio is the correct method of definition. This will have a significant effect on weight analysis in all weight comparisons that have been made and will be made. I think that this will bring us down closer to the weight of bell nozzle so it will be more competitive from a weight standpoint.

The effect of discrete throats is such that we realize something less than 1 percent loss in thrust. I should say that there is little penalty associated with engine segmentation rather than no penalty.

From the results and trends that we have observed, there does not seem to be any performance loss due to external flow at high-pressure ratios, and with our high chamber pressure interest we believe that we are beyond the possible effects of external flow, that we can design essentially for still air performance obtained from cold flow tests.

As far as base bleed is concerned, it appears that mass addition has been the sole manner of improvement. The effect needs a good deal more study. If this air can be brought onboard without penalty, this is an improvement.

As far as recommendations are concerned, I think that the advanced nozzles have shown in our cold-flow and hot-flow programs, a considerable degree of promise as far as the aerodynamics are concerned, such that they should be given serious consideration for future engine designs. I think that we are to the point now where it is necessary for us to solve some of the other problems associated with the advanced nozzles, such as heat-transfer and fabrication problems.

I do not think that we can ignore the aerodynamics completely in that from my mission analysis there still is about 12 percent payload possible, and I believe it is worth further effort. This should be attacked in my estimation through base bleed.

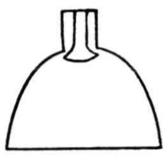
The problem is how to bring it onboard in such a manner that base pressure is increased, rather than just have mass added to the flow. And second, tests should be made with external flow to see whether this flow is brought onboard and what losses are incurred.

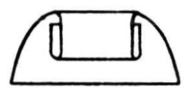
Finally, an effect which has not been discussed, but which is worthy of mention, is the effect of base area ratio. If we increase the size of the base in relation to the throat area, I think that we can reduce the jet pumping effects which tend to reduce our base pressure, or possibly delay the jet pumping effects to higher pressure ratios. This may be a worthwhile area of investigation.

For the coming year the Air Force has contracted with Aerojet to continue integrated components work and in the advanced nozzle work effects of base bleed and varying base area ratio will be determined. This is due for completion in approximately June of 1964.

Free expansion nozzles

Radially inward flow

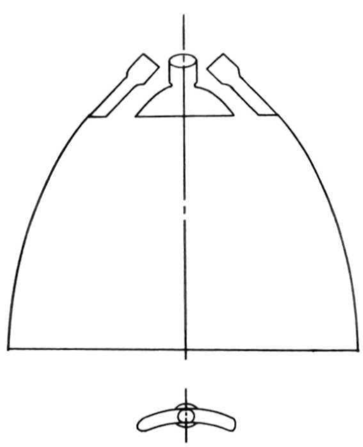




Radially outward flow

Figure 1

DISCRETE THROAT R.O.F. NOZZLE



ϵ DEFINITION

DISCRETE THROAT EFFECTS

EXTERNAL FLOW EFFECTS

BASE BLEED EFFECTS

Figure 2

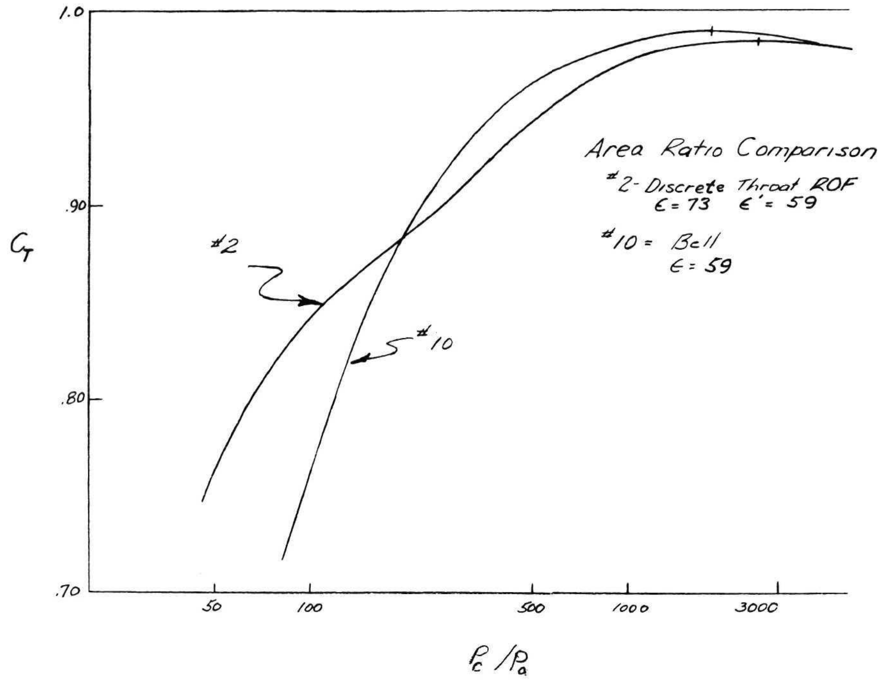


Figure 3

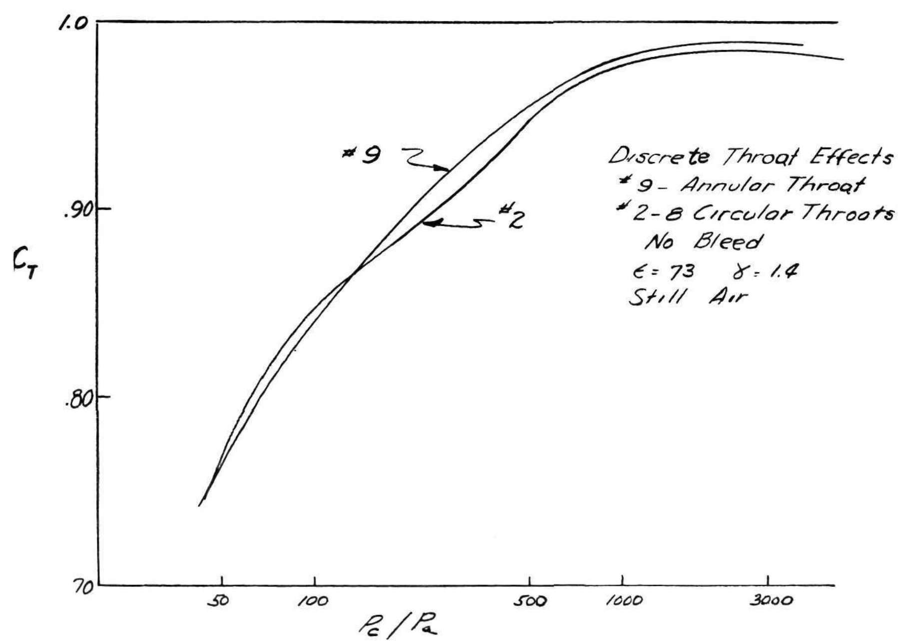
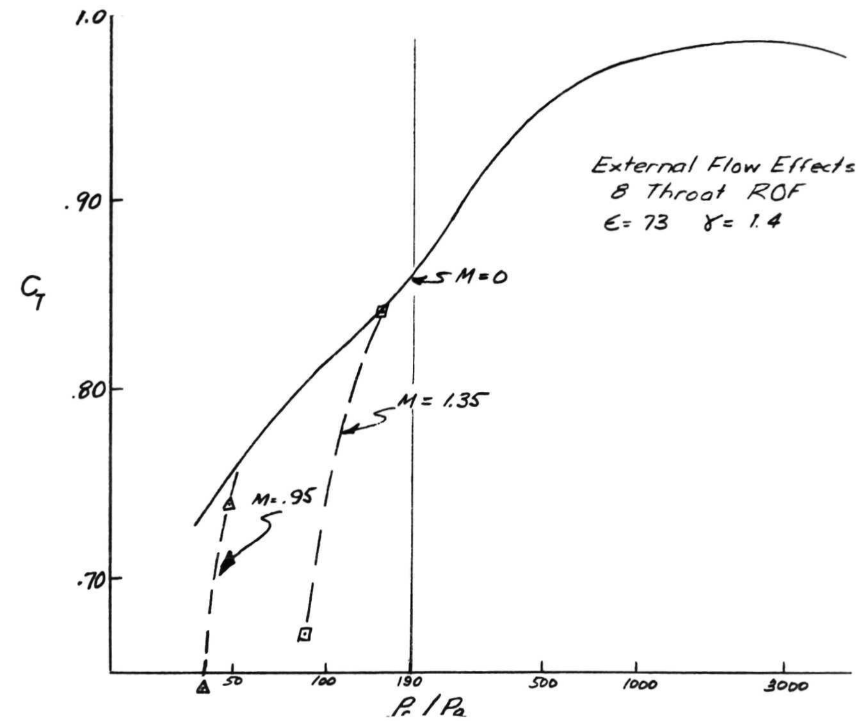


Figure 4



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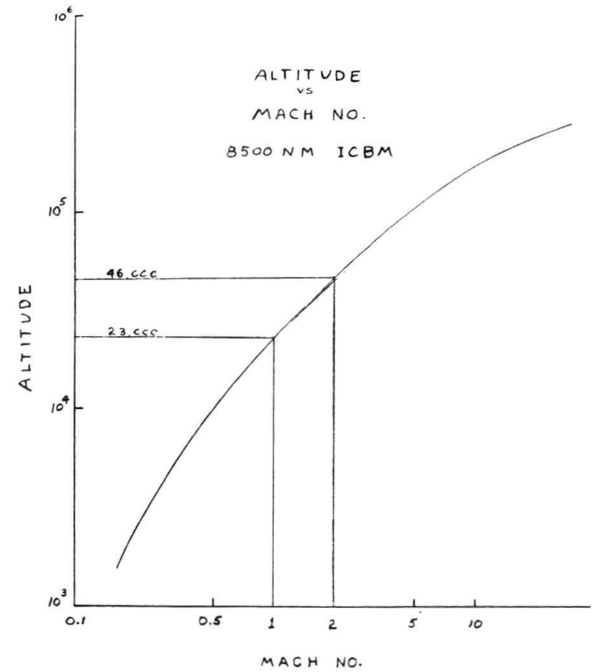

Figure 6

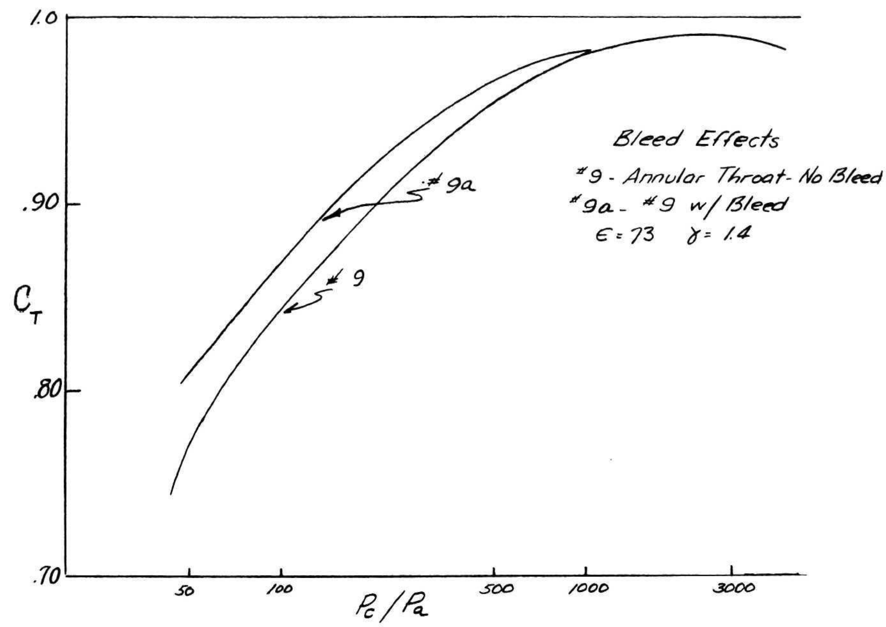

Figure 7

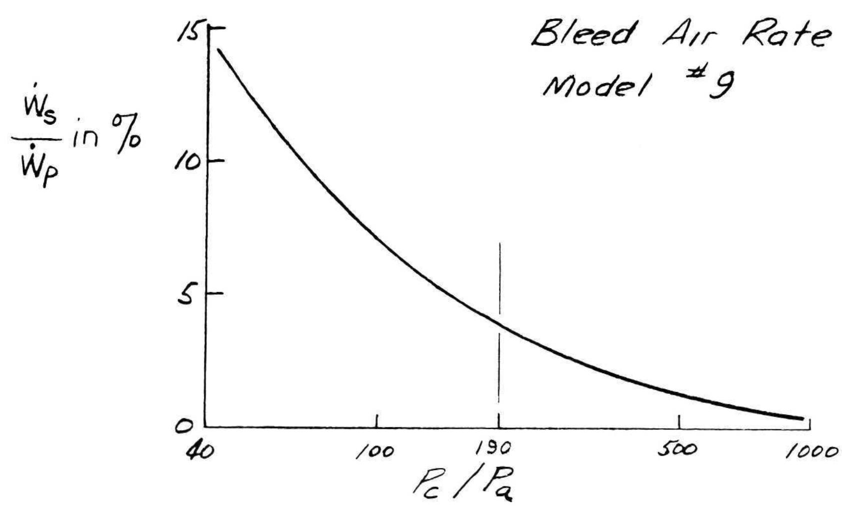

Figure 8

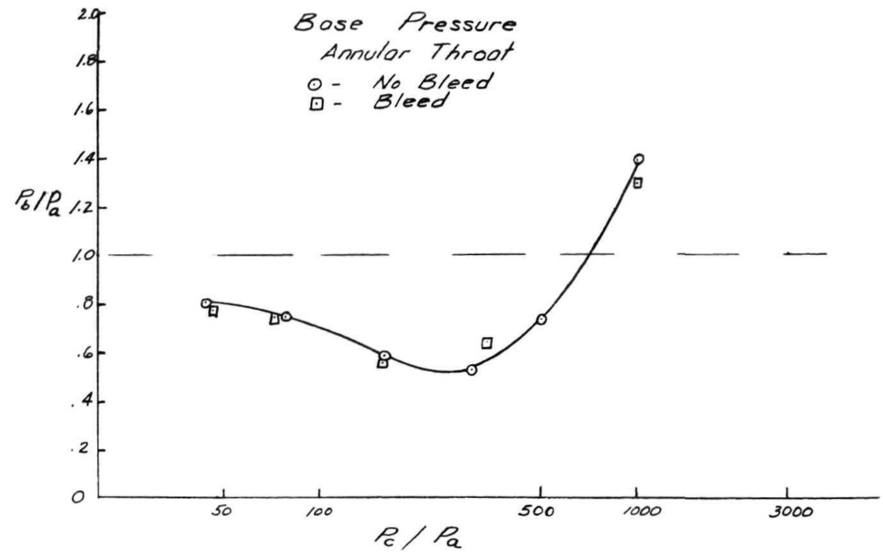

Figure 9

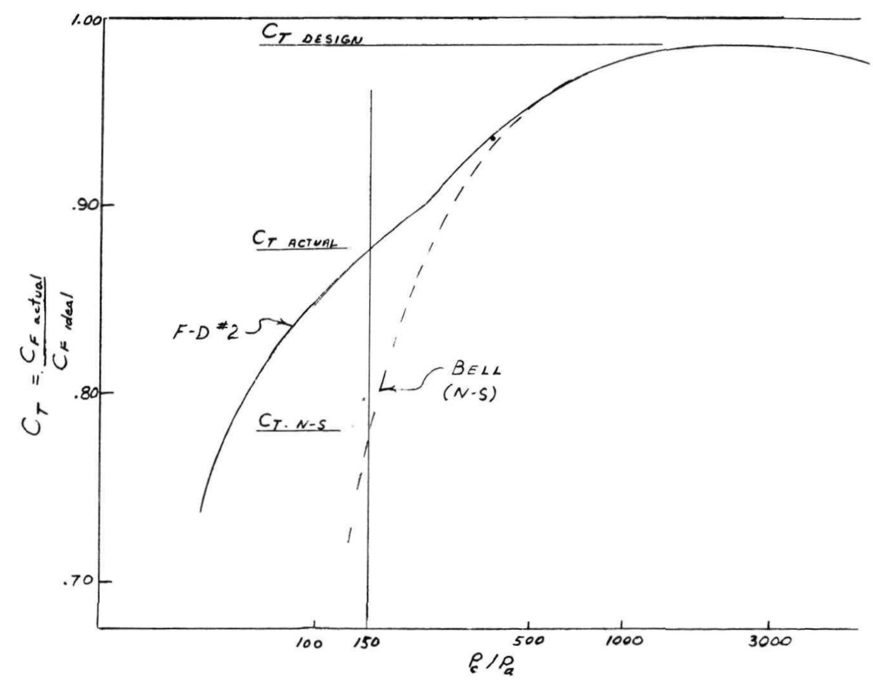

Figure 10

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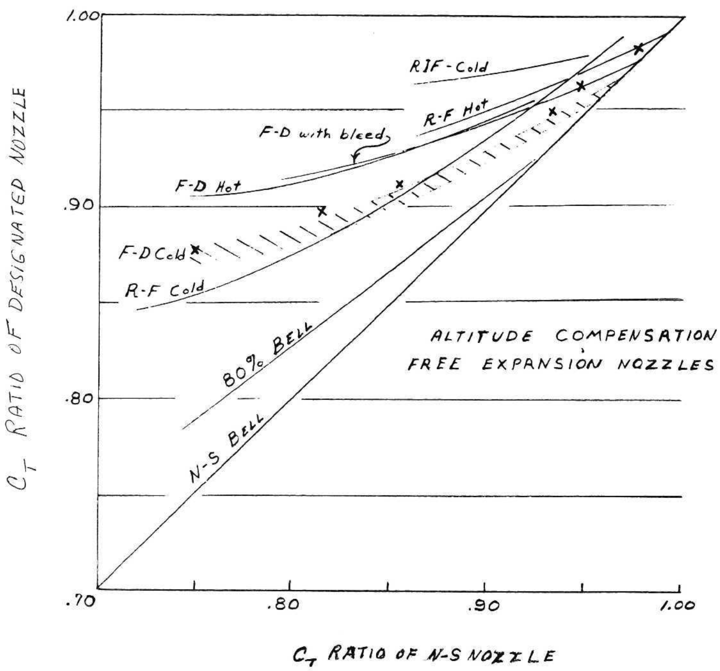

Figure 11

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28. EJECTORS AND THRUST AUGMENTATION

By Milton A. Beheim

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This discussion is a general comment on the controversial subject of air augmentation for rocket motors.

The payload fraction of a launch vehicle, of course, is sensitive to the propulsion system specific impulse, and means of increasing the specific impulse are of a great deal of interest.

It is well known that the specific impulse of air breathers substantially exceeds that of the highest performance chemical rocket engines, and the well-known specific impulse $\left(I_{sp}\right)$ curves are shown in figure 1.

Of course, the rocket performance is low compared to the air breathers. It is important to point out that this is on design performance for the air breathers. Air breathers are basically low-pressure devices and, therefore, to obtain high thrust big inlets and big exits are needed. If operation of these air breathers is attempted over a wide Mach number range, such as that of a launch vehicle, there are very serious off-design drag problems of the inlets and exits.

Although the high performance potential of air breathers is very difficult to apply to launch vehicle propulsion, there is the possibility that certain aspects of air-breathing propulsion may be used to augment the performance of the relatively simple and lightweight chemical rockets, thereby achieving only a modest gain in performance but perhaps useful gains in payload.

In recent months the term "air-augmented rocket" has been used to describe such a composite system, and its essential features are shown in figure 2. The primary rocket jet is used as the primary fluid in an ejector, and the secondary flow is provided by an air inlet. Mixing and combustion occur in the ejector and possibly additional fuel may be added to the air in the inlet.

