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LUNAR LANDING PROPULSION CONSIDERATIONS

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National Aeronautics and Space Administration
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SUMMARY

Lunar landings require 0-g engine start, non-aerodynamic descent to surface, landing and takeoff from an unprepared site without ground crews, and very precise descent and ascent flight paths. Because of these requirements, some characteristics normally considered desirable for propulsion systems will be necessary for rocket propulsion systems of lunar landing vehicles.

The propellant must have high density impulse, low ignition lag, and storability in space. High specific impulse and self-ignition are both desirable.

The overall propulsion system must have operational reliability, simplicity, low weight, and throttling capability. Restarting capability is desirable for descent and ascent phases.

Four propellant combinations considered in the report are ranked in order of relative burn-out velocities: F_2/N_2H_4, F_2/H_2, N_2O_2/N_2H_4, and H_2/O_2. Also, OF_2-diborane should be investigated as a propellant combination because preliminary consideration indicates that it may be superior in some respects to these four combinations.

INTRODUCTION

Certain characteristics, which are desirable for any propulsion system are necessary for rocket propulsion systems of lunar landing vehicles. These characteristics are determined partly by the payload-carrying limitations of earth launch vehicles planned for the next 5 years or so and largely by the conditions peculiar to operation in the lunar environs. Such lunar conditions include: engine start in 0 g in lunar orbit; non-aerodynamic descent to surface; landing and takeoff from an unprepared site, with no ground crews; and very precise flight paths on descent and ascent with little or no margin for error.

This report is a list and review of some propulsion system characteristics which may be desirable or necessary (in light of these lunar conditions):

<u>Propellant</u>

1. high specific impulse } interrelated
2. high density impulse }
3. self-ignition and low ignition lag
4. storability, for several days in space

<u>Overall System</u>

5. operational reliability
6. system simplicity and low weight
7. throttling and restart capability.

1. SPECIFIC IMPULSE

It is an axiom of rocket propulsion performance that the specific impulse (lb-sec/lb) should be "high"--just as specific fuel consumption in a turbojet engine should be "low". This axiom holds for the lunar rocket, subject to certain qualifications to be discussed later. How then, can we obtain a high specific impulse?

To begin with, propellants must be chosen which give a high available energy (high combustion temperature) and yield combustion products of low molecular weight. It is instructive to compare two propellant combinations on these bases (Table 1).

We see that the F_2/H_2 mixture has the higher performance even though the O_2/H_2 has the higher available energy. The reason is that the

O_2-formed molecules are more easily dissociated than the F_2-formed molecules (H_2O versus HF), and thus some exhaust kinetic energy is lost in the dissociation process. This low dissociation factor lends importance to F_2 as an oxidizer.

Table 1

COMPARISON OF PROPELLANT PROPERTIES
OF OXYGEN-HYDROGEN AND FLUORINE-HYDROGEN MIXTURES

	O_2/H_2	F_2/H_2
Available energy of stoichiometric mixture, kcal/kg	3,600	3,110
Optimum equivalence ratio (r*)	2.27	3.33
Temperature, ^{o}K	2,760	3,323
Molecular weight of mixture	9.0	10.01
Specific impulse (I_{sp}) -- pressure ratio, chamber to exit = 20:1	350	364

Now the specific impulse for any given propellant combination is modified by other factors, notably the mixture ratio r = (fuel mass flow)/(oxidizer mass flow), or the equivalence ratio r^*, where $r^* = r/r_s$, with r_s being the stoichiometric mixture ratio -- that ratio corresponding to complete combustion. A fuel-rich mixture is shown by the inequality $r^* > 1$.

Thus, the term r^* shows immediately and conveniently the off-stoichiometric operating conditions of a bipropellant. The optimum I_{sp} generally lies in the direction of rich mixtures ($r^* > 1$), with the maximum depending on the proportion of hydrogen in the overall mixture. In Fig. 1 we see the change of I_{sp} as a function of r^*, for F_2/H_2, with a maximum I_{sp} reached around $r^* = 3$. This curve shows also that beyond $r^* \simeq 2$ the I_{sp} is relatively insensitive to Δr^*. Therefore, an F_2/H_2 propulsion system need not be "highly tuned" for precise control of r or r^*, and a mixture ratio controller is unnecessary if the system has a reasonably high "hydraulic rigidity" (high pressure drops across