With proper matching of the inlet and ejector flow characteristics, the composite system can be expected to provide some degree of altitude compensation for the rocket jet. In addition, combustion in the secondary air at supersonic flight speeds should provide some ramjet thrust, and at subsonic flight speeds the ejector pumping action could provide lip suction forces if a suitable blunt lip can be provided at take-off.

At all flight speeds mixing of the rocket jet with the secondary flow decreases the jet exit velocity and, hence, tends to increase the propulsive efficiency. In effect, it is hoped that mixing can be used to transfer total momentum from the primary to the secondary flow, thereby increasing the total mass of gas being accelerated by the propulsion system.

[redacted]

A difficult problem that must be considered when attempting to achieve these performance gains is the total pressure losses incurred in mixing, heat transfer, and combustion process.

In concept this system is similar to the ducted rocket or the ramrocket that has been studied over the past several years, with one essential difference - it emphasizes lower secondary flows, thereby achieving less of a performance gain but also incurring less of the weight penalties of large ducting and also less of the off-design drag problem.

At present, only limited experimental data on the performance of this composite system at these moderate secondary flows are available, and estimates of propulsion system performance are difficult because several modes of operation are required over the Mach number range: as a jet pump on take-off, as a composite ramrocket at moderate flight speeds, and as a pure rocket at high altitudes. The system will be operating off-design much of the time, since the weight penalties of variable geometry probably cannot be tolerated.

An evaluation of the usefulness of the air-augmented rocket for launch vehicle propulsion is fairly complex because it requires consideration of the complete vehicle design for a particular mission. In addition to the difficulties of predicting propulsion system performance, the concept will require much additional structural weight because of the ducting and also because lower flight trajectories and, hence, higher dynamic pressure will undoubtedly prove optimum in order to increase the benefits of air augmentation.

An adequate mission study employing this propulsion system has not yet been made.

Preliminary in-house studies by industry appear to be conflicting, or, at least, they appeared to be conflicting until yesterday.

A Martin-Denver study, publicized some months ago, indicated large increases in payload for some missions (for example, the payload doubled at high altitudes). Yesterday it sounded as though they had changed their minds.

Rocketdyne has also made in-house studies of such missions in the past, and they concluded that the payload gains were not interesting. Yesterday it sounded as though they thought maybe they had missed something, and maybe they had better go back to look at it again.

To clarify the situation, NASA has initiated a contract with the Boeing Company to undertake an analytical study of air-augmented rockets and of launch vehicle missions employing this propulsion system.

This six-month effort is intended to make a realistic evaluation of the potential advantages of the propulsion system so that a decision can be made on whether additional effort is justified as a launch vehicle propulsion system. In addition, the study is intended to define the technical areas requiring additional effort in future experimental work if the concept is promising.

Turning now to some of the technical problems associated with the concept, one of the principal problems mentioned earlier is that of matching the inlet and ejector flows over a wide range of flight conditions. Conventional performance of an inlet at supersonic speeds is shown in figure 3. The supersonic inlet is shown on the top and typical pressure recovery-mass flow curves on the bottom. The effect of flight Mach number is indicated.

Critical operation corresponds to inlet operation at the highest pressure recovery and the highest mass flow simultaneously, and in the inlet this means that the normal shock is positioned at the minimum area of the diffuser.

During subcritical operation, the inlet discharge is restricted either by mechanical means, or in air-augmented rockets it would be restricted by the ejector pumping characteristics. This moves the normal shock out in front of the inlet and spills air; high drag results and inlet flow instability can occur.

By decreasing the discharge restriction of the inlet, the normal shock goes back into the inlet, and there are large total pressure losses in this supercritical operating mode. The integrated static pressures inside of the duct become diminished, and, therefore, the pressure forces acting in the thrust direction are less.

Extensive inlet studies have already been made on a variety of configurations. However, installation of inlets on launch vehicles might lead to some unique problems that have not been studied sufficiently yet. Some configurations which have been considered are illustrated in figure 4.

The inlets may be wrapped around the rear of the launch vehicle, stuck out on pods, or the launch vehicle can be built as an inlet. There are obvious problems with each of these possibilities. Certainly the solution suggested in the upper sketch of figure 4 the boundary layer at the rear of a launch vehicle (which will be a couple of feet deep) will create problems of making an inlet operate at satisfactorily high mass flows and pressure recoveries.

At low speeds the inlet functions differently, as illustrated in figure 5. Take-off performance might be quite important because a lot of structural weight is being added to the launch vehicle for air augmentation.

At takeoff, air is accelerated into the inlet by the pumping action of the ejector. If a blunt lip is used, as illustrated on the left, the streamlines of the accelerated flow are curved as they enter the inlet. This produces a static pressure gradient on the face of the lip, thereby producing a lip suction force which acts to augment the propulsive forces.

The blunt lips are unsatisfactory, however, at high supersonic speeds because of high drag and low internal performance. Sharp lips, as illustrated on the right of figure 5, are required at high speeds. However, then there is the problem that the lip suction force at take-off no longer can exist, and there are internal pressure losses, resulting from separation of the flow as it is accelerated around the sharp edge.

Therefore, there is a compromise necessary in high- and low-speed performance, or some form of variable geometry such as an inflatable lip is required.

In the air-augmented rocket, the inlet discharge conditions are determined by the ejector pumping characteristics. The basic types of ejectors that have been studied extensively in the past are illustrated in figure 6.

The industrial ejectors are characterized by long mixing sections and by their ability to pump fairly large quantities of secondary flow with modest quantities of primary flow. The pumping action of the industrial ejector is derived from the turbulent exchange of momentum from the high velocity primary flow to the low velocity secondary flow; hence, mixing is of much importance in determining the performance of this ejector.

Aircraft ejectors, on the other hand, are characterized by short lengths because of the requirement for light weight. Three different types of ejectors have been studied: convergent, cylindrical, and divergent.

The aircraft ejectors derive their pumping ability from a static pressure gradient imposed upon the secondary flow by the high-pressure-ratio primary jet. As the underexpanded primary jet plumes out, it restricts the flow area available to the secondary stream. This imposes a static pressure gradient which accelerates the secondary flow. The ejector is short to minimize weight, and, within these limits on length, mixing is slight and has minor effects on performance.

Typical ejector thrust and pumping curves are illustrated on the lower left of figure 6. Here the ejector to ideal primary thrust ratio and the secondary to primary total pressure ratio are shown as functions of the primary to ambient pressure ratio. These are shown for different values of the corrected secondary flow ratio, defined as the secondary to primary weight flow ratio multiplied by the square root of the temperature ratio divided by the molecular weight ratio.

These pumping and thrust curves can be substantially influenced by the heat transfer from the high temperature primary jet into the secondary flow. They can also be influenced by combustion of the secondary air.

For application to air-augmented rockets, lightweight ejectors such as the aircraft ejector type undoubtedly are required. If convergent or cylindrical ejectors are used, probably divergent nozzles will be required downstream of them to provide sufficient expansion ratio. To decrease overall length the divergent ejector is of particular interest.

The pumping characteristics of the three types of ejectors are substantially different and depend on details of the geometry of each ejector. A final choice of ejector design must be based on a careful evaluation of the ejector-inlet matching the problem over the wide range of flight conditions to be encountered.

This matching problem is especially difficult because of the probable need for using fixed geometry inlets and ejectors. However, regulation of the secondary fuel injection may prove useful as a means of controlling matching.

Although mixing of primary and secondary gas generally is slight in short ejectors, fairly extensive mixing is probably desirable in the air-augmented rocket. The rocket motor is operated fuel rich to maximize rocket specific impulse. Hence, by mixing primary and secondary streams, combustion of excess fuel in the rocket jet with secondary air can occur and thereby increase overall specific impulse. It may also prove advisable to augment this ramjet mode of operation by injection of additional fuel into the secondary air.

An additional effect of mixing primary and secondary flow is to decrease the magnitude of the primary jet exit velocity thereby increasing propulsive efficiency and decreasing wasted jet kinetic energy. However, thermal efficiency decreases simultaneously, and the overall efficiency, which is the product of thermal and propulsive efficiency, must be considered in selecting the optimum jet exit velocity.

A problem exists in that means of achieving rapid mixing in short length ducts without incurring excessive total pressure losses are not known, and analytical tools for predicting mixing rates are meager. The phenomenon is complex in that velocity, temperature, and concentration spread at different rates, and experimental measurements of mixing are difficult.

An obvious means of increasing the mixing achieved in a given length is to increase the surface area of the mixing zone by employing multi- or annular-primary nozzles. It is anticipated that combustion processes in the presence of shock waves between mutually interacting supersonic streams in the ejector will also promote rapid mixing.

It is likely, however, that the optimum mission performance will be achieved with less than complete mixing.

However, it is important to recognize that total pressure losses result from mixing, heat transfer, and combustion proccesses such as those in the ejector. Since large thrust penalties can result, these losses must be considered. However, mixing has not been found to be an efficient means of transferring energy between two streams. The industrial ejector is a case in point.

It is apparent that experimental data will be needed on air-augmented rocket performance if analysis shows sufficient promise in the concept for launch vehicles. As indicated earlier, existing data are meager and subject to some uncertainty in interpretation. In particular, augmentation ratios that have been measured in past tests with small hot rocket motors are influenced by the combined effects of the many factors affecting performance without providing detailed information on each of these factors. Hence, the results may not be generally applicable to other propulsion systems which may be similar but differ in detail, such as scale and rocket motor combustion efficiency.

To extend the experimental data pertaining to air augmentation the Martin Company has been contracted recently by the Air Force to perform additional tests with JP-lox rocket motors in a test stand at AEDC. Air augmentation of a cluster of 12 motors will be determined.

It is anticipated that additional tests, proposed for Fiscal Year 1964, will be conducted in the Lewis and Langley wind tunnels if preliminary study results are promising.

In conclusion, air augmentation of chemical rocket motors tentatively appears to be an interesting means of increasing the payload capability of launch vehicles. A thorough evaluation of such a composite propulsion system is quite complex, but the initial steps have been taken. If preliminary analysis proves merit in the concept, extensive experimental work will be required because of the present lack of experimental data, and because the complexity of the flow processes means they are not easily analyzed.

It is doubtful that simple mixing will in itself provide adequate augmentation, but secondary combustion and rocket jet altitude compensation provided by an air augmentation device may prove advantageous. Liquefied air augmentation, mentioned in previous sessions, is also a new and interesting idea, but presently it appears to be horrendously complex.

COMPARATIVE POWERPLANT PERFORMANCE

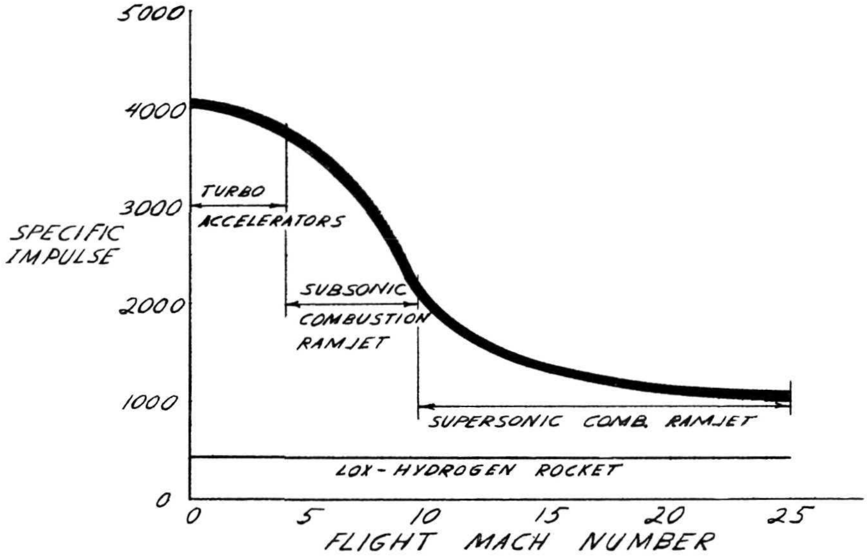

Figure 1

FEATURES OF AIR-AUGMENTED ROCKET

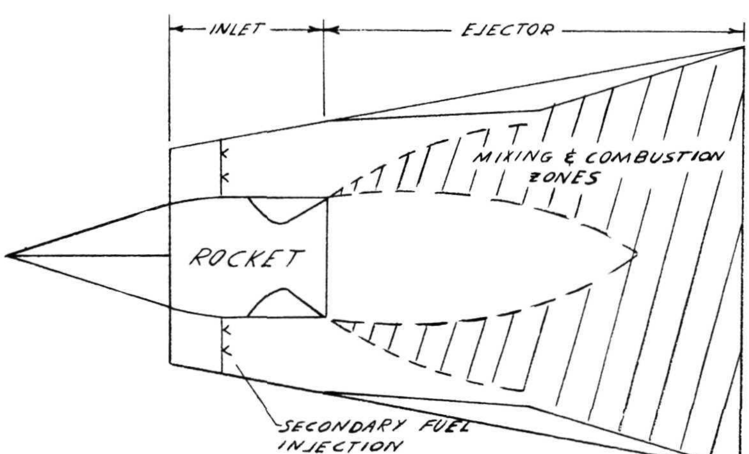

Figure 2

SUPERSONIC INLET PERFORMANCE

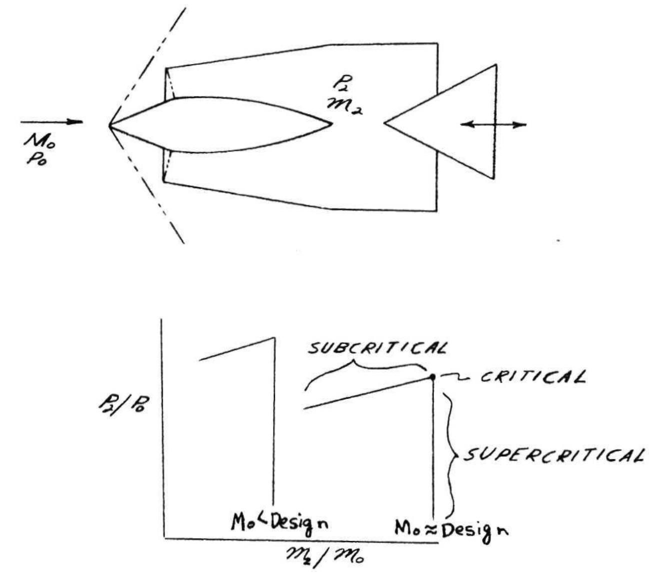

Figure 3

VEHICLE INTEGRATION

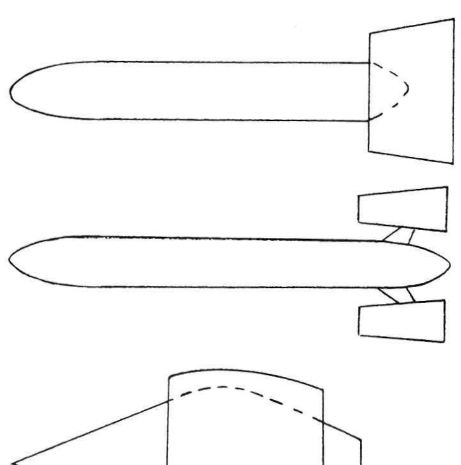

Figure 4

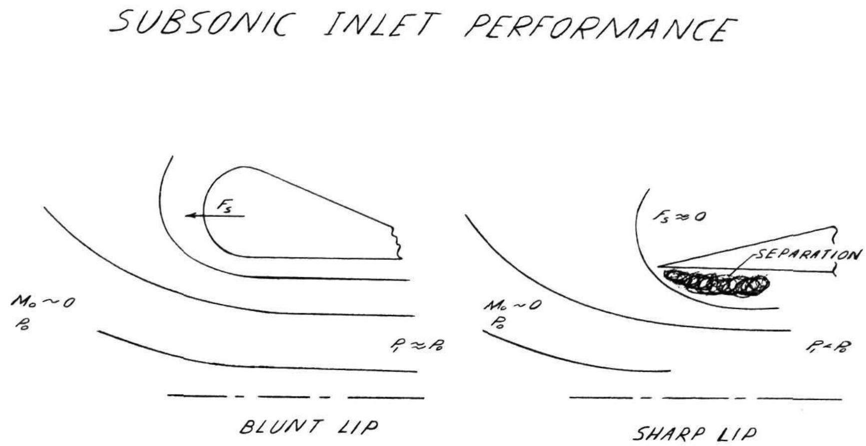

Figure 5

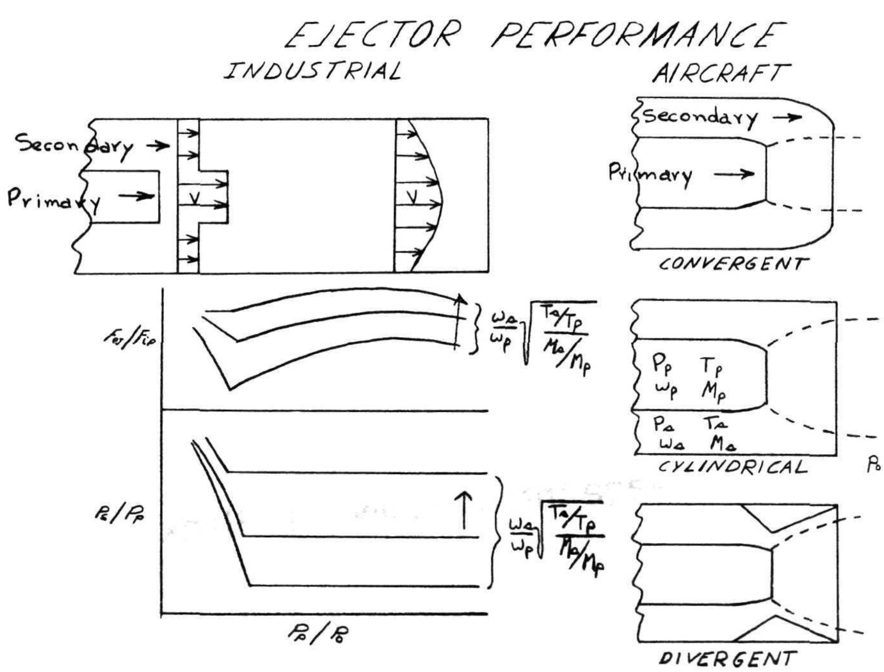

Figure 6

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29. REACTING FLOW CALCULATIONS AND NOZZLE OPTIMIZATION

By T. R. Mariani

Lewis Research Center, NASA

Reacting flow calculations and, especially, nozzle optimization techniques are certainly areas of interest in delineating better design criteria whereby large liquid rocket engines can be developed.

The first part of this presentation deals with the state-of-the-art; a search was conducted to determine what type of computer programs are available. Figure 1 shows the work that was conducted at Lewis and at Aerojet. These programs use a one-dimensional flow analysis with the Bray approximation. This figure also shows a two-dimensional program available from Marquardt, developed under contract for the Air Force. It also uses the Bray approximation and results in a freezing boundary instead of the freezing point.

Figure 2 shows one-dimensional equilibrium programs. The Rocketdyne program includes kinetic flow in the nozzle but considers only the three-body reactions. Figure 3 shows kinetic flow programs. These programs consider reaction kinetics in an exact calculation. However, they are research programs - they will not be available for at least 6 months, maybe as long as 18 months. One of the complex features of these programs is that they must start from equilibrium. The introduction of a consistent set of initial conditions has, in fact, eliminated the starting problem. However, the general procedure for calculation of the initial conditions remains to be done.

Part II of this presentation is consideration of a program, funded by the Office of Advanced Research and Technology, and conducted by the United Aircraft Corporation (the Research Laboratories, East Hartford, Connecticut, in cooperation with the Pratt & Whitney Division) under contract NASw-366.

The technical objectives of this program were to conduct order-of-magnitude calculations to determine the relative effects on nozzle performance of lack of equilibrium in the nozzle due to continuing reaction or recombination lag, thermodynamic relaxation, velocity and thermal lags associated with two-phase flow, condensing flow, and heat transfer to a wall with a nonequilibrium boundary layer.