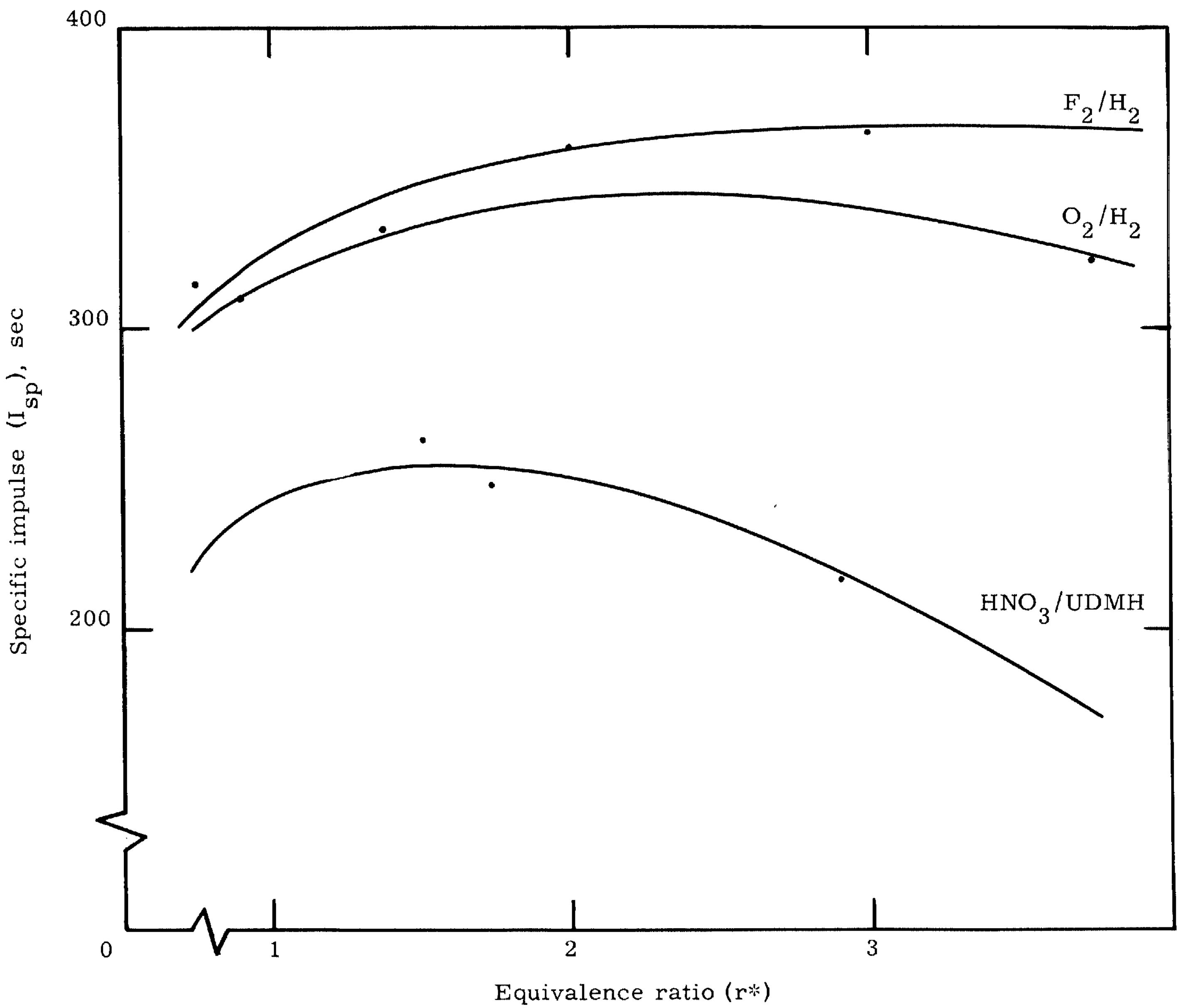

Fig. 1 RELATIONSHIP OF EQUIVALENCE RATIO AND SPECIFIC IMPULSE

flow controls and injector). Thus, insensitivity of F_2/H_2 to r simplifies system design somewhat.

The O_2/H_2 and HNO_3/UDMH mixtures, on the other hand, are more sensitive to Δr or r* (Fig. 1), more care would have to be taken with tank and feed line and injector design, and higher source (tank) pressures would be in order. It is assumed that the lunar propulsion system is pressure-fed for landing and takeoff. A pump-fed system would improve the mixture-ratio control picture and obviate the need for high tank pressures (for any given chamber pressure.) A "cavitating venturi" installed in feed lines near the injector also has a certain flow controlling and stabilizing function through its throttling effect, although a high pressure drop (usually 20% of static) accrues.

It is likely that the judicious use of such venturis would be most beneficial on r control. Incidentally, constant operation at design r assures a good propellant usage ("outage") which, in turn, helps to achieve design total impulse. Achieving such design total impulse (I_t) is a most critical factor for a lunar rocket vehicle when every gram of propellants may be needed.

Another very desirable operating feature is the use of low source (tank) pressures, consistent with the stated needs for hydraulic rigidity and the assumption of a non-turbo propulsion system. Low tank pressures mean low tank weights, low operating demands on flow controls (especially regulators), reduced strains on joints and couplings, and reduced gas absorption in propellants. Thrust chamber pressure plus the feed-line pressure drops determine the tank pressure.

Now, in general, specific impulse does not appreciably increase with an increase in chamber pressure P_c (Fig. 2). The effect of P_c on the characteristic exhaust velocity C* is unimportant. The I_{sp} is a product of two terms, C* and the thrust coefficient C_f; thus,