The propellant combinations studied under contract NASw-366 were

hydrogen-oxygen $(H_2\text{-}O_2)$
hydrogen-fluorine $(H_2\text{-}F_2)$
diborane-oxygen fluoride $(B_2H_6\text{-}OF_2)$
hydrazine-nitrogen tetroxide $(N_2H_4\text{-}N_2O_4)$
beryllium-hydrogen-oxygen $(Be\text{-}H_2\text{-}O_2)$
hexane-oxygen fluoride $(C_6H_{14}\text{-}OF_2)$

[redacted]

The H_2-O_2 combination was used for all the calculations. The H_2-F_2 and the C_6H_{14}-OF_2 combinations were also studied, but the results were significant only for the Bray analysis that was made. The B_2H_6-OF_2 combination was studied, but it was determined that it played an important role for only condensing flow losses, and they were insignificant. The N_2H_4-N_2O_4 combination was considered important especially for the vibration-relaxation calculations. The two-phase flow calculations considered the Be-H_2-O_2 combination.

The second portion of the program was to refine a UAC computer program by including the one-dimensional reacting gas equations for a variable passage. This work has been completed and the program can be used to determine the effects of throat contouring, freezing point locations, various combustion efficiencies, and at the same time consider chemical kinetics.

Two other programs are being developed. They are two dimensional and consider finite chemical kinetics. One of these programs can be used to evaluate performance for a prescribed nozzle contour. The other is a design program and can be used to determine optimum nozzle contours. Both of these programs have been run but are not operational.

A fourth portion of the program was to check out the validity of the two-dimensional computational procedures, and this was done by performing the experiment to study the reaction for the recombination of nitrogen dioxide $\left(NO_2\right)$ to form nitrogen tetroxide $\left(N_2O_4\right)$. This was done in the hot test wind tunnel.

Some of the results are as follows.

The order-of-magnitude calculation employed a "sudden freezing" criterion in a hypothetical high expansion ratio exhaust nozzle and showed that recombination lags can reduce the nozzle specific impulse from the theoretical maximum value obtainable under equilibrium flow for all propellant systems considered. The reduction in impulse is largest at the lowest pressure employed in the analysis, which was 60 psia; ranges of losses were from about 6 percent to 11 percent at this pressure at an expansion ratio of 100. Reductions in nozzle specific impulse were less than 1 percent at pressures to about 1,000 psia.

The order-of-magnitude calculations for vibrational relaxation freezing point for the propellant combinations studied under the first portion of the program suggest that no significant reduction in performance will occur because of vibrational freezing except in the case where diatomic nitrogen is present in large concentrations. Performance losses due to the presence of nitrogen may be appreciably reduced if water is also present in the exhaust nozzle flow. The freezing area ratios are generally greater than 10 to 1 at a chamber pressure of 60 psia, and increase markedly as the pressure is increased.

Table I shows a comparison between chemical recombination and vibrational relaxation. The N_2H_4-N_2O_4 combination was used at chamber pressures of 60 and 1,000 psia. These calculations are hand calculations made with a Bray-type analysis for chemical recombination and vibrational relaxation. The calculations show that for optimum ratio and a conical nozzle, the freezing points for

TABLE I.- SAMPLE BRAY CALCULATIONS USED TO ASSESS THE IMPORTANCE OF CHEMICAL RECOMBINATION AND VIBRATIONAL RELAXATION

$N_2H_4 \rightarrow N_2O_4$

OPTIMUM O/F FOR CONICAL 15° HALF-ANGLE NOZZLE

Chamber pressure, psia	Critical freezing area ratio	
	Chemical (H_2O)	Vibrational (N_2)
60	1.015	2
1000	2.55	10

the vibrational relaxation approach the freezing points for chemical recombination. For these cases we must consider the losses due to vibrational relaxation and recombination since freezing occurs close to the throat.

These calculations will hold also for the 50-50 aerozine combination since the reaction equations that we considered here would be the same for the 50-50 combination.

This study indicated the importance of the coupling of chemical recombination and vibrational relaxation when considering nonequilibrium effects. The Marquardt people claim their program (noted on fig. 3) considers vibrational relaxation along with chemical recombination.

Performance losses can be considerably increased because of the velocity and thermal lags associated with two-phase flow. For the $Be\text{-}H_2\text{-}O_2$ propellant combination in a typical nozzle configuration, specific impulse reduction can range up to 10 to 20 percent of the ideal impulse, depending upon the chamber pressure, particle size, and thrust level. These losses increase as particle size increases and decrease as the pressure and thrust level increase.

Losses in performance due to condensation are associated only with the $B_2H_6\text{-}OF_2$ system of the six propellant combinations studied. However, condensaoccurs well downstream of the nozzle throat and specific impulse losses are, therefore, not significant.

Changes in the heat flux to nozzle wall are of small magnitude when nonequilibrium recombination of the $H_2\text{-}O_2$ system is permitted in the boundary layer. Under the most severe condition of a perfectly catalytic wall in the vicinity of the nozzle throat, the heat transfer associated with the complete recombination at the wall is only 5 percent of the total heat transfer at the throat.

Specific impulse losses for the H_2-O_2 system computed by means of the one-dimensional kinetic machine program for nozzles with varying throat configurations amounted to from 2.5 percent to 4.3 percent of the ideal equilibrium impulse at an area ratio of 400 to 1. The assumed chamber pressure was 60 psia and thrust level was 150 pounds.

A low thrust was chosen for this case because of the limited amount of machine time available. One of the limiting factors for calculations of this type is the time it takes to compute the performance, and as the thrust goes up the time becomes excessive. Actually, on the basis of this calculation, it was determined that nozzle throat contour changes do not appear to provide significant effects on the performance.

The experimental portion of the program was successful. Good agreement was obtained between predicted and measured reactant compositions for a two-dimensional test nozzle in which the recombination of NO_2 to form N_2O_4 was followed. It is concluded from this agreement that the calculation procedures employed in the two-dimensional finite kinetics machine programs are valid and provide results representative of the supersonic flow field.

For the future and as an extension to this work, we are planning to initiate additional studies in order to validate the calculations, to determine some guidelines whereby we can use these programs, and to reduce machine running time. We plan to study in detail the sensitivity of performance calculations to reaction rate data because the reaction rate data that are available, in many instances, are questionable. We would like to know how sensitive the computations are to reaction rates. This portion of the program will use the H_2-O_2 and 50-50 combinations.

Calculations will be made as a result of the exact kinetic and Bray-Type analyses. These calculations will be compared to determine the limits of applicability of the Bray criterion.

Analytical studies will be performed to determine nozzle performance under assumptions of shifting equilibrium and kinetic flow. These calculations will be made for various combustion efficiencies and nozzle contours. The calculations will be compared to determine the effect of the gas model on nozzle performance. We hope to answer the question of what type of gas model to use in considering kinetic flow.

Kinetic flow calculations on the basis of fixed chamber conditions will be made for various thrust levels. These calculations will be compared to determine a scaling criterion whereby nozzle performance for small thrust levels can be used to predict nozzle performance for larger thrust levels.

In summary, it is important to know that the one-dimensional kinetic flow program can be used to calculate all the variables of interest in both the subsonic and supersonic passages of variable cross-sectional area. A subsonic or supersonic combustion chamber may be followed by a variety of convergent-divergent, or divergent nozzles. A nozzle calculation can be initiated from a set of prescribed conditions without calculating a combustion chamber including starting directly from equilibrium. Various combustion efficiencies can be included by fixing the length of the combustion chamber or by prescribing the final slope of the temperature distribution.

The program can be used to study the effect of various throat contours. The running time of the program is dependent on the number of reactions and species involved; the maximum errors allowed; and the proximity to equilibrium, chamber pressure, and/or thrust level. The program cannot compensate for shocks in the flow or automatically locate an effective freezing point. The integration procedure can solve a maximum matrix of twenty by twenty, that is, twenty species and twenty reaction equations, and a minimum of two by one.

For the two-dimensional kinetic flow nozzle programs, the method of characteristics is employed. The calculations take place by integrating the reaction kinetics equations along the segment of streamline employed in the construction of a mesh point. The performance calculations can be made to determine the flow field associated with a given supersonic nozzle contour and prescribed conditions for the flow entering the throat. Calculations can be made to determine the coordinates of a family of modified perfect nozzles from which optimum nozzle contours can be selected. These modified perfect nozzles have axial exit flow but not necessarily uniform speed.

[redacted]

ANALYTICAL DESIGN & EVALUATION TECHNIQUES FOR ROCKET ENGINES

AEROJET
LeRC - L. FRANCISUS
ONE-DIMENSIONAL; BRAY APPROXIMATION

MARQUARDT - AF-33(657)-8491
TWO-DIMENSIONAL; BRAY APPROXIMATION

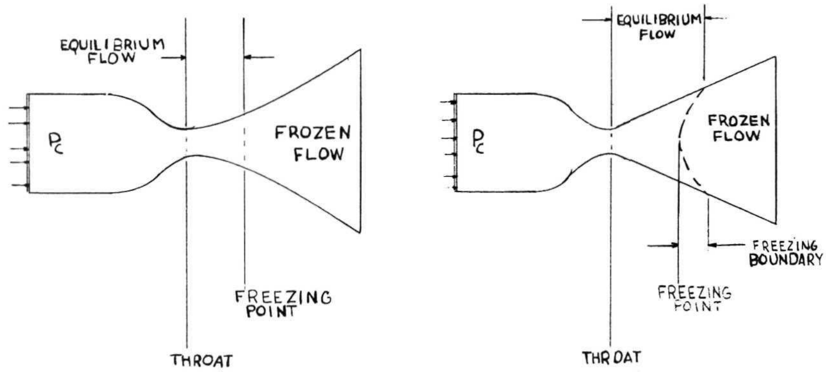

Figure 1

ANALYTICAL DESIGN & EVALUATION TECHNIQUES FOR ROCKET ENGINES

ROCKETDYNE RESEARCH DEPT
ONE-DIMENSIONAL; EQUILIBRIUM-KINETIC

LeRC - S. GORDON
ONE-DIMENSIONAL; EQUILIBRIUM

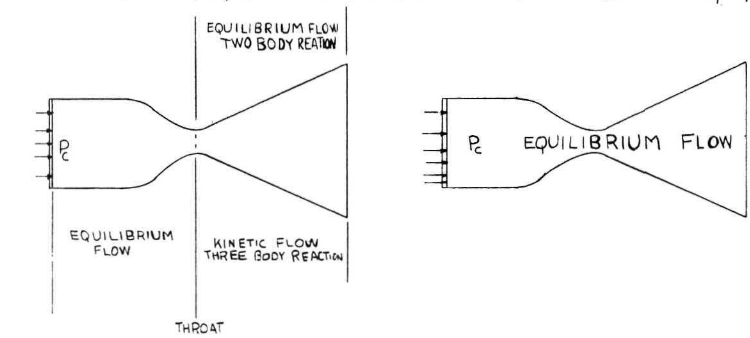

Figure 2

[redacted]

ANALYTICAL DESIGN & EVALUATION TECHNIQUES FOR ROCKET ENGINES

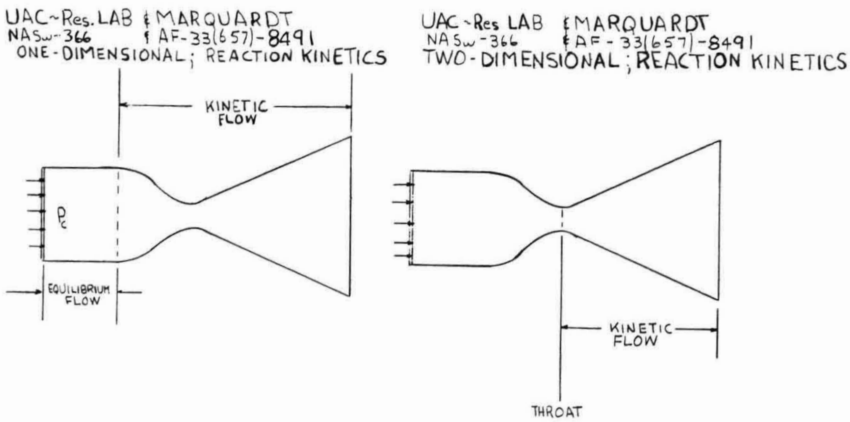

Figure 3

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30. SUMMARY DISCUSSION

MR. THOMPSON: I have a question for Mr. Connors. Were you taking net thrust minus drag with your large center air flow model?

MR. CONNORS: Yes, we did.

MR. THOMPSON: Your performance curves were then based on that -

MR. CONNORS: It was stricly nozzle thrust. We had momentum breaks in the inlet and exit of the bleed passage, and we ran tear loads on any configuration of lip drag. What I presented is strictly nozzle thrust.

MR. THOMPSON: You showed there a compensation effect which seemed to be in the lower pressure range, something more than 54 percent.

MR. CONNORS: More like 100 percent.

MR. THOMPSON: Okay.

My second question: You mentioned that the Lewis laboratories are going to engage in a nozzle program next year investigating various nozzle concepts. My question is, why did you decide to do it - should I put it this way: You mentioned your wind tunnels that you have available there, so I thought that you might be doing it in-house. Maybe I should phrase my question this way. Are you going to do your work all in-house or contract it out?

MR. CONNORS: All in-house. I think it is rather more that we do it in-house so that we have a little control over the configurations that enter into the program. I think there are a lot of unanswered questions about - aside from the details of measuring technique and getting a common frame of reference for these thrust measurements which are not the easiest things to come by, I think it is important that we look at such configuration variables as discrete throats, voids in the annular configuration.

MR. THOMPSON: Can you amplify why you don't think you would have proper control if you went to a contractor?

MR. CONNORS: I think it is very important, if you have done any nozzle work in the past, to check the instrumentation that is coming up with thrust measurements. There is also the question when you are moving from one facility to the next of the calibration techniques. Unless you have a bast bulk of data to base this on, I think it is well to have one test environment.

MR. THOMPSON: You made another statement at the start of your talk which I thought was quite intriguing, something about control on these things the past two years or so, concepts going this way and that. I didn't quite understand that. If that is the case, why would you say we haven't been able to exercise closer control over new nozzle concepts?

MR. CONNORS: That is a pretty good question.

The control I am talking about is control of the concepts that go into the design. Each company has in turn come up with their variations of an altitude-compensated nozzle, and they have all taken their own prerogative of putting their own particular label on it. I was sort of bemoaning the fact that we haven't had a chance to discuss the design or provide any input to the NASA effort in this, and I think for a long time we have been, if you will, looking ahead and seeing what the problems would be in attaining altitude compensation.

The aerodynamics are rather straightforward and to be able to control the free expansion service you need to be able to provide this ambient pressure service. We couldn't see how you could do it on the unvented configurations proposed. When you are talking about discrete engine modules, as in the Aerojet, I think that is a very interesting approach. I think there is a lot of aerodynamics involved in deciding what sort of engine spacing and how much vent air you will have to provide.

It is difficult at this point to determine what is the minimum center core bleed air. I think this is one of the variables we would like to inject into our own program.

MR. THOMPSON: Do you think the nozzle that you designed would be comparable in weight to the, shall we say, Rocketdyne or Aerojet type nozzle, the one in which you obtained close to 100 percent compensation?

MR. CONNORS: You missed the point. This isn't a flight configuration. This is a basic aerodynamic study, and it wasn't intended to be a flight configuration. It was more an exploration of aerodynamic principles.

I think the concept of going to the individual modules and circular throats, like that of Aerojet, is interesting. I don't think it is well to look at our particular test hardware as a flight vehicle on which to base any weight analyses. I tried to indicate in one of the follow-up sketches that our test model was subjected to the expediency of the test. We were using cold air, we had to have long hollow struts and a settling chamber before we brought the air out through the nozzle. You have a different concept when you are talking about - say you did go along with an annular combustor, then you can shorten up the configuration.

I think this is one of the important questions, the weight analysis. My own feeling is that we ought to get into a detailed vehicle structural analysis.

MR. THOMPSON: I agree with you 100 percent in your concept which seemed to show such good performance if it could be made comparable weightwise.

MR. BEHEIM: I have a comment on Mr. Thompson's question on in-house work. At Lewis we think that it is important to maintain competence through an in-house program within our center, and certainly it is one means of keeping things objective.

In addition to our own in-house work, we can test selected models consistent with our schedules and resources, of configurations and using test procedures which are of technical merit in our opinion, upon the specific request by another group, and approved by headquarters.

MR. CONNORS: I think the Nova studies might have been a little more profitable if there had been tighter restriction to the range of design variables.

MR. WILLIAMS: This is not said in a provocative manner. It has been brought up a couple of times on the restrictions on the Nova study, as well as being able to provide the necessary direction and support not only to the Nova studies but to any study. Quite frankly, I can't speak for all of Marshall, for those studies that we have been involved in, and they are numerous, with I think a very wide cross section of industry, we have had absolutely no trouble at all. In fact sometimes they respond too rapidly to our wishes for changes in requirements or desires. But as far as the Nova study, per se, is concerned, we have exercised rather stringent control over what the contractors do.

In fact, we almost approve, not by the manhour, but by the expenditure of resources. We have given them quite detailed guidelines, a volume some 6 inches thick of just guidelines and specifications of a technical nature, even of a procedural nature.

Granted these may not be adequately defined or there may be desire or need for further control. This becomes a resource problem within the government itself.

I would like to at this time offer an opportunity to any of you to work with us on the Nova study. We need better guidelines. We need restrictions in certain areas. We also, I might add, on the other hand, have people who are trying to open up all the doors.

Let's look at everything from plasma propulsion down to low P_c pressure-fed systems, launched at sea. We have even had proposals by headquarters people that we hang Nova between wires between mountains and launch them out of things like this - and sled proposals, fluorine all the way.

We have provided, I think, quite a bit of restriction. I solicit your support and interest in this area, and any ideas you might have I am sure will receive considerable attention before they are either incorporated or excluded.

Thank you.

MR. WEIDNER: I would like to add one thing that Mr. Connors said about our contractors. Since I was the one responsible for bringing the engine contractors in here yesterday, I would like to say what these people essentially offer to us was not the result of NASA contract work but of their own doing. We at NASA have somehow failed as part of the Nova studies to make these people a contracted part of our arm over here. Therefore, I think I must in their absence somehow speak for them here.

I think what we see and have heard is a mixture of, (a) their own thinking, and (b) their sales department type of spirit. But certainly it is not something which is in a gross way controlled by us or not controlled by us, and, therefore, I think you must somehow see the position.

It is a noncontracted kind of effort which we are seeing here. Maybe this is something we should correct in the future by really making these people an arm of our efforts here so that we can exert some influence.

MR. CONNORS: Eliminate the sale department.

MR. WEIDNER: That is right. Exactly what I mean.

MR. MARIANI: I think all the research going on at Lewis in these fields provides support for our contract work when it is fitting, along with the rest of the work that is being sponsored out of Headquarters. This is one area, especially with the experience our research people have, that we should look into very seriously, along with the other work that is going on.

MR. THOMPSON: This was really the only point I was trying to bring out, that it does take a proper mixture of in-house and contractor effort to accomplish the best out of advanced technology. You cannot accomplish it all by doing it all with contractors, and you cannot accomplish it all by doing it all in-house.

It won't stand alone on one leg or the other.

MR. CONNORS: You misjudged what I said. I was confining my remarks to a one-tunnel experimental program, one of evaluating nozzle aerodynamics. I think that this ought to be all done in one facility. I wasn't taking in the whole scope of the big engine problem.

MR. BEHEIM: Mr. Sloop.

MR. SLOOP: Mr. Thompson, you left one thing unsaid for the rest of the group that I think should be commented on here, and that is that a small and almost spontaneously organized coordinating group, consisting of Thompson and Beheim and Pierpont from Langley, have gotton together several times now and discussed air augmentation work. I assume that they can't separate from other nozzle work. I think this is a good example of intra Center discussion and coordination. We certainly hope more of this will occur in the future.

MR. GRAHAM: Listening today to what is happening in propulsion, hearing this morning about boost pumps and varieties of turbines and systems sitting around, and then hearing about nozzles, altitude compensation, and air augmentation this afternoon, I begin to wonder about the control concept of this kind of a vehicle. I have heard nothing about this, about partial thrusts, terminal velocity, precision, and so forth in this whole mission concept. This is something I think that we need to keep in mind as we begin to think about these kinds of devices that we are throwing into this whole propulsion system.

MR. BARTZ: For Mr. Mariani's information there is one other program that is available now, for a price. James Kleagle, at Space Technology Laboratories, does have a two-dimensional finite kinetic program that is working, and it is working now to reduce the computer time by expanding it about the equilibrium case.

Apparently, as equilibrium is approached these things get very difficult. He is now expanding it as differences about the equilibrium case, but he has a program working for the normal way of doing it.

The next step he is taking is this expansion around the equilibrium case. This has been done on STL's own money, but I am sure that for some kind of price it will be for sale.

SESSION VII

THRUST VECTOR CONTROL

Chairman: Milton A. Beheim
Lewis Research Center, NASA

[redacted]

31. INTRODUCTION

By Milton A. Beheim

Lewis Research Center, NASA

Steering control is required during all phases of flight, obviously, to correct for thrust misalinement and to provide engine-out flight capability, to overcome wind jet stream effects at altitudes near 35,000 feet, to provide pitch control during trajectory maneuvers, and to overcome unwanted perturbations during engine shutdown and staging.

The conventional technique of steering launch vehicles is by means of thrust vector control. The usual question that must be resolved is to determine the thrust vector control technique that provides the highest reliability with the lowest weight. When undertaking development of the new concept, however, such as a large rocket motor, it is appropriate to review other means of steering control if for no other reason than to be sure that thrust vector control is still the best choice. Other means that have been studied in the past include auxiliary rocket motors and aerodynamic surfaces, or some combination of both. Obviously, the aerodynamic surfaces are useful only in the high dynamic q portion of the flight.

Installation of these devices on a vehicle is schematically illustrated in figure 1. This figure schematically illustrates that fins or rockets may be used in aft or forward locations. The forward locations have some advantage over the aft locations for the following reasons: the larger moment arm from the center of gravity decreases the steering force requirements, it decreases the normal acceleration of the vehicle since the steering force acts counter to the body normal force, and the maximum internal bending moments are reduced.

In addition, there is the possibility that the rocket motors in the forward location might provide steering for all stages.

A novel means of combining the auxiliary motor with an aerodynamic surface is illustrated in figure 2. Here the nozzle exit is housed in the base of a canard fin with the rocket motor itself at the root of the fin. It is a highly swept configuration which has been tested for its thrust performance at Lewis. That performance is shown on the upper portion of the slide.

Since it is an external expansion nozzle, it maintained high thrust coefficient over a wide range of pressure ratio.

In addition to these tests, canard fins will be investigated on the Atlas-Centaur configuration in a forthcoming test in the Lewis wind tunnels. They will be evaluated as a means of decreasing the bending moments during flight through the wind jet stream at transonic speeds.

A correct evaluation of these auxiliary devices is difficult since the efficiency of such a device is sensitive to the mission and to the vehicle details.

The problems to be considered include the control of aerodynamic surfaces which must be deflected and the control of multi-restart rocket motors, and the change in location of center of pressure and center of gravity during flight and during staging.

Studies to date have shown that the auxiliary devices cause weight penalties substantially greater than thrust vector control, primarily because the auxiliary rocket motors are not small.

However, a factor that has not been satisfactorily considered in some studies, because of its sensitivity to mission, is the possibility of multipurpose use of these devices during all phases of a complete mission.

To carry this multipurpose concept to an extreme as a means of illustration, the jet canards on the final stage possibly could provide steering control of all stages during launch, provide final stage propulsion into orbit and during space maneuvers, and provide propulsion and steering control during reentry and landing. The multipurpose technique may prove desirable for some applications but not others.

Because development of large engines generally precedes knowledge of the missions for which they are to be used, it may be necessary to select thrust vector control simply because of its flexibility for a variety of missions.

Assuming, for the time being, then, that steering control is required from the main engines, in this session we shall consider the possible solutions for thrust vector control of large engines. The subjects to be covered are hot gas injection, reactive fluid injection, and mechanical systems.

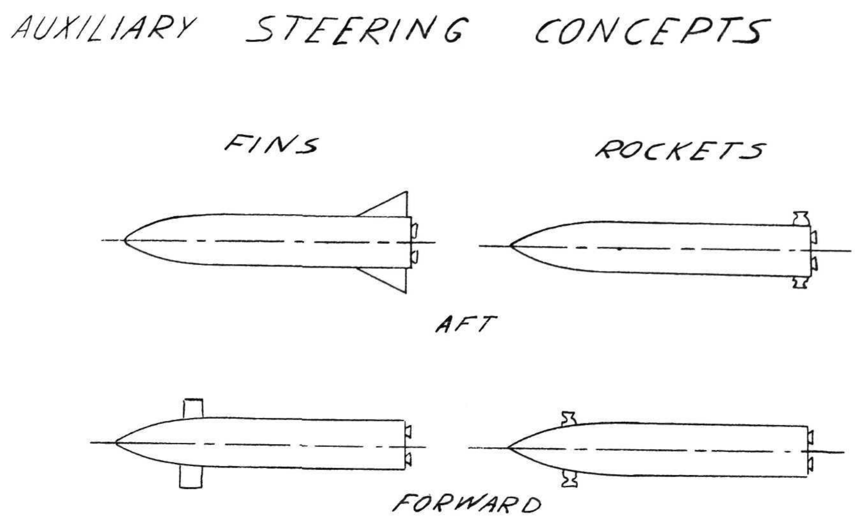
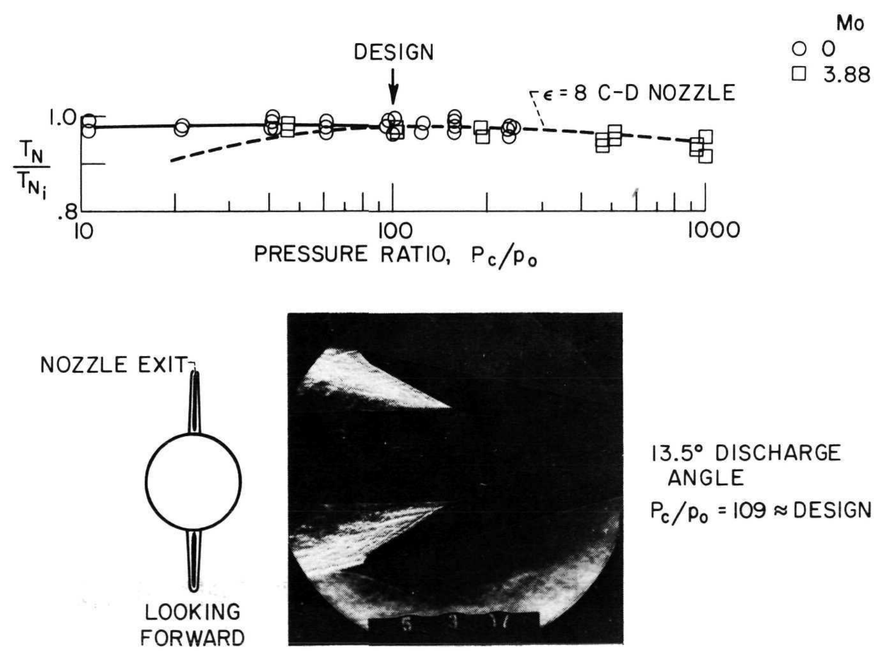


Figure 1



JET CANARD PERFORMANCE

Figure 2

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32. HOT GAS INJECTION

By Donald D. Thompson

George C. Marshall Space Flight Center, NASA

One of the proposed methods for obtaining directional control of advanced vehicles is the use of secondary gas injection into the rocket nozzle. As a result of this evidence of feasibility, considerable effort has been applied recently to evaluate the concept.

The investigations and studies have been numerous, and the applications considered have run the full range of both present and advanced space vehicles.

One of the present studies considered to be representative of the present state of the art is contract NAS8-5070 with the United Aircraft Research Center.

Background developments on the subject are sketchy because the theory is, up to this point, not very well defined. Therefore, in discussing application, some coverage must be given to those areas which seem to be inadequately defined at this time.

The use of secondary injection to obtain thrust vector control was conceived about 1949. The concept was first investigated and shown to be feasible in the work of Hansmann of the United Aircraft Research Laboratories in 1953. This work, which was limited to gaseous injection into nozzles having low area ratios, showed that the lateral force produced by secondary injection into a rocket nozzle is significantly greater than the theoretical value for a system utilizing a separate vernier nozzle.

Early investigations here and elsewhere employed sonic circular secondary injection ports located in the walls of the primary nozzle. Recent investigations have shown that significant improvements in performance can be obtained by employing a secondary injection port which directs the flow upstream through a circumferential slot (or equivalently through a circumferential array of holes) located at the nozzle exit.

In addition to the improved performance, this configuration is more practical for fabrication since it does not require the structure of the primary nozzle to be altered to provide for secondary injection.

The parameters which are of significance in understanding a secondary injection system may be cataloged as primary and secondary nozzle variables. The secondary nozzle variables considered are the weight-flow ratio, the angle of injection, the width of the injection region, the injection Mach number, and the secondary gas properties. The primary nozzle variables considered are the Mach number or area ratio of the primary nozzle, the pressure ratio, the size of the nozzle, and the gas properties of the primary flow.

A schematic diagram of the flow model employed is shown in figure 1. The secondary flow is injected through a flush port at the nozzle exit; it is injected upstream with an angle, α, measured relative to the perpendicular to the nozzle axis. Thus, positive α implies upstream injection and negative α implies downstream injection. To the right is an end view of the nozzle showing the wide injection port; the width of this injection is expressed in terms of the circumferential angle, ψ.

The lateral force produced by secondary gas injection is the sum of the secondary jet reaction force and the induced pressure force, marked by arrows, in the interaction region upstream of the injection port.

It is primarily a function of the Mach number, injection angle, and the secondary weight flow. The lateral induced force is the vertical component of the induced pressure area forces in the interaction region. These induced forces are evaluated by treating the secondary flow on a blockage element which causes shock induced separation of the primary flow.

The high pressures in the separated region create a force on the primary nozzle wall and a rearward force on the secondary stream turning it in a downstream direction. This rearward force is equal to the change in axial momentum of the secondary flow. The relationship between the induced wall force and rearward force is evaluated from separation characterization of a turbulent supersonic boundary layer. Typical shock induced separation data for a blockage element are shown in figure 2.

The pressure distribution on the wall in front of the blockage element is shown in the left-hand plot of figure 2, and the pressure distribution on the face of the blockage element is shown in the right-hand plot. This figure shows the following important features: the average base pressure, P_B, is greater than the average wall pressure, P_W, which in turn is greater than the measured separation pressure, P_S. The ratio of the net induced wall force to the total rearward force or base force is called the separation parameter, G. The value G obtained by integrating the pressure difference along the wall and the pressure on the face of the step for the data shown is 2.13.

Thus, from these data it could be concluded that the net induced wall force for a two-dimensional interaction is 2.13 times the total rearward force which is equal to the change in axial momentum of the secondary stream.

For the analysis, the flow was assumed to be separated by a two-dimensional shock giving the step rise in pressure as shown by the dotted line. The value of the separation pressure rise was obtained from a data correlation by Chapman, Kuehn, and Larson; the average wall pressure rise was assumed to be 50 percent greater than the separation pressure rise; and the average base pressure rise was assumed to be 50 percent greater than the average wall pressure rise.

The theoretical value of G obtained by analysis is 2.0. The value of G for a semicircular blockage element was obtained in a similar manner by employing conical shock relationships. The theoretical value of G obtained in this manner was approximately 1.3 compared to 2.0 for the 2-D case.

The values of G for the two-dimensional and semicircular blockage elements set an appropriate level of values to be used in the side force calculations; however, it is necessary to evaluate the conditions under which the two-dimensional, the conical, or intermediate values should be used.

On the basis of limited data obtained with physical blockage elements, intermediate values of G were found to correlate with the aspect ratio of the blockage element. The aspect ratio is defined as the width divided by the height of a rectangular blockage element.

The reaction and induced forces discussed previously were combined into a single expression shown in a simplified form in the following equation:

$$\frac{F_2}{F_{1_0}} = \left[\phi_R + CG\phi_i\right] S\frac{W_2}{W_1}$$

This equation relates the total side force, F_2, to the nonvectorized axial thrust, F_{1_0}.

The jet reaction term ϕ_R is a measure of the lateral component of the secondary jet reaction; it is primarily a function of the secondary Mach number and injection angle as mentioned previously.

The induced term, ϕ_i, is a measure of the base force which acts on the secondary jet turning it rearward; it is a function of the primary and secondary Mach numbers, the injection angle, and the pressure ratio of the primary nozzle.

The separation parameter, G, is defined as the ratio of the induced force to the base force; it is a function of separation characteristics as discussed previously. Thus $G \times \phi_i$ is a measure of the induced wall force.

The spreading parameter, C, is defined as the ratio of the directed induced force to the total induced wall force. It is a measure of the efficiency of the induced pressure forces in producing a side force in the desired lateral direction; the inefficiencies result from lateral spreading of the interaction region to the sides of the nozzle where the incremental pressure area forces are not parallel to the desired force direction. Thus, $C \times G \times \phi_i$ is a measure of the directed induced force.

The similarity parameter, S, is directly proportional to the square root of the temperature and inversely proportional to the square root of the molecular weight of the secondary gas and is a weak function of the ratio of the specific heats. For primary and secondary gases having equal gas properties, the value of S is unity. The ratio $\frac{\dot{W}_2}{\dot{W}_1}$ is the ratio of the secondary to primary weight flows.

Although the expression for total vectoring force appears to be linear with the injected weight flow ratio, it is not, because the parameters C and G are functions of the injected weight flow ratio. The resulting side force is, therefore, nonlinear with the weight flow ratio in such a manner that the amplification factor, K_2, decreases with increasing weight flow. The lateral amplification factor, K_2, is defined as:

$$K_2 = \frac{F_2/F_{1_0}}{W_2/W_1} = \frac{I_2}{I_1}$$

The amplification factor, K_2, is important as it is the key parameter in judging secondary injection performance.

This analytical expression, equation (2), was then employed to evaluate the importance of the various parameters affecting secondary injection performance. Where possible, the calculations were made to show a direct comparison with existing data.

Figures 3 to 5 show data obtained in cold-flow tests using air for both the primary and secondary gases.

In figure 3, as well as in figures 4 and 5, the theoretical performance is shown by solid lines and experimental performance by the data points. The variation of the side force ratio with the weight-flow ratio is shown for sonic injection through both a slot with upstream injection and through a circular port with injection normal to the wall.

For both cases, the theory and experiment are in good agreement even to the extent of the nonlinearity of the force-weight-flow relationship. The slot configuration obtained amplification factors (K values) which ranged from 2.4 to 1.6 compared with the value of approximately 1.2 for the circular port.

It should be noted that a separate vernier nozzle as efficient as the primary nozzle would have $K = 1$, and that, when values are greater than 1, it can be inferred that lateral specific impulses greater than that of the primary nozzle can be obtained by secondary gas injection.

In order to select an optimum secondary injection configuration, it is necessary to evaluate the effects of the various geometric parameters which define an injection port.

As shown in figure 4, the parameters are the upstream injection angle; the slot width, defined by ψ; and the area ratio or Mach number of the secondary nozzle.

In these plots, the theoretical performance obtained by varying each of these parameters separately is expressed in terms of the lateral amplification factor for 4 percent secondary weight flow. The results show that, increasing the injection angle, the slot width, and the Mach number all cause improvement in performance.

These results help explain why the slot configuration obtained superior performance to the circular port as shown in the preceding slide; i.e., the circular port subtended an angle of less than 10 deg compared to the 45 deg subtended by the slot, and the circular port injected the flow perpendicular to the nozzle wall at an angle of -15° compared to $+30^{\circ}$ for the slot configuration.

The effects of the primary nozzle parameters are shown in figure 5. These parameters are area ratio, Mach number, and size, expressed in terms of a Reynolds number. The performance is expressed in terms of the lateral amplification factor for 2 percent secondary weight flow. As shown in the top graph, the secondary injection performance is significantly improved as the pressure ratio of the nozzle is reduced below the design point.

This improvement reaches a maximum near the pressure ratio at which the primary nozzle begins to separate. Below this pressure ratio the secondary injection becomes ineffective.

This improvement could be very significant during rocket launch because most rocket nozzles are overexpanded but not separated during the low altitude portion of their flight and because the largest corrective forces are frequently required at this time.

Thus, the 25 percent increase in side force shown above may allow a 25 percent smaller control unit to be installed in the nozzle. The effects of nozzle area ratio and size, expressed in terms of a Reynolds number, are shown in the bottom two plots. The TVC performance is very insensitive to both of these parameters. This is a fortunate happenstance because it indicates that data obtained with small-scale models having an incorrect area ratio can be employed with only minor corrections to predict the performance of full-scale TVC systems.

The preceding discussion was based on calculations and data for room temperature air in both the primary and secondary nozzles. The value of using these data to predict the performance of a rocket nozzle employing hot gases other than air depends upon the validity of the similarity parameter.

The effect of dissimilar molecular weights, as shown in figure 6, were obtained in cold-flow tests employing air as the fluid for the primary nozzle and helium (molecular weight of 4), or argon (molecular weight of 40) as the secondary injectant. Also shown in this figure are the reference values obtained from the air-air tests of the slot configuration shown previously. As plotted in terms of the force and weight-flow ratios, the data do not correlate; the light helium injection shows extremely good performance, better than twice that of the air data, whereas the heavy argon is poorer than air injection. When replotted in terms of the corrected weight-flow ratio, $S(W_2/W_1)$, the data for all three gases are in agreement.

Similar results are shown in figure 7 which presents data obtained from small-scale hydrogen/lox firings. When plotted in terms of the force and weight-flow ratios, the data do not correlate with the cold-flow air-air data presented previously; however, when replotted in terms of the corrected weight-flow ratio, the data fall about the corrected value for a hot primary nozzle.

This figure and the previous one showed that data with different primary and secondary gases can be correlated by employing the similarity parameter which is proportional to the square root of the temperature and inversely proportional to the square root of the molecular weight of the injected gas. This relationship forms the basis for investigating a TVC system for a hydrogen/lox rocket which obviates the need for a hot-gas valve capable of operating at combustion chamber temperatures. The proposed TVC system would bleed a small fraction of the primary combustion products, and these gases would be cooled by diluting with cold gaseous hydrogen, thereby reducing both the temperature and the molecular weight as shown in figure 8. The results are plotted as a function of the mixture ratio, MR, which is defined as the ratio of cold hydrogen to total secondary flow; thus MR = 0 represents the use of direct combustion chamber products and MR = 1 represents the use of cold hydrogen. As the mixture ratio is increased from zero to one, the gas temperature is reduced from approximately 6000° to 500° R, and the molecular weight is reduced from 12 to 2. The TVC performance, expressed in terms of the lateral specific impulse that could be obtained from this system is shown in the bottom curve. This figure shows that side specific impulses in excess of twice the primary rocket, approximately 430 seconds, can be obtained over a wide range of mixture ratios. More specifically, gas temperatures in the range from 2500° to 1500° R will produce side force performance comparable to combustion chamber bleed. Such temperatures can be handled by current valve materials.