$$I_{sp} = \frac{C^* C_f}{g}$$

(where C_f is primarily and indirectly dependent on specific heat ratios and

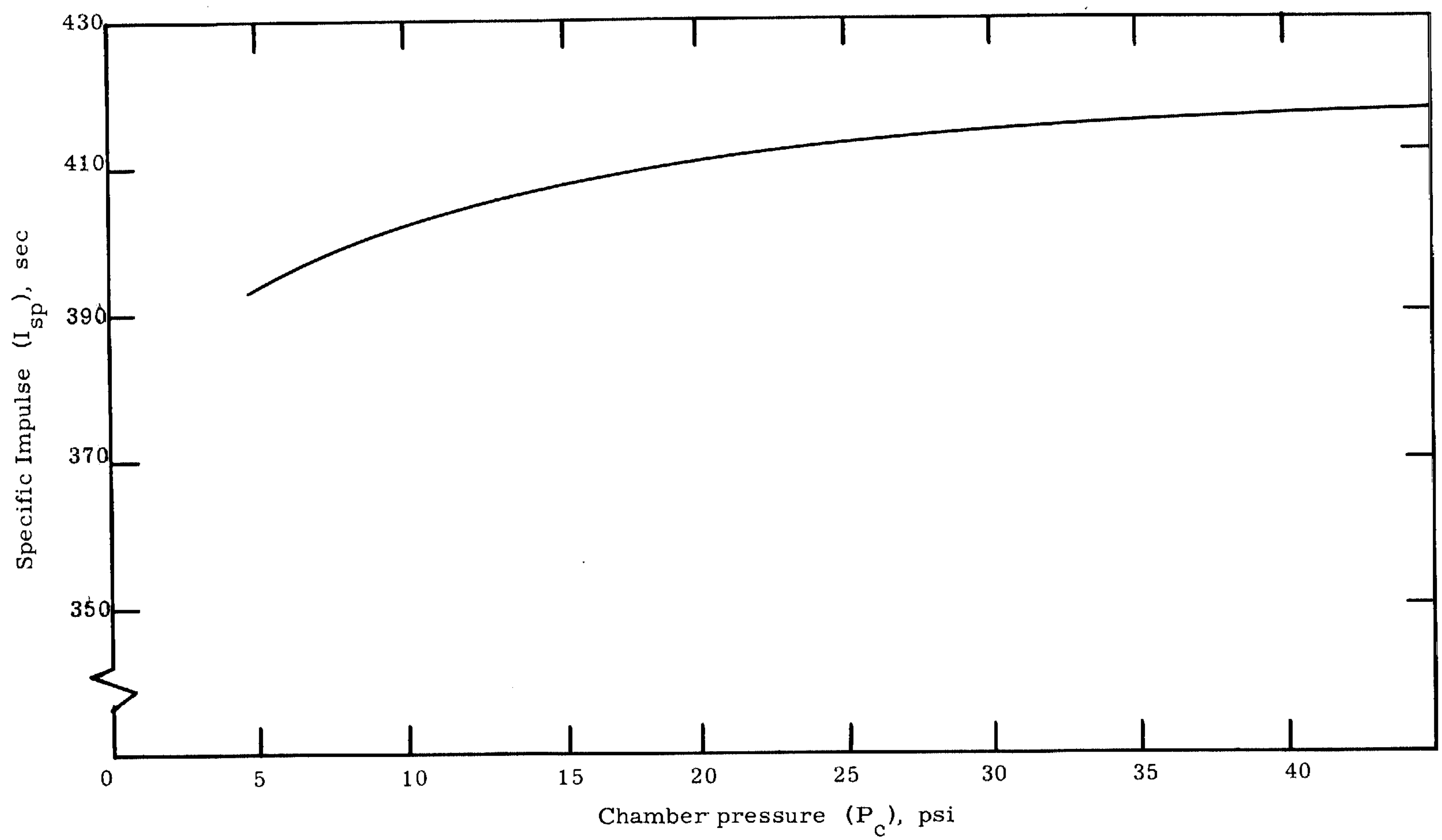

Fig. 2 EFFECT OF CHAMBER PRESSURE (P_c) ON SPECIFIC IMPULSE (I_{sp}) OF FLUORINE-HYDROGEN MIXTURE

(Expansion ratio -- 50:1; frozen equilibrium) expansion ratio; that is, (area of exit)/(area of throat) = $A_e/A_t = \varepsilon$); while

$$C^* = \frac{RT_c}{N}$$

where

R = the specific gas constant

T_c = adiabatic combustion temperature

N = a function of the specific heat ratio (usually about 0.6).

We see then that P_c is not a direct factor, although it does have a small effect on the specific heat ratio.

Thus, the specific impulse can be improved by increasing the nozzle expansion ratio ε, thereby increasing the C_f. Now, as ε is increased, the pressure at the exit plane P_e is reduced to small fractions of atmospheric pressure; this means that a high ε is feasible only at high altitudes (i.e., at low atmospheric pressures or, better yet, space vacuum) if a "negative" C_f is to be avoided.

As an example, increasing ε from 10 to 100 yields a 30% increase in I_{sp}; i.e., (N_2O_4/N_2H_4) increases from 242 sec I_{sp} to 322 sec. This assumes "frozen flow"--i.e., no recombination of dissociated species in the nozzle and, therefore, no further increase in enthalpy or heat release. Actually, in a large long nozzle of ε = 100, equilibrium or recombination reactions would probably further increase the I_{sp} by another 30 sec or so.

To conclude the I_{sp} case, then, it should be evident that a large ε, in a vacuum, accrues in an increase in I_{sp} for any given propellant combination; and even such a "medium performance" combination as N_2O_4/N_2H_4 can yield a respectable I_{sp} of 330 sec or so. A lunar propulsion system, then, especially if dependent on a combination such as N_2O_4/N_2H_4, should have:

a. an $\varepsilon > 40$ if an I_{sp} of around 300-330 sec is to be obtained

b. a low P_c consistent with stability and regenerative cooling criteria--a P_c of about 5 atmospheres is entirely feasible. (The low P_c factor would apply to <u>any</u> propellant combination, of course.)

It should be emphasized that a low P_c can be a mixed panacea, since for a particular thrust level the chamber and nozzle volumes increase with a P_c decrease. An increase in P_c can result in a thrust chamber with the requisite L^*, $\left[\text{(volume of chamber)}/A_t,\right]$ and ε, with an overall volume reduced proportionally (from a low P_c chamber). Furthermore, control moments are reduced (if the chamber is gimballed), dissociation is reduced, and, therefore, C^* is increased somewhat, while the heat transfer rates (Q/A) do not rise in proportion to P_c, so that the liquid cooling capacities are improved. However, despite these considerations, the considerable tank and plumbing weights needed if a pressure fed system is used are serious drawbacks when weight is at a premium, as in a lunar propulsion system.