In figure 9, the side force performance of this system for a hydrogen/lox rocket is compared with that of two liquid injection systems employed on solid propellant rockets. The liquids employed are nitrogen tetroxide and freon. The TVC performance, which is presented in terms of the force and weight flow ratios, shows that the hydrogen-diluted system provides considerably greater side force efficiencies than either of the liquid systems. On the basis of the secondary weight flow required for a given side force ratio of approximately 2 percent warm gas, the hydrogen-diluted system is almost three times as efficient as the nitrogen tetroxide system and five times as efficient as the liquid Freon system. However, use of this figure is not meant to imply that the hydrogen-diluted system should be employed for solid propellant rockets; the choice of a secondary fluid, gas or liquid, depends on many factors. Figure 9 is intended to compare the level of performance obtained by the hydrogen-diluted system with the level of performance of other systems.

It could be legitimately concluded at this point in the discussion that here is a concept which is apparently in good shape and, therefore, it has been presented to a group which has the responsibility of making specific recommendations for additional technology programs. This might be misleading because hot-gas secondary injection currently has several areas of deficiency. Among them are the following:

1. Roll Control: An acceptable method for obtaining the third degree of freedom is yet to be demonstrated. However, this will be investigated shortly.

2. Combined Pitch and Yaw: Effort is now underway to evaluate what, if any, are the interaction effects brought about by a combined vectoring command.

3. Port Location Optimization: Although highest performance is associated with injection at the nozzle exit, system considerations may dictate that the location be otherwise. True, this is partially a design optimization problem, but certain aspects need further technological investigation.

4. Heat Transfer: As yet no adequate information is available on heat transfer with boundary layer separation. This problem should be investigated experimentally and analytically. In some of the tests performed in Florida, there was a rise which might or might not be detrimental.

5. Engine Vehicle Integration: Structural integration of a fixed engine, which is implied in use of this method since there is no requirement for gimbaling, needs to be investigated.

6. Control Integration for Secondary Injection: Allowable frequency and force change magnitudes in axial and lateral directions must be determined.

7. Structural Integration: Circumferential and longitudinal stiffness of the nozzle, plus thermal effects, must be studied.

8. Flow Integration: Secondary injection hardware weight trade-off, turbopump weights, and overall cycle efficiencies as mixture ratio changes.

Controls interactions, such as transient and steady state operation with and without secondary injection.

Minimum amount of secondary injection flow to create a shock, and continuous flow quantities required for thrust vector misalignment.

In summary, the technological status of hot-gas secondary injection at the present time is sufficiently advanced to be regarded as a proven, practical concept.

Some may consider this overoptimistic, but hot-gas secondary injection is already flying on some vehicles. There are, as discussed, certain technical areas which need additional work but the real need at this time is a good comparative systems analysis. The commonly held mistaken concept of building an engine and then fitting the vehicle to it is a logical one at this time, because it is based on much of NASA's past experience. Economic considerations as well as best engineering judgment may force design of the next spacecraft as a single entity, with the vehicle and power plant being given equal consideration as a unitized whole from the start.

Preliminary work indicates that such an effort pays off handsomely in the application of secondary injection to large launch vehicles. Elaboration on this point is beyond the scope of this discussion.

In relation to the system study concept, in the past NASA has been proceeding something like this: There is a vehicle in one place, the engine people are in another geographically remote area, and there is a government representative in still another area. The communication between contractors is through the government representative and that leaves the government representative swamped with integrating concepts, piecing together all types of information, and trying to make the best picture out of it. There is not enough manpower within most NASA installations to get this job done well.

Here is another way that the problem could be handled - adding to the present procedure a little box marked "Liasion" on the flow chart. This would get the contractors together so that when they come to NASA, they have all the bugs out of their concepts, have integrated ideas, and have come out with something near an optimum system. Then all the government representative must do is judge that system, not put all the pieces together.

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SCHEMATIC DIAGRAM OF HOT GAS TVC FOR HYDROGEN-LOX ROCKET

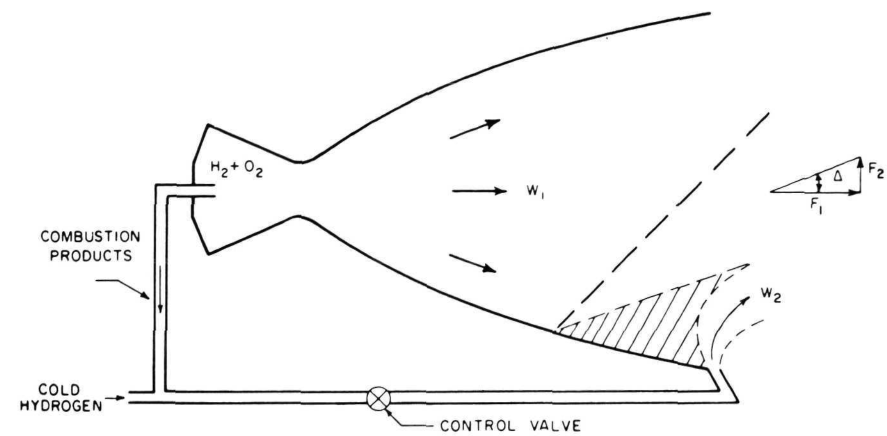
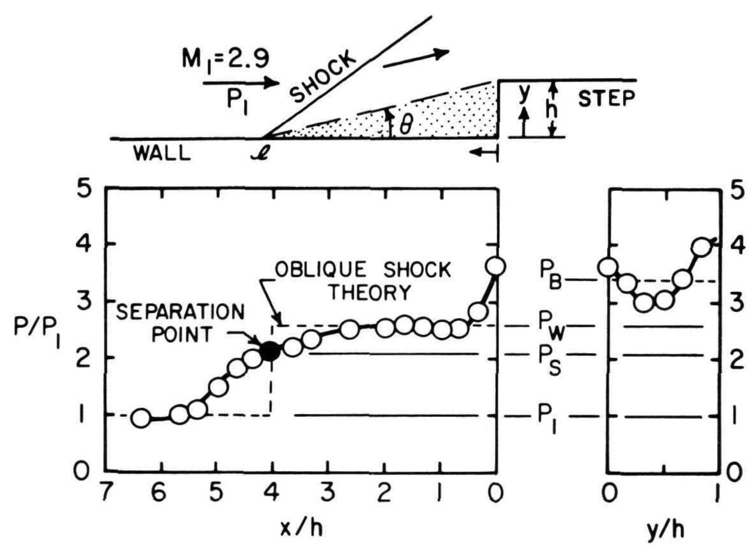


Figure 1



$$G = \frac{F_i}{F_B} = \text{SEPARATION PARAMETER}$$

Figure 2

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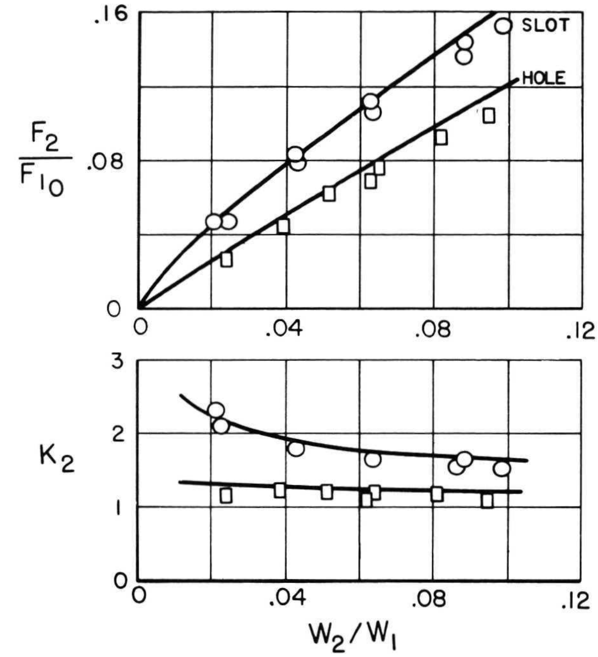

Figure 3

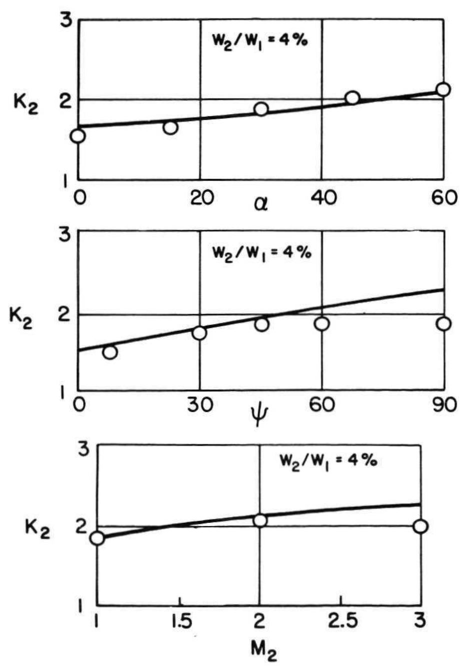

Figure 4

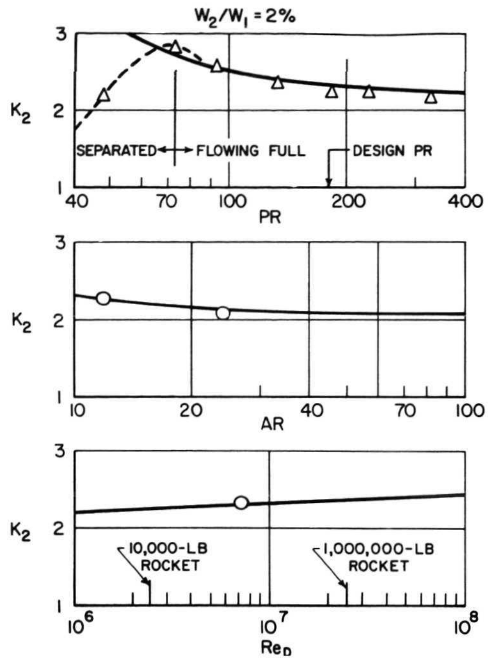

Figure 5

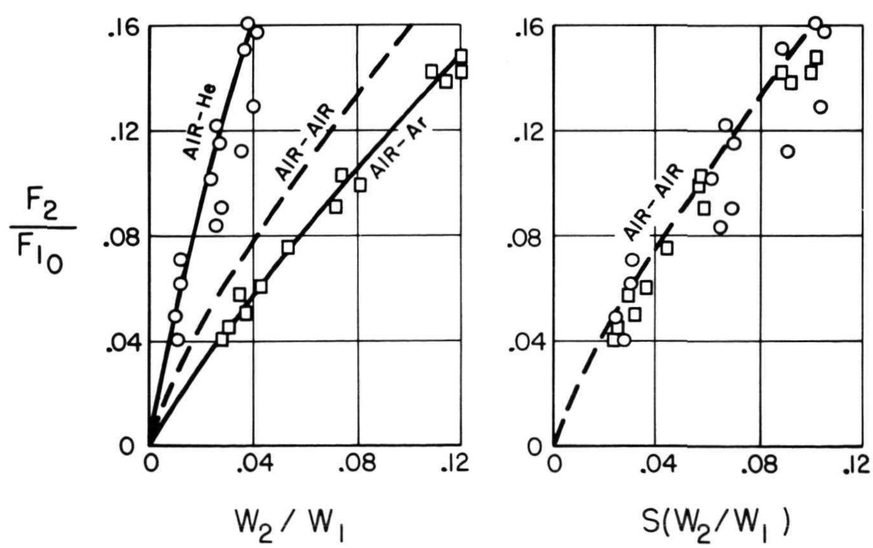

Figure 6

SONIC INJECTION THROUGH CIRCULAR ORIFICE NORMAL TO WALL

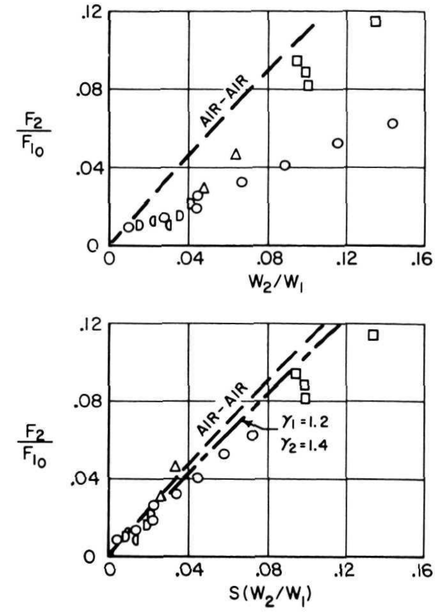

Figure 7

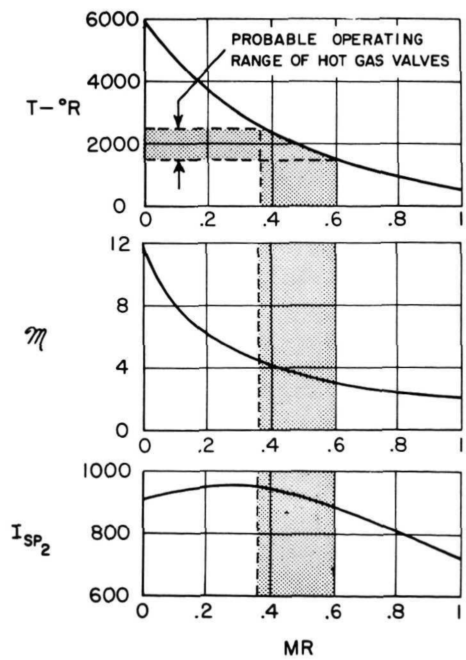

Figure 8

COMPARISON OF GAS AND LIQUID INJECTION TVC

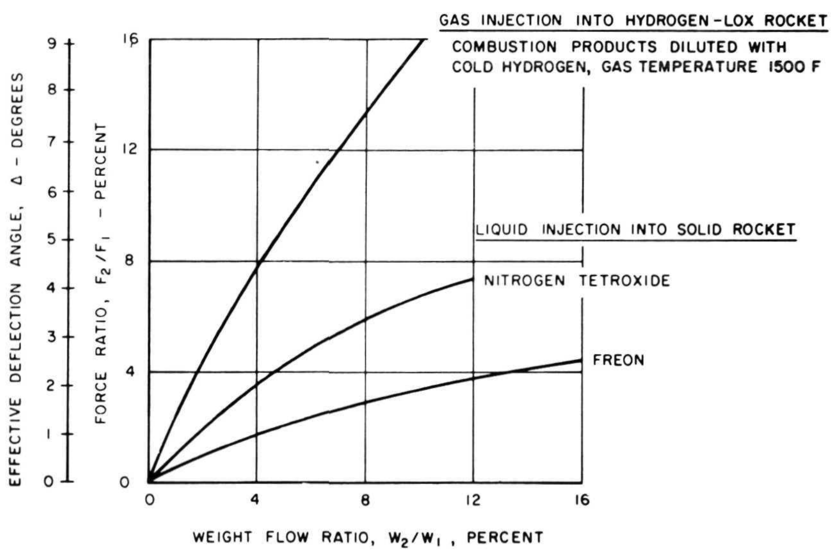

Figure 9

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33. REACTIVE FLUID INJECTION

By Foy McCullough

Naval Ordnance Test Station
China Lake, California

This discussion will start with descriptions of secondary injection or reactive fluid injection. When a fluid, either liquid or gas, is injected in the cone of a nozzle, a side force, as shown in figure 1, is generated.

The concept shown here was originally developed for use on solid propellant motors, in trying to avoid the high-temperature materials problem that was being encountered on the Subrock, Minuteman, and Polaris a few years back. In this slide is depicted the use of bleed gas to pressurize a tank containing a control fluid; Freon is currently the most widely accepted liquid injectant of the inert variety.

Systems of this type, such as on a second stage Polaris A-III, with gas from a separate gas generator, are now flying and have been flying since about January. Hence, this sort of system is now more or less operational.

Figure 2 illustrates the components of this side force. This is actually a data plot of nozzle wall pressure as a function of length. The injection port shown happens to be for air into air. At the injection port a spike is shown. The separation region, which is shown rather poorly here, actually extends quite a ways in front of the injection port, depending on the mass flow rate.

Immediately downstream of the injection port there will be a region of negative pressure which is a result of overflowing the nozzle. Then further downstream will be another pressure rise which contributes to the side force.

There are three major components that make up the side force. First, of course, is the thrust of the stream itself. Second is the high-pressure separated region forward of the injected stream. And the third is the high-pressure downstream area which in the case of a liquid injection is due to vaporization and, hopefully, some reaction.

The two pressure area terms are normally described simply as being a static pressure recovery region, or static pressure recovery of the main stream gases.

In inert liquid injection, this pressure recovery term comprises about 80 to 95 percent of the total side force that is generated.

With gas injection, this forward separated region is the major contributor to the side force.

Figure 3 presents an idea of the magnitude of the side force involved in liquid injection. This figure is a plot of the ratio of the side force to the main thrust as a function of the ratio of secondary flow to main flow, and, as mentioned, the slope of this curve is the performance ratio or amplification factor.

This is a plot of injection with Freon and is fairly typical of some Freon performance in this particular configuration. It is a single orifice injection.

The inert liquids normally give performance ratios on the order of about 0.2 to about 0.6. With the reactive fluids performance ratios are increased to the region of 1 or better. With hot gases it is possible to achieve a performance ratio of 2.

This plot is perhaps a little misleading. There is degradation in performance with any type of injection if the effect is too widely distributed over the circumference of the nozzle wall. As the secondary flow rate is increased, the pressure field spreads around the nozzle so that some of this force acts in a direction which does not increase the performance ratio.

Another effect is that a shock may be reflected on the opposite wall of the nozzle, and this can actually degrade the performance enough to bend the curve downward with a negative slope.

Figure 4 provides an explanation for some of the superior performance that is occasionally attributed to inert fluid injection. If a fixed diameter orifice is used and injection is up to some pressure, in this case 1500 psi, the results will follow a more or less linear plot similar to the one shown in figure 4.

With very small orifices and high injection pressure, the performance ratio can be quite high. If a variable area orifice is used with constant pressure injection, a different curve results, as illustrated. So, theoretically down in the region of about 1 percent flow ratio, performance ratios of 2 or more can be achieved.

This variable area injection scheme is the reason for so much activity in variable area valve development for both the Polaris and the Minuteman.

The Polaris performance follows a curve somewhat similar to one of these. The thrust vector angle required with the Polaris second stage is 8° per nozzle. This corresponds to a force ratio of approximately 12 percent and is achieved using only two of the nozzles on the Polaris.

With Freon 114-B-2, the injectant specific impulse at high injection flow rates is only around 80 pound-seconds per pound. At lower flow rates it is in the vicinity of 130 pound-seconds per pound.

The Polaris employs a dump feature; that is, any fluid which is not used at a certain point in flight is dumped overboard. Herein lies one of the advantages that makes it possible to use these low performance techniques. The vehicle actually carries a lower dead weight for the total distance than with a mechanical thrust vectoring system.

For fluid injection, the desirable fluid properties are: low specific heat, low boiling point, low heat of vaporization, low molecular weight of the injectant or its reaction products, high density for compact packaging, and high heat of reaction or exothermic decomposition.

Other things of importance would be high injection pressure. This is an aid primarily in penetration of the injectant into the main stream. The injection location is important. It must be pushed as far upstream to take advantage of the downstream pressure field as possible, but at the same time the reflected shock condition must be avoided. The pattern is known to be important: multiple orifice injection gives higher performance than single port injection. The work of Newton and Spade at JPL indicated what the pressure fields looks like in a real nozzle - at high injectant flow rates the performance falls off.

Figure 5 shows a concept as it might be applied to a liquid propellant motor, showing an artist's concept of tapping off from, say, one of the propellant feed lines in a regeneratively cooled motor, through a flow control valve, and injecting into the main nozzle.

Desirably, something like liquid oxygen would be injected into a fuel-rich stream in the hope of getting some additional reaction.

Figure 6 is a bipropellant concept which will be discussed in more detail later. Because this figure was prepared for the benefit of the solid propellant people, tankage is small. It depicts mixing the two propellants just at the wall of the nozzle, thereby allowing both mixing and reaction to occur in the main exit cone of the nozzle.

Figure 7 presents an alternative possibility for mixing with the use of a premix chamber or precombustion chamber essentially to prepare the propellants for reaction as they enter the main nozzle. However, if this procedure is carried too far a hot gas system would result.

Figure 8 shows the same general concept of bipropellant injection; it depicts the use of a shock-triggered reaction or a standing detonation wave. If an obstruction is located downstream of the point of injection such that a shock is created, it may be possible to establish a standing detonation wave.

At NOTS, standing-wave engines have been fired with the main combustion gases burned fuel-rich, by adding the excess oxidizer either at the throat or through a special type of injector, and with additional combustion occurring in a standing detonation wave. So this concept is possible and should offer some potential for increasing the performance of slowly reacting injectants.

Little work has been done with reactive fluids, and there are little or no kinetic data available for the combinations of constituents, and these conditions.

There has been some work done with monopropellants by Lockheed at Sunnyvale (on the Polaris program) where they injected 90 percent hydrogen peroxide into a liquid test motor. By comparison, the peroxide gave an injectant specific impulse of 116 pound-seconds per pound, compared to about 70 seconds for freon under the same conditions.

They also tried saturated solutions of sodium perchloride in water, but this turned out to be no good. Water is about the poorest injectant that has been tried. They also injected liquid oxygen as liquid, and achieved a measured injectant specific impulse of 144 seconds compared to 100 for freon.

In both cases, the liquid oxygen and the peroxide gave performance increases of approximately 45 percent over freon. Under proper conditions over a wide range of flow ratio freon can give specific impulses up to 140 pound-seconds per pound.

It is conceivable that the increase reported by Lockheed might be due to other things, especially in the case of liquid oxygen. But this is unlikely, and they probably were seeing some reaction.

In later tests it was reported that liquid oxygen had given measured injectant specific impulses of around 200 seconds.

Nitrogen tetroxide (N_2O_4) has been injected. One test conducted at Tullahoma on the X-248 motor gave inconclusive results. The performance of the N_2O_4 in this test was actually lower than had been achieved with Freon which is contrary to NOTS experience with N_2O_4 in other tests on other motors. No data are available as yet on some of the tests in which N_2O_4 has been used on the large solid boosters.

Figure 9 presents information on the performance of bipropellant injection; it shows the use of UDMH-RFNA being injected into the exhaust stream of a UDMH-RFNA test motor. Also illustrated is the performance of several other fluids; the solid portions of the curves give the actual data span.

The injection scheme used in this particular series of tests was the same as that shown in the figure where the mixing actually occurs at the nozzle wall as it flows into the main stream. It was a very poor injector because it was a single impinging pair, and under these conditions gave rather poor mixing.

However, the performance ratio - in the order of 0.55 - is higher than has been obtained with any of the other relatively inert fluids on this particular engine at these conditions. By using slightly recessed mixing points, the performance has been increased under the same conditions to about 0.65. At flow ratios around 2 percent, performance ratios of 1 have been achieved.

Most of the work, and certainly all of the work that we have done at NOTS on bipropellant injection, has been performed with nozzles of an expansion ratio of around 12 to 14. The expansion ratio of the nozzles used by Lockheed on Polaris was 14. The path that the injectant has to flow in the nozzle is quite short, on the order of 3 to 4 inches. Hence, only microseconds are available for mixing and reaction. (The kineticists shudder a bit at this.)

We have also conducted hot gas injection tests at NOTS in an attempt to see whether or not under these conditions we might get a reaction in our short path nozzle. Figure 10 shows one liquid test motor which has a water-cooled chamber at pressures up to about 1500 psi. The secondary chamber and the main chamber operate on UDMH-RFNA, and about 3500 pounds thrust is produced.

The main chamber has been operated at oxidizer-to-fuel ratios as low as 1.58, rather fuel-rich, and the secondary chamber, at the same time, has been operated at oxidizer-to-fuel ratios of 5.25. In the study, there was no difference in performance over this range of operation.

As has been stated, there is very little known about the reaction kinetics that apply. However, the kineticists seem to think that the simple combinations, or elemental combinations, such as liquid hydrogen and oxygen, or hydrogen and fluorine, will probably occur in the time span that we have in the larger nozzles. Certainly, if a standing detonation wave or shock is established, they should react.

Some work is being conducted in supersonic combustion. Antonio Ferri, on an Air Force contract, has found that there is no combustion problem with hydrogen and air (as related to supersonic combustion ramjets), but that mixing is the biggest problem. Penetration in the big boosters may be a problem in achieving sufficient mixing with the quantities of fluid that are required.

Figure 11 is a part of the actual film strip from a test of a Lucite nozzle.

Figure 12, from the same film strip, shows what occurs when Freon 12 is injected. The injected stream is actually almost half-way across the nozzle and folds back, and the excess then spills out the nozzle exit. The flow ratio here is around 15 percent.

It was found, however, that in calculating penetration, the assumption must be made that the injected material will follow a ballistic path. In other words, by assuming discrete particles an approximation of what happens in the nozzle can be obtained. Even with Freon, vaporization was incomplete in these tests.

The potential of reactive fluid injection is essentially wide open - not enough is known about it to make any knowledgeable guesses about the potential of reactive fluid injection.

It would appear that the performance in large expansion ratio nozzles can approach that of primary propellants and probably exceed it at the lower flow ranges.

To achieve thrust vector angles of 3° or 4° in large boosters, the performance of liquid injectants, even inert liquids, can be quite high.

There are some areas that need further study to answer questions about reactive fluid injection. These studies really should be made before anyone engages in a large program.

First, systems analysis studies are needed to determine the value or the trade-offs of fluid injection for the motors being considered, using a broad range of assumed performance.

Second, theoretical and experimental work is needed to determine the reaction kinetics of possible combinations at motor conditions, and, probably, shock tube experiments will prove useful as means of obtaining these rate constants

Third, experimental data are needed to determine the extent of penetration of the injectant stream into the main exhaust stream to determine the flow injection pattern, the mixing pattern, and the controlling parameters. Also, some knowledge of velocities behind the front of this disturbance would be useful.

Somewhere in the midst of all this, it might be wise to conduct a very limited number of tests with reactive fluids with one of the larger engines, something of the order of the F-1 or M-1, simply to generate data for the analytical types to try to answer why something happened or why it did not happen. Sometimes such testing activity seems to stimulate research.

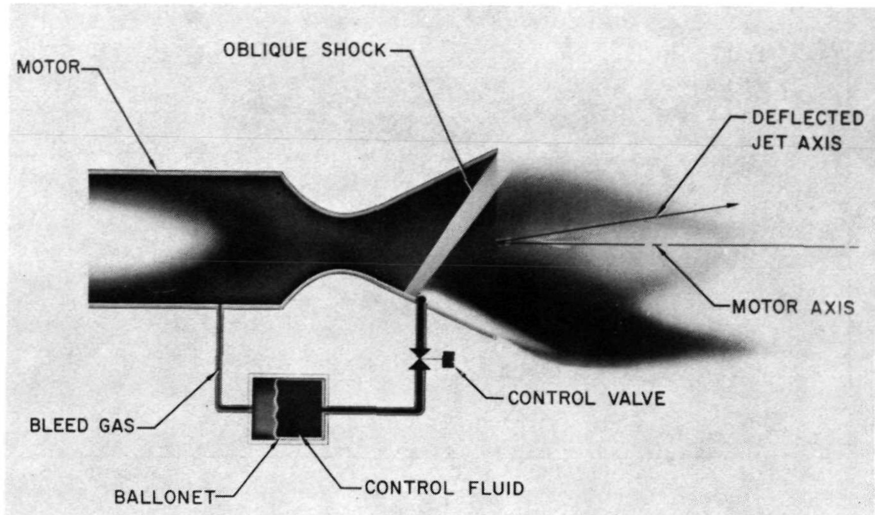

Figure 1.- Thrust vectoring by secondary injection.

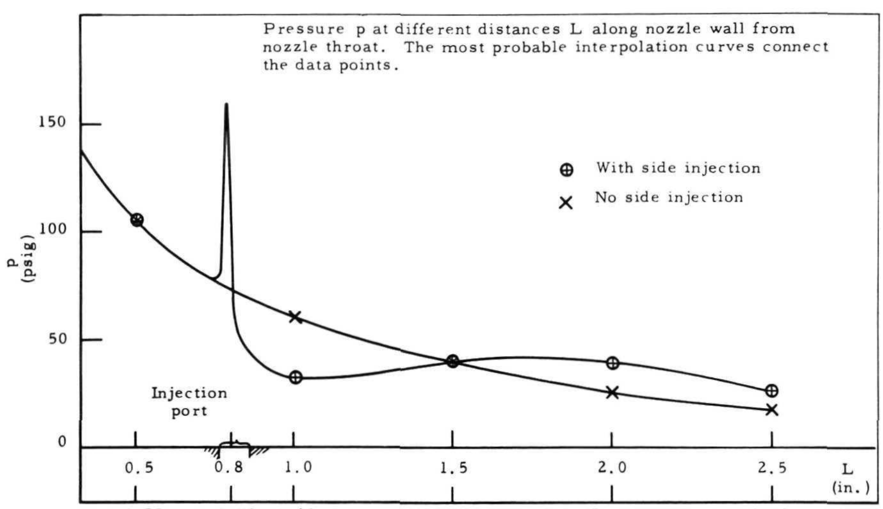

Figure 2

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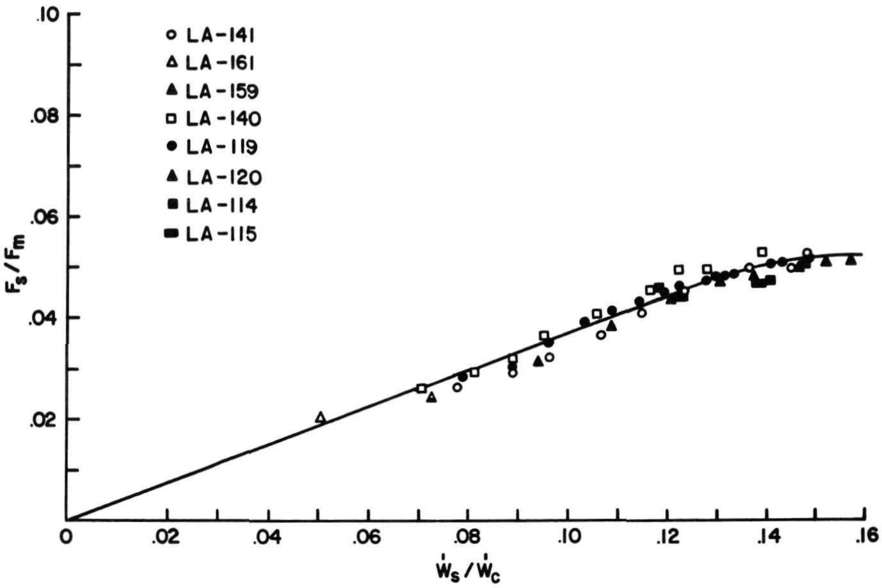

Figure 3.- Liquid propellant applied research motor-injectant freon 12 position 12.

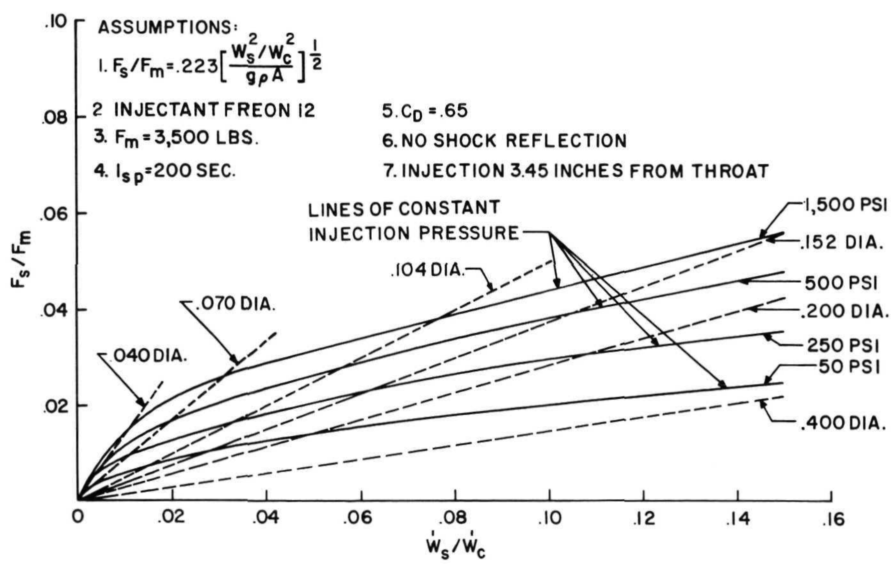

Figure 4.- Effect of orifice size on performance.

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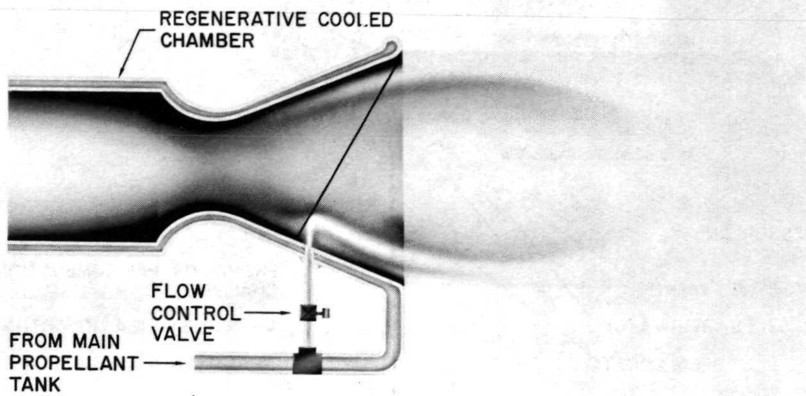

Figure 5.- Secondary injection thrust vector control (liquid propellant motor).

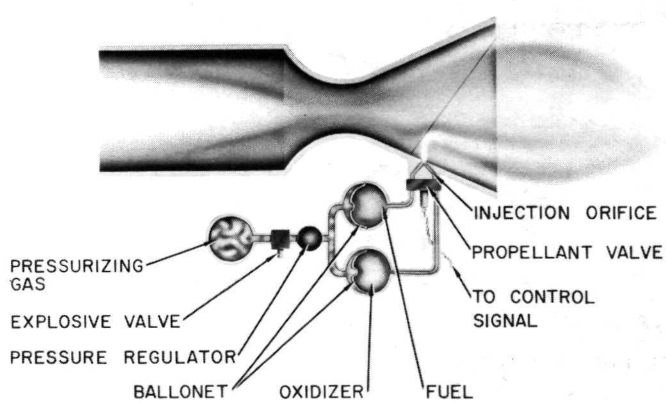

Figure 6.- Secondary injection thrust vector control with bipropellants (energy release in the nozzle).

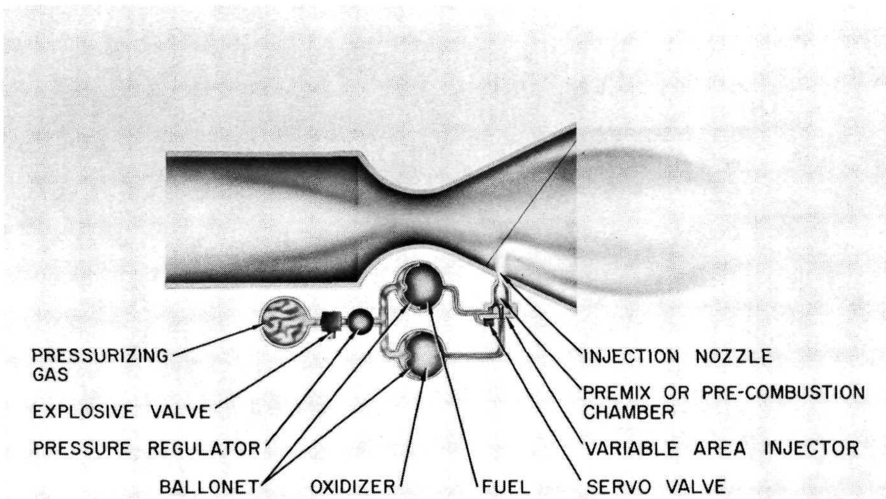

Figure 7.- Secondary injection thrust vector control with bipropellants (premix or precombustion).

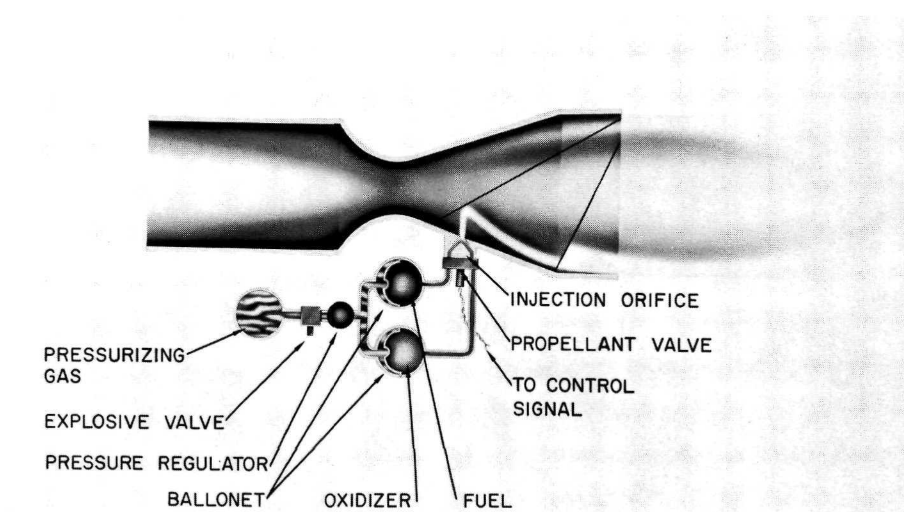

Figure 8.- Secondary injection thrust vector control with bipropellants (shock-triggered reaction).

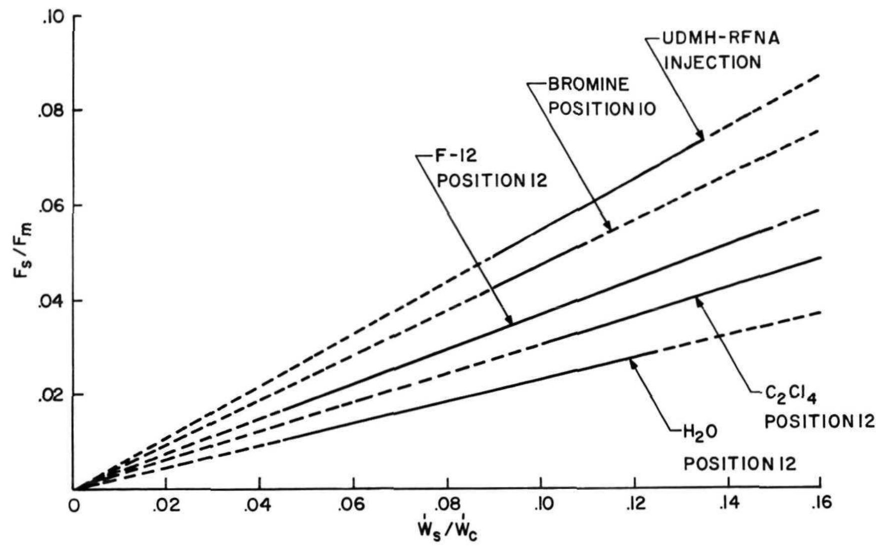

Figure 9.- Liquid propellant applied research motor performance summary, STR 355.

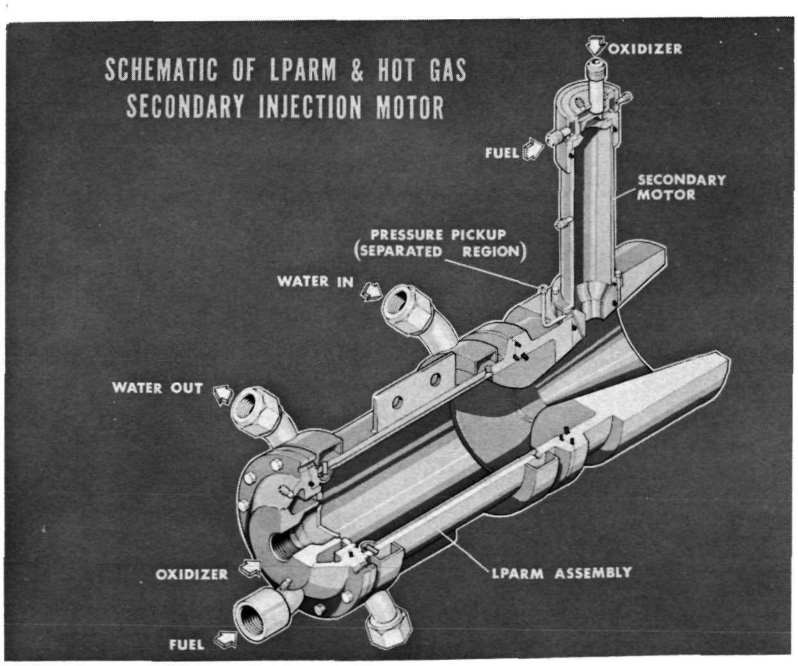

Figure 10

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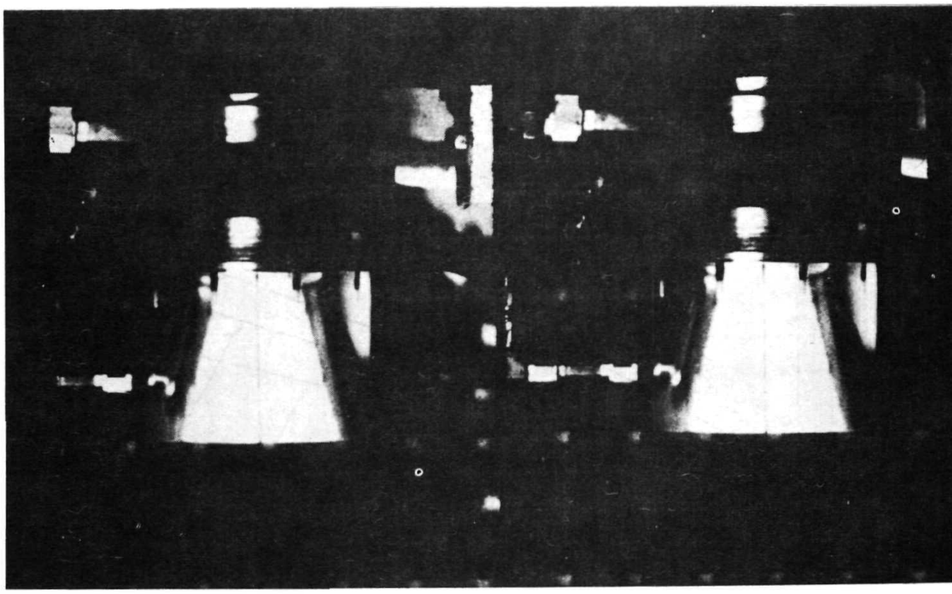

Figure 11

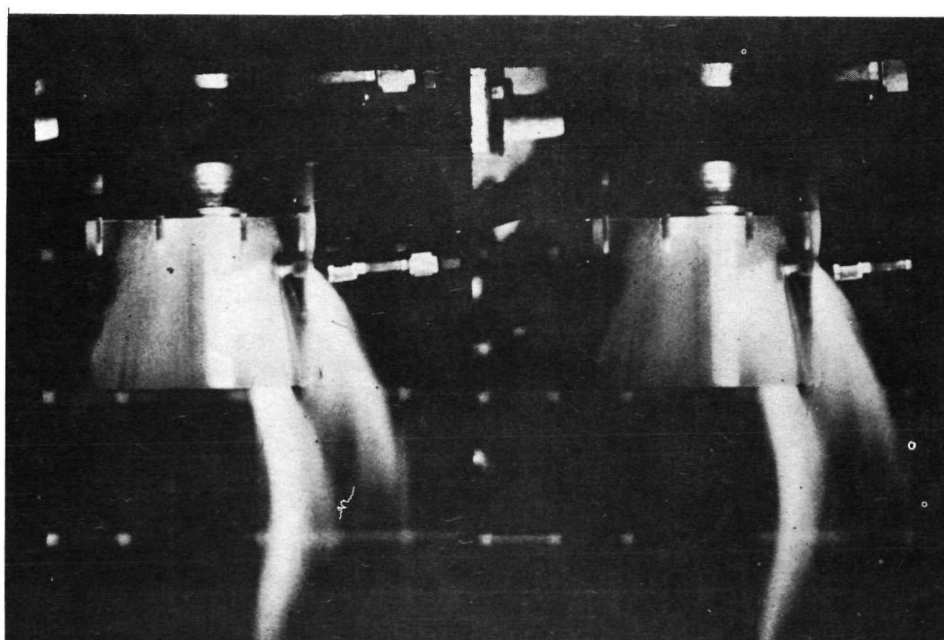

Figure 12

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34. MECHANICAL SYSTEMS AND DIFFERENTIAL THROTTLING

By Charles F. Zalabak

Lewis Research Center, NASA

Table I is a list of several mechanical systems for achieving thrust vector control. In the left column are the more desirable schemes; in the right column are what at the moment appear to be the less desirable schemes. The early liquid rockets have employed the jetavators, jet vanes, and jet tabs. In general, they are too heavy for the current concepts of the liquid engines, and the modern engines have been gimbaled.

Under the gimbaled engine system are listed two types: (1) the conventional gimbal in which the thrust loads are effectively taken out at a point and which uses a relatively small gimbal bearing and (2) the cam ring. A part of the thrust structure, a cylindrical section with nonparallel ends, is rotated about its axis to provide thrust vectoring. In its more advanced form the cam ring can provide the same kind of angular motion of the thrust axis that a conventional system produces.

In the right column, the gimbaled chamber concept, first of the less attractive systems, may have promise for the can-annular plug engine. For liquid rockets regenerative cooling will be required, particularly in the high engine pressure regime, and the difficulties of sealing take the canted and the gimbaled nozzle out of the picture.

The net result is that gimbaling is the attractive form of vectoring among the mechanical systems. It is a proven concept. It appears to be the lightest. And studies by Aerojet and Rocketdyne indicate that there are no conceptual difficulties in extending the capabilities of the gimbaling system to a thrust level at least as high as 8 million pounds. Therefore, the remainder of this discussion will be largely confined to the gimbaling system.

Figure 1 is a very rough schematic, intended only to indicate the way the thrust loads of the various engine types might be transferred to the vehicle. The bell nozzle has already been described. Thrust loads are most conveniently taken out in a thrust cone.

In the forced deflection nozzle, thrust loads are distributed over a large diameter, lightweight structure; therefore, the thrust structure is most conveniently represented by a large diameter cylinder.

The same is true with the plug concept.

There are still many problems in regard to the gimbal system proposed for the M-1, which has a conventional bell nozzle; table II lists some of the items to be considered. In brackets at the top are the M-1 gimbal specifications - $\pm 7.5^\circ$, angular rate of 15° per second, and a ringing frequency of something greater than 7 cycles per second.

The problems associated with the gimbal design are listed: the suction lines, as they determine the length of the thrust structure and the inter stage;

[redacted]

the volume compensation requirement as it influences the length; the natural frequency as determined by the engine packaging and the gimbal-bearing radius; the gimbal bearing design and whether it is to be ball and socket or roller bearing; and the choice of an actuator, either hydraulic or pneumatic since we are now working with the cryogenics.

Figure 2 is a schematic of suction lines. At the bottom is a schematic representation of the F-1 type suction duct which has been used as a design basis for the M-1 engine. The forward and aft U-joints take the angular deflection of the suction joint, and the sections on either side of center take the axial deformation. The section in the middle provides axial expansion when the sections on each side undergo axial compression, and vice versa.

The length evaluations are depicted in the sketch at the top of the figure. The correction "called out" is the angle corresponding to the misalignment factors that occur because of unknown positions of the pump inlet and the tank flange. When the total vectoring angle is large compared to the angular capability of the U-sections, the length is determined by the differential angle θ-δ and the amount of lateral displacement. With the differential first established the duct lengths required were extreme. It has since been determined that the angular capacity can be expected to be better than that initially used and acceptable duct lengths are the result.

When the angular capability becomes large enough, the determining length of the suction ducts is the volume compensator. From some studies in this area at Lewis and at Aerojet, it has been established that the need for volume compensation is relatively small; that it is quite reasonable to expect to have sufficient ullage pressure at the time that it is needed. The need is expected to occur only in the early stages of engine operation as the result of pressure surges associated with the maximum gimbal motion requirements.

As is indicated in figure 3, the natural frequency is proportional to the length of the lever arm (the normal distance between the gimbal center and the actuator attachment point), and to the square root of the ratio of the effective spring constant to the polar moment of inertia of the engine about its gimbal center.

The actuator can be designed with high enough stiffness that the effective spring constant is determined by the vehicle structure and the engine ring-and-tripod assembly at the actuator attachment points. With a specific value for the spring constant of the vehicle structure the engine ring-and-tripod assembly was varied to obtain the best combination of spring constant, weight, and actuator arm length.

Then the engine packaging was varied in order to lower the polar moment of inertia. The pumps were moved from station 1 to station 2. The inertia about the center of gravity was lowered. The gimbal center was relocated to position 2 by increasing the gimbal bearing radius. The net result was a reduction of the polar movement of inertia about the gimbal center. For an optimum arrangement of the actuator arm and ring-and-tripod assembly the natural frequency was about 5 cycles/second for the pumps-forward configuration and 7 cycles per second for the pumps-aft configuration.

Although the M-1 plan presently conceived does not include the development of actuators, some consideration was given to the area and some characteristics of various types of actuators are listed in table III. Gimbal actuator characteristics for the modern engines are presented in table IV. The actuators for the H-1, F-1, and RL-10 are hydraulic; those for the J-2 actuator represents those of a configuration in the early stages of development. We do not expect the weight of a pneumatic actuator for the M-1 to be extensively different. Certainly it would not be scaled up according to torque output and engine size. The primary difference would be the length of the stroke.

Figure 4 is a schematic of this actuator. It comprises a pneumatic motor driving a planetary transmission and ball screw. There would be no change in concept for M-1 application - only a resizing of the motor, primarily in speed, and a lengthening of the stroke.

In summary, the problem areas of the gimbaled design are shown in table V. The angularity limits that are imposed by the suction duct bellows result in high interstage and thrust structure weight due to excessive line length. The bellows design and material selection can be improved to give a higher degree of angular capability. Other concepts, for example, the arched suction line, in which the motion between pitch and yaw planes is separated, result in a larger angular capability.

A principal cause of natural frequency is the moment of inertia. This is a significant problem area with the bigger engines. As a result of lower natural frequency, there will be problems of coupling with the bending and slosh modes of the vehicle requiring greater control complexity. Solutions, as pointed out in the case of the M-1, are component arrangement and a larger gimbal radius. There are other concepts, one of which is the cam-ring gimbal, that attempt to solve the problem. The spring-mass system is characterizied by the inertia of a part of the thrust structure, and torsional vibration modes are excited.

Gimbal bearing friction plus the structural deflection of the engine can cause the control dead band and resulting control instability. Potential solutions are the selection of suitable materials and lubricant and the bearing design.

The effort involved in the solution of these problems appears to have been incidental to each engine design. It is possible that advanced technology effort in some of these areas could be very helpful.

The thrust vector can be controlled by throttling. Two modes of throttling are available and impose similar performance penalties regardless of engine type. One mode is that of plus and minus control for engines on each side of the vehicle axis, in which case the engines will be subdesign capability during the nonvectoring time periods. If engine thrust is decreased to the only freedom allowed, the total vehicular thrust oscillates with vectoring requirement.

With the conventional-bell engine vector, control by throttling is limited to multi-engine configurations. The control capability can be determined from the thrust axis and vehicle geometry and the degree of throttling. In the case of the forced-deflection nozzles with multichambers, the work to date indicates only small corrections are available. The same is true for can-annular plug configurations. The capability can be increased by increasing the control complexity. For instance, control over half the periphery improves the capability, but with pitch-yaw control require a significant complexity problem is imposed on the controls. Experimental results indicate a decrease in vectoring capability for increasing nozzle-pressure ratios.

To define the area of applicability of thrust-vector control by throttling, experimental studies should be made on the forced-deflection and can-annular plug configurations to establish the performance at high pressure ratios. The desirability of a vehicle with this capability should be evaluated. Also, there is a need to establish the feasibility of vehicle operation with large axial thrust variations that result from throttling.

TABLE I

MECHANICAL SYSTEMS FOR THRUST-VECTOR CONTROL

GIMBALED ENGINE
- CONVENTIONAL
- CAM-RING
- HYDRAULIC-SLIPPER

JETAVATORS

JET VANES

JET TABS

GIMBALED CHAMBER

GIMBALED NOZZLE

CANTED NOZZLE

ASYMMETRIC NOZZLE EXTENSION

TABLE II

M-1 GIMBAL DESIGN

$$\left[\pm 7\frac{1}{2}^{\circ},\ 15^{\circ}/\mathrm{SEC},\ f_n \geq 7\ \mathrm{CPS.}\right]$$

SUCTION LINES
- LENGTH
- VOLUME COMPENSATION

NATURAL FREQUENCY
- ENGINE PACKAGING
- GIMBAL-BEARING RADIUS

GIMBAL BEARING
- BALL AND SOCKET
- ROLLER

ACTUATOR
- HYDRAULIC
- PNEUMATIC

[redacted]

TABLE III

ACTUATOR COMPARISON

HYDRAULIC	WELL DEVELOPED, PROVEN RELIABILITY, CAPABLE OF BEING SCALED TO SIZES REQUIRED, HAS REQUIRED STIFFNESS, REQUIRES TEMPERATURE CONDITIONING AND AN AUXILIARY PUMP.
PNEUMATIC	OBTAINABLE WITH APPROPRIATE POWER CAPABILITY, DOES NOT NEED TEMPERATURE CONDITIONING OR AUXILIARY POWER SOURCE, STIFFNESS IS A PROBLEM IN SOME CONCEPTS, IS NOT WELL DEVELOPED.
ELECTROMECHANICAL	MOST APPROPRIATE FOR SMALL SYSTEMS, MANY RESTARTS, FOR APPLICATIONS WITH READY SOURCE OF ELECTRICAL POWER.

TABLE IV

ENGINE-GIMBAL-ACTUATOR CHARACTERISTICS

	H-1	F-1	RL-10	J-2	M-1
GIMBAL RANGE, DEG	±7.5	±6	±3	±7.5	±7.5
OUTPUT TORQUE, FT-LB	18,300	380,000	2,770	34,800	557,280
OUTPUT VELOCITY, DEG/SEC	19	10	8	9	15
ACTUATOR WEIGHT, LB	22.5	300	8.9	75	150
ENGINE THRUST	190K	1.5M	15K	200K	1.5M

[redacted]

TABLE V

PROBLEM AREAS IN GIMBAL SYSTEMS

	PRINCIPAL SOURCE	CONSEQUENCES	SOLUTIONS	EFFORT
1. ANGULARITY LIMITS	SUCTION-DUCT BELLOWS	HIGH INTER-STAGE AND THRUST STRUCTURE WEIGHT DUE TO EXCESSIVE LINE LENGTH	BELLOW DESIGN, MATERIAL SELECTION, IMPROVED CONCEPTS (E.G., ARCHED-LINE CONFIGURATION)	
2. NATURAL FREQUENCY	MOMENT OF INERTIA	COUPLING WITH BENDING AND SLOSH MODES, CONTROL COMPLEXITY	COMPONENT ARRANGEMENT AND GIMBAL RADIUS TO REDUCE MOMENT OF INERTIA, IMPROVED CONCEPTS (E.G., CAM-RING GIMBAL)	INCIDENTAL TO EACH ENGINE DEVELOPMENT
3. CONTROL DEAD BAND	BEARING FRICTION PLUS STRUCTURAL DEFLECTION	INSTABILITY	MATERIALS AND LUBRICANT SELECTION, BEARING DESIGN	

THRUST-LOAD DISTRIBUTION

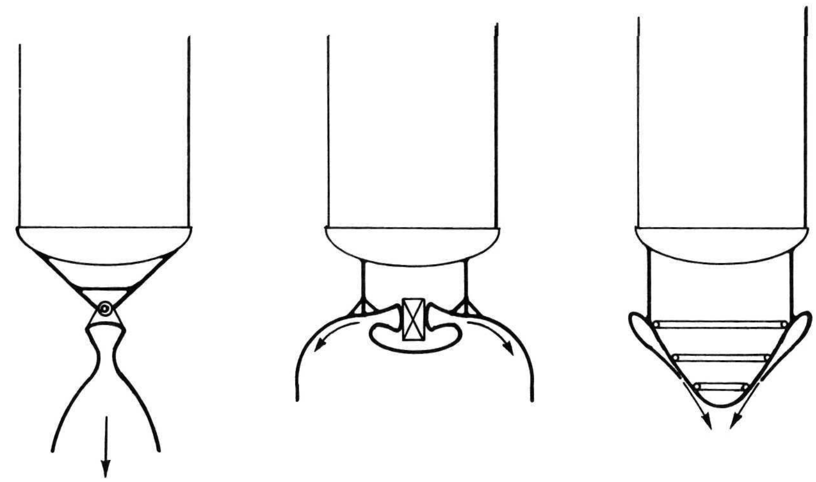

Figure 1

SUCTION LINE-LENGTH DETERMINATION

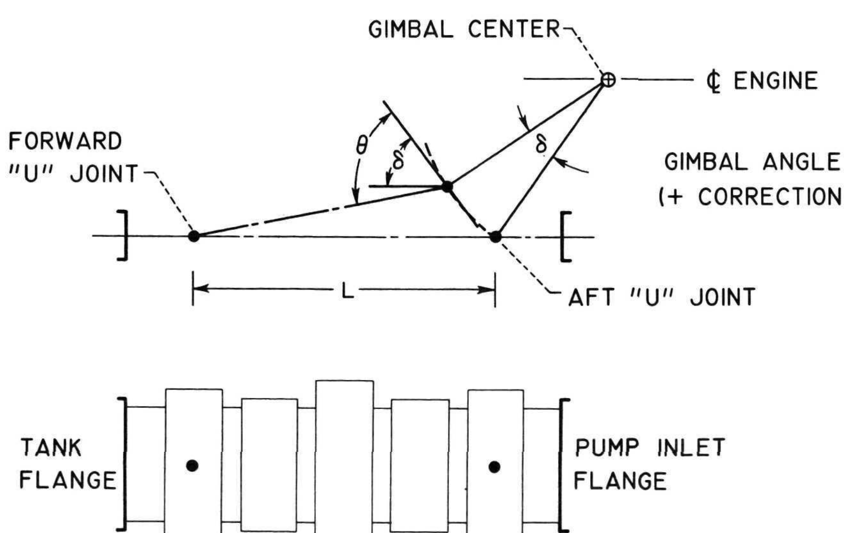

Figure 2

NATURAL FREQUENCY REQUIREMENT

$$f_n \propto \text{LEVER ARM} \sqrt{\frac{\text{SYSTEM SPRING CONSTANT}}{\text{POLAR MOMENT OF INERTIA}}}$$

($f_n \geq 7$ CPS FOR M-1)

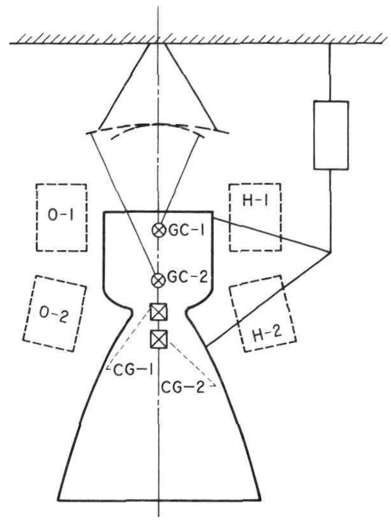

Figure 3

PNEUMATIC GIMBAL ACTUATOR

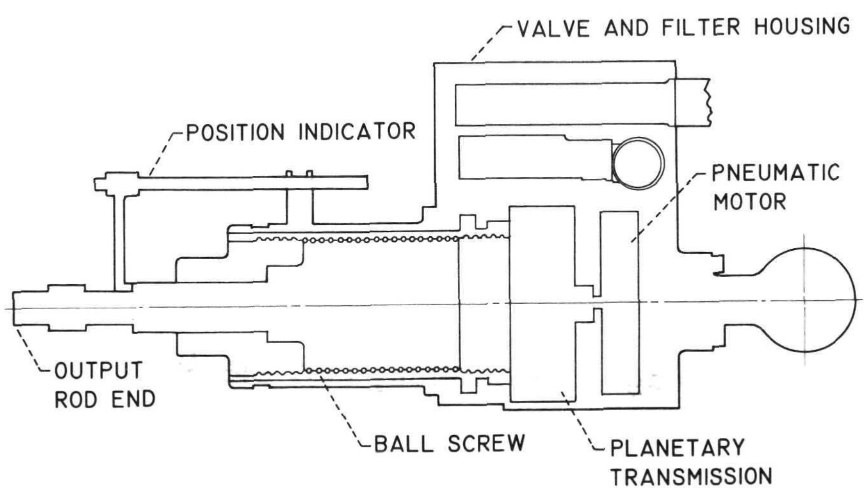

Figure 4

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35. SUMMARY DISCUSSION

MR. COMPITELLO: An engineer, working for the Navy, is working with what chemists call perhydrides - I just call them the heavy hydrides, such as FE-86. Do you have any plans of using any of these as a propellant reactive fluid, Mr. McCullough?

MR. McCULLOUGH: At the present time we are essentially - at least I think we are - phasing out of liquid injection. We are now working on hot gas injection as applied to solid propellant motors. So we have no plans in this particular area.

MR. HAWK: Just one comment, Mr. Beheim, not a question, really in that we have talked about the advanced nozzles today, and all the work on TVC that has been discussed has been done with the bell nozzle configuration. We have done a small amount of work with the annular radially outward flow nozzle, and the correlation does not appear to be very good, primarily due to the fact that at low pressure ratios, when we have a narrow annular supersonic jet, we are penetrating the jet with the secondary flow, and we are limited to extremely small secondary flow rates. So, at sea level conditions and, say, under 10,000 feet in our particular application at high chamber pressure, we could not get very effective K factors with our secondary injection techniques in the advanced nozzle.

One last point: Our slot performance at altitude seemed to be the best, but when you consider the practical aspects of building a slot into a rocket nozzle, it doesn't look attractive.

Dr. Rao, consulting on our program, came up with a relationship between spacing and sizing in an attempt to simulate a slot. The results were that we did, in fact, very closely simulate a slot, in some cases getting even better performance with round holes, which would be easier to work into a practical configuration.

MR. ZALABAK: One comment with regard to the selection of the vectoring system for the M-1. There will be a report out in the near future by Aerojet concerning a system study for secondary injection.

MAJOR RESEARCH AND TECHNOLOGY PROBLEMS

Chairman: John L. Sloop
Director, Propulsion and Power Generation
Office of Advanced Research and Technology
NASA

36. SUMMARY OF PREVIOUS DISCUSSIONS

AND FINAL PROBLEM REVIEW

MR. SLOOP: At the beginning of the meeting I said I would call on the session chairmen for any comments they cared to make, including a summary of their particular sessions and any other things that they feel called on to make about the whole session. I would like to do this in reverse order, since we went from general to detail.

Since Milton Beheim was just up here, I would like to ask him if he has any additional comments to make, in general or specifically, about his part or anything else he has heard.

MR. BEHEIM: On the matter of altitude compensation, conclusions regarding the importance of external stream effects are certainly influenced by the choice of chamber pressure. As chamber pressures increase, undoubtedly the problem of altitude compensation will become less difficult. However, the matter of achieving the additional base bleed effectiveness has not been achieved and requires more work. On the matter of thrust-vector control, I think the doors are still open to all systems. It does appear that the reactive fluid approach has not been studied sufficiently, but might prove interesting for very large rocket engines.

MR. GINSBURG: I have no additional comments. I think the last slide this morning covers in general recommended problem areas that would warrant serious consideration for additional or extension of technology programs.

MR. MORRELL: I want to re-emphasize that there really isn't a detailed direction you can give to the advanced technology program. The engines will just get bigger, and support should be given to anything that leads to better understanding of larger systems.

I would like to make a proposal, however. I think that a group of people representing those Centers and NASA Headquarters that have an interest in large engines ought to go further than the Nova studies, and actually come up with a real selection for a large engine and vehicle combination, if we had to do the job right now. This development program should be continued on paper, bringing in the new technology as it comes along and revising it from time to time, so that NASA internally has an objective toward which we are moving, so that when the time comes that we must get financing we will have something ready to go that is feasible at that time.

Maybe we can do it, maybe we can't. I don't know. We certainly won't do it in auditoriums. We will have to get some people together who can speak for their Centers, at least technically, and when they get through with that maybe they can speak for management and get some approval or disapproval of the idea. I think we have to have something like this going, otherwise it will always be a flop.

MR. BARTZ: I agree with Gerry Morrell on the point of establishing some kind of direction for a future development direction because what one does about the cooling is quite dependent on how this beast is going to look.

I think to cover all bets will be too costly and too time-consuming for the resources that we have until you come to such decisions as will it be one-stage to orbit, and, therefore, are we talking hydrogen/oxygen, are we really talking 3,000 psi, and so forth. Until matters like this are narrowed down, we are just going to be taking stabs here and there.

In order to get a more concerted effort toward getting a .complete set of answers that will do the job at a given time, I think we will have to take the steps Gerry Morrell just outlined.

In regard to the cooling problem, in general, I have long been opposed to studying rocket motor heat transfer in rocket motors. I think I will have to change my attitude when it comes to high-pressure engines because it seems to me the only real place, the only real hope for very high chamber pressures now - I will define that as something over 2,000 psi. The only real hope for cooling such engines, in my opinion, lies in the mass transfer methods, and when we talk about mass transfer methods, I think we have to study these things in an actual combustion system because we can't separate the combustion and heat-transfer problems. They are one and the same, intimately locked together.

Therefore, I think that there needs to be more studies of high-chamber-pressure combustion and high-chamber-pressure film and transpiration cooling, if we are really going to be serious about this direction.

As I see it now, we have only one such program in the country, that at Pratt & Whitney, and it wasn't clear to me that there was an assurance of a long-range continuous funding on that program. If it should go down the drain, I am afraid that the last or the only program that I know is specifically directed at getting that kind of information.

I might make a brief comment in response to something Mr. Thompson brought up about hot gas thrust vectoring. There could be a heat-transfer problem associated with that, I think, judging by our experience in diffusers. The problem is this: When a supersonic boundary layer separates, and then, for some reason, re-attaches, there is quite a heat transfer because of the very thin boundary layer where it re-attaches.

I could possibly envision a re-attachment of a separated boundary layer at the edges of the separated flow that occurs in hot gas thrust vectoring. When that occurs I could, therefore, envision a very high local heat transfer at the edge and, therefore, possibly, such a high heat transfer that it would result in burn-outs.

MR. WEIDNER: From what we have heard today, and yesterday, I believe that there are too many unrealistic concepts floating around. We must somehow bring some order into this matter.

Most people in the field believe that today's problems are not funded big enough. So I don't expect the step of advancement that we are always talking about to be as big as some people hope it to be. I think somehow, once the chips are down, we will have to be satisfied with something much smaller than we may hope to be able to do.

I think we cannot go on charging in our research in all directions, but we have to somehow soon restrict ourselves, and wisely. I think what is going on in Congress now is an indication that money might not be too plentiful in these areas in the coming years. And that is more reason to concentrate the dollars we have in this area which looks promising, and close to a realistic line. Also, we should somehow make a wise decision, a restrictive decision, and, dropping some things, say this is where we want to work. This might be a gamble, since this decision has to be made on not enough knowledge, as we all heard. But the gamble is a much smaller one, I think, if we do this now, than if we do it a couple of years from now at a time when we may have to move on a much bigger scale.

We can't afford, I might say, with this preliminary decision to be wrong. Though it will not be a killing one, it will be nasty. Therefore, I think I am pleading for some sort of pulling together to make such a decision.

This pulling together, I think, must start at the Washington level. I think people like John Sloop should be the starters. This would be a recommendation on my part. I think the Centers should be willing to help do the job. I think it is a combined job between the organizations here.

Maybe we should establish some group of people who are at least at decisive enough levels that they can influence things and are willing to stick their necks out and say, "This is what we should drop, and this is what we should do."

MR. SLOOP: Thank you, Hermann Weidner.

The data used by vehicle designers and studies must come from engine people. I think it is our duty to be realistic in furnishing these data to the user of vehicle designers. Frank Williams has headed this part of our discussion. I think he has taken quite a bit of comment from us, and it is his chance now for rebuttal.

MR. WILLIAMS: This two-day session has been quite informative to me in general. I have learned, as I think most of you have, that there is not universal agreement on the question of what is Nova. To go one step further, it is not clear, even if we know what basic configuration we would use in Nova, what propulsion system would be used.

To take this additional step, even if we decided what the vehicle looked like, what propulsion system we would have in terms of thrust, I_{sp}, if you will, we still don't know what kind of engine we want, whether it would be ED, plug, or bell, or what-have-you.

In fact, as one of the Aerojet engine people, they are still looking at the low P_c pressure-fed.

Of course, we concentrated on liquid systems. We have not heard anything from the solid propellants people, and they at least think they are still in the running and in the minds of many people they are.

Even if we picked the vehicle, picked the propulsion system, picked the type of engine, it is quite evident to me that there is no agreement on what P_c level we should have, what kind of cycle we should have, even the number of pumps that we might need to generate the 20-odd million pounds of thrust. And even if we picked some of these things, just froze them somewhat arbitrarily, there is no agreement. At least, that is my impression, after listening to the sessions, on such things as the various aspects of the pump design, the type of turbines you would use, and the type boost pumps.

So, in general, I think we have reassured ourselves that there is a lot of work to do in practically every area.

One thing that concerns me, and it was voiced by several people - Mr. Weidner just before me - that we can go along working for the next year, all of us assemble back here next July, hear what has been going on, but still not have the answer to the question: What propulsion system or engine should we use for Nova, or what is Nova?

Granted, we might be a lot wiser; we may have a lot more competence level in many areas of technology, but I still don't think that we will be able to answer the basic question of what is Nova and its propulsion system.

To a degree I think that we are not really utilizing the vehicle systems work. I will be the first to admit that we do not have the detailed answers to all the questions and the various criteria that we have under consideration. But I do think that if we, NASA, put ourselves into the study, taking a greater part and participating to a larger extent, can provide much higher competence level in the trends and the data that are coming out of these. We can use these as a backdrop to project our programs in advanced technology, particularly in the propulsion field, because in the overall vehicle system propulsion is our critical area. It is one where we have the greatest number of unknowns, even though much is unknown about recovery, the reuse of vehicles, the engines themselves, and longevity. But propulsion, per se, is our number one problem, notwithstanding such problems as money and program approval.

I also think that if all of us take a more active part - and I urge this - in the vehicle study activity, we would have even higher competence level in some of the trends and conclusions that I think we can draw. The trends that we recognize now, and the conclusions that we can draw at the present time, and that we would be able to have a little better feel for the very mushy areas, vague areas, throughout the system and, particularly, in the vehicle area.

You may have noticed one thing in our vehicle studies, costs or dollars - we boil it down to cost effectiveness - is a very important parameter.

If we plot payload, and pick one over a hundred - if we plot payload versus P_c - you all know what the curve looks like, I won't bother putting numbers. It begins to bend over around 3,000 psi. You may have 2,000, maybe 3,000, somewhere along in here.

We also plot versus P_c, the cost, development cost of an engine system, or the manufacturing cost after it is developed, or, if you will, the total program cost. We get a curve that may look something like this. If we trade these things off versus P_c and call this cost effectiveness, if you will, the cost of the total system, developed and operating over a ten-year period and getting payload to orbit or people to Mars or what-have-you, we get a curve that maybe looks like this, if we take our nominal values.

If we take the dispersion, it may look like this. At best it may look like this. These are the kinds of things we are going through. I feel that you people, along with the engine people, because GD/A and Martin have people that are up to date and trying to keep current in the state-of-the-art and propulsion systems and various elements thereof, but they themselves do not have the ability in my mind to really define such curves as this, this, and numerous others, which really boil down to things like this.

And these are the types of criteria, right, wrong, or indifferent, that we are using at least as a measure of the desirability of one system versus the other. We take into consideration manufacturing, development risk, launch facility requirements, all the way across the board. But here is where I feel we need definite support from you, and we can enhance our competence level in the final trends and conclusions that come out of the study and we can give better results in this area.

Another point that I would like to make in looking back into the history of airplanes or launch vehicles, guided missiles, et cetera, and trying to project this into the future, I believe that when the time comes to start major elements of Nova, the engine, or pumps - or maybe even the Nova program itself - that we are not going to have absolute proof that what we have selected is the optimum, or the best. In fact, in some cases even the feasibility of some of the items that we may have to choose may not be proven.

With that in mind, maybe we should establish a new criterion that would help us in defining where we go from here. Maybe we should optimize a system to the minimum distance to which the NASA Administrator must stick his neck out to get the program approved, or in approving the program itself. In doing so we have a limited amount of money, we have a large area of technology, particularly in the propulsion area.

We have a whole area of unknowns in the propulsion area. We can take the shotgun approach and spend a little money here, there, and everywhere, and arrive back here next year with a lot of data, feeling better about certain areas, but still not be able to get from the center of the problem out to a solution which I think we could if, as Mr. Weidner said, we concentrated our efforts.

This takes risk, but in optimizing the total system I think maybe the risks are better in providing the NASA Administrator the kind of material he needs to make a decision to approve certain things if we take such approach.

I don't have the ability and doubt that anyone in the audience would like to take the chance to say where we should spend our money in the coming year in the propulsion area. But it is within the capability, as voiced by several people, that if we set our minds to it and not necessarily form a committee - in fact, I think that may be the worst way to go - but get a working group of people together and assess the problems, the unknowns, the advantages and disadvantages, of various elements, utilizing what we know, some of our technology work and the vehicle system work, that we could really define this problem.

This is just an example of what I mean. One of the things that has come out of our study is that we want a two-stage Nova. Maybe a two-stage Nova is what we want. In the GD/A presentation, they concluded that the first stage should have lox/RP, should have an ED nozzle, about 2,000 psi with an expansion ratio such that it is fully expanded at sea level, and about 6 million pounds of thrust.

The second stage should use hydrogen and oxygen, again an ED nozzle, of a little bit higher combustion chamber pressure.

Why not go out on competitive bids, on a predevelopment or feasibility development for these two engines, and select a company for each of them, or maybe the same company for both of them, and spend a great percentage of this coming year's advance technology money to really solve the problem of an engine system, rather than, if you will - and I hope you will pardon me if I offend you - take the shotgun approach and do a little bit here, a little bit there, and a little somewhere else.

Maybe I am all wet in this approach, but I think this would direct the effort along the line such that we would be much farther along in the nearer future to answering the question of what is Nova. It may not be the very optimum solution because, I don't know, some people have the philosophy - and I think most of the time I fall into this category - the better is the enemy of the good. I think we can select a system that when we do get it developed, it will be a good system, a workable system. And we could have it ready in sufficient time to have the competence level such that the Administrator, when it comes time to approve this, or help sell it, he will have all the ammunition he needs to do so.

That about concludes it. I will pull my neck in now. I would like to invite each of you to Marshall Space Flight Center the latter part of August. We will have a major Nova systems study review. Right now it is tentatively set for the 20th, 21st, and 22nd. That will be a two-, possibly three-day session. Each of you will be informed via the Center representatives that are members of the Nova management team as to the precise date, time, et cetera. I would encourage each of you to come. We need your support. I think we can help you.

MR. HAWK: The only comment I had - this is strictly a personal observation as I don't have any Air Force sanction on this - it is my concern that we have met here the past two days, and we have discussed major type problems of combustion instability, advanced nozzles, pumping, et cetera. Needless to say, these are major problems and need immediate and concentrated attention for large systems. But one thing that has in my opinion always plagued us, so to speak, are the nickel and dime parts, the subsystem paraphenalia that goes into putting an engine together - valves, bellows, lines. When we start talking about a 20-million-pound engine and some of the peculiar techniques of putting them together that have been discussed over the past few days, that we cannot afford to ignore or pass by these areas which right now seem of lesser importance but will loom up quite large as we get closer and closer to actual advance technology or experimental development of such an engine.

MR. SLOOP: There are two headquarters offices quite interested in advanced engines. The Office of Advanced Research and Technology and the user office, in this case of the large liquid rocket engines, is the Office of Manned Space Flight.

Mr. Tischler and I have worked together planning this meeting. I want to call on him now if he has a few words of wisdom.

MR. TISCHLER: Anything I say may be too much, and I don't want to take too much time.

I think there has been a flame smoldering for some time - let's call it a torch - in this Nova concept. It was pretty well squelched a year ago by the fact that there wasn't money for it, and that condition seems to prevail even now. I think it can be brought back to a pretty bright flame if we can simply carry it forward fast enough. I believe that we have been handed this torch, John, you and I, and I think we have a job to do in bringing together an agressive effort in the technology area, and obviously within the funds that will be available, with the idea of developing the concept into a clear picture that can be defined and presented so that it becomes salable to our Administrator and to the nation as a whole at the earliest possible date.

I believe sincerely that we must do this on a fairly aggressive basis because, in the long run, whether we continue to operate in space on a large scale may depend strongly on how much we accomplish in defining our objective.

I think timewise there is a two- or possibly even four-year period in which this must be identified. I do believe personally that this is adequate to bring us to the point where we could initiate serious development.

I must take issue with Frank Williams a little in that I don't think this is the time to start developing a model engine, although that may be a logical step along the way. I think there are too many problems that haven't been answered. One of the principal ones is the cycle that should be adopted. The system concepts - turbopumpwise, particularly and combustion chamber, and nozzlewise secondarily - must be a part of that cycle concept. So this is one of the first areas perhaps that needs to be attacked.

I do agree with his argument in the sense that a hypothetical engine must be defined at an early date to tell us which technology to pursue to the greatest degree.

If we find blind alleys here we will have to change this hypothetical engine to accommodate those blind alleys and develop new areas. But certainly this is another part of the task.

MR. SLOOP: I hope that we at Headquarters can provide the catalytic action, but the steam must come from the Centers, because they have the people and the detailed know-how.

MR. WEIDNER: And money.

MR. SLOOP: And money. We can help with the money.

As many of you know, Henry Burlage has been in research and technology involving large engines since 1960, so, I wonder if he wants to say anything at this time, about any of his impressions.

MR. BURLAGE: I don't think so. What more can we say?

MR. MELLOW: I think it was very well stated for the time being.

MR. SLOOP: I would like to thank the session chairmen and the speakers for a very interesting two-day session. I think that it is to your credit as well as to the audience that we have stuck it out for ten hours yesterday and about nine today. We have exchanged a lot of information in these two days. I hope that each session chairman will continue to assist us in looking over the record here, and in putting it in some form that we can show our colleagues.

I hope that each of you have benefitted from this exchange. I didn't expect a clear set of future plans to fall out of this meeting; however, I do think that it has served to sweep out some cobwebs and, perhaps, to jolt us out of a few of our favorite ideas or approaches. It has given all of us a better insight into the problem areas that should be funded in the coming year.

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