2. DENSITY IMPULSE

As stated earlier, propellants are usually compared on the basis of I_{sp}. Such comparisons are concerned only with the performance of the propellant-engine combination. For the lunar propulsion system case, another, and perhaps more useful criterion is that of volumetric specific impulse, also called "density impulse" ($I_{sp,\,d}$ or I_d, where d is the mean density of the propellants). The lunar propulsion system may be considered to be really the total aggregate of tanks plus engine, especially in the closely coupled situation of a pressure-fed system. The density of the propellants is a powerful factor in the velocity and range equations of a missile, since the density d is reflected in the mass ratio ζ of the vehicle which is equal to: (initial mass)/ (final mass) $= M_i/M_f$. Looking at the burnout velocity (v_b) equation we have the familiar equation, $\Delta v_b = I_{sp} \cdot g_0 \cdot \ln (M_i/M_f)$, with drag and gravity losses disregarded; at any rate, drag is absent in the lunar environs and $g_{lunar} = 0.166\ g_{earth}$.

Obviously, the logarithmic factor of ζ operates more powerfully on the Δv, than does the I_{sp}. Thus, it is important to obtain a high ζ, or what really is the same thing, as large a loading of propellants as possible in a given volume--with this volume severely circumscribed in a lunar vehicle. But reducing vehicle or tank volume <u>alone</u> does not often increase ζ proportionately--indeed it is easier to achieve a high ζ with a large booster than with a smaller (such as a lunar) vehicle. The propellants (having a cubic-volumetric relationship and structure having essentially a square-area relationship to the size variation of a rocket vehicle) thus achieve a <u>relatively</u> larger fraction of the overall vehicle weight with vehicle size increase. Naturally, the converse is true also.

Now a lunar takeoff propulsion system, vehicle structure, and payload will severely tax the designers' ingenuity in order to achieve a "good" overall M_i/M_f of about 5. The tanks and supporting structure will have to be stronger (and, therefore, heavier) than is the practice in conventional launch vehicles. The vagaries of possible rough landings and need for complete hydraulic and pneumatic integrity will demand unusual structural factors.

The designer thus cannot readily lighten the structure. At the same time, he is being constrained by lunar vehicle volume and weight limitations, while desiring a velocity increment of about 2, 500 meters/sec, for landing or return.

Let us now return to the density impulse (I_d) factor. Consider the initial rocket mass (M_i), propellant mean density (d), and the tanks' volume capacity (V_t); then: $M_i = V_t d + M_f$, so that

$$v_b = I_{sp} g_0 \ln \left(1 + \frac{V_t d}{M_f} \right) . \tag{1}$$

Let us assume also that the best value of $(V_t\, d)/M_f$ that can be obtained with a lunar vehicle is $\simeq 1$ or < 1. Substituting and developing Eq (1) as a series, we have $\Delta v_b = K\, I_d$, with I_d as the governing parameter. It is equally true, of course, that for long-range single-stage vehicles with <u>high</u> mass ratios (including the payload factor) where $(V_t\ d)/M_f \gg 1$, $\Delta v_b \simeq I_{sp}\, g_0 \ln \left[(V_t\ d)/M_f\right] \simeq K'\, I_{sp}$, with the I_{sp} now being the governing factor.

Actually, for the lunar vehicle, with $V_t\ d/M_f$ around 1 and probably not exceeding 3, both I_{sp} and I_d are powerful operators with I_d having the somewhat greater effect. It is instructive to present the effect of propellant bulk density on the Δv factor of a rocket (such as a lunar vehicle) as a function of specific impulse and propellant bulk density (Fig. 3). (The Δv factor is a relative term, permitting comparison of velocities based on 1 for O_2/NH_3.)

Let us then consider the effect on range of four propellant combinations, any one of which might be used in a lunar rocket. We can, of course, transform the Δv parameter into any parameter we choose, such as payload increase, all-up weight, range, etc. Table 2 is a comparison of five combinations.

Table 2

COMPARISON OF FIVE PROPELLANT COMBINATIONS

Propellant	Mixture Ratio (r)	Bulk Density, g/cm^3	I_{sp}, sec	I_d, sec	Δv Factor *
O_2/NH_3	0.769	0.88	266	234	1
O_2/H_2	.125	.43	317	136	1.2
	.286	.26	364	95	1.25
F_2/H_2	.526	.73	338	253	1.4
	.222	0.32	374	120	1.35
N_2O_4/N_2H_4	.909	1.20	263	316	1.3
F_2/N_2H_4	0.500	1.30	316	411	1.8

*Based on Δv of 1 for O_2/NH_3.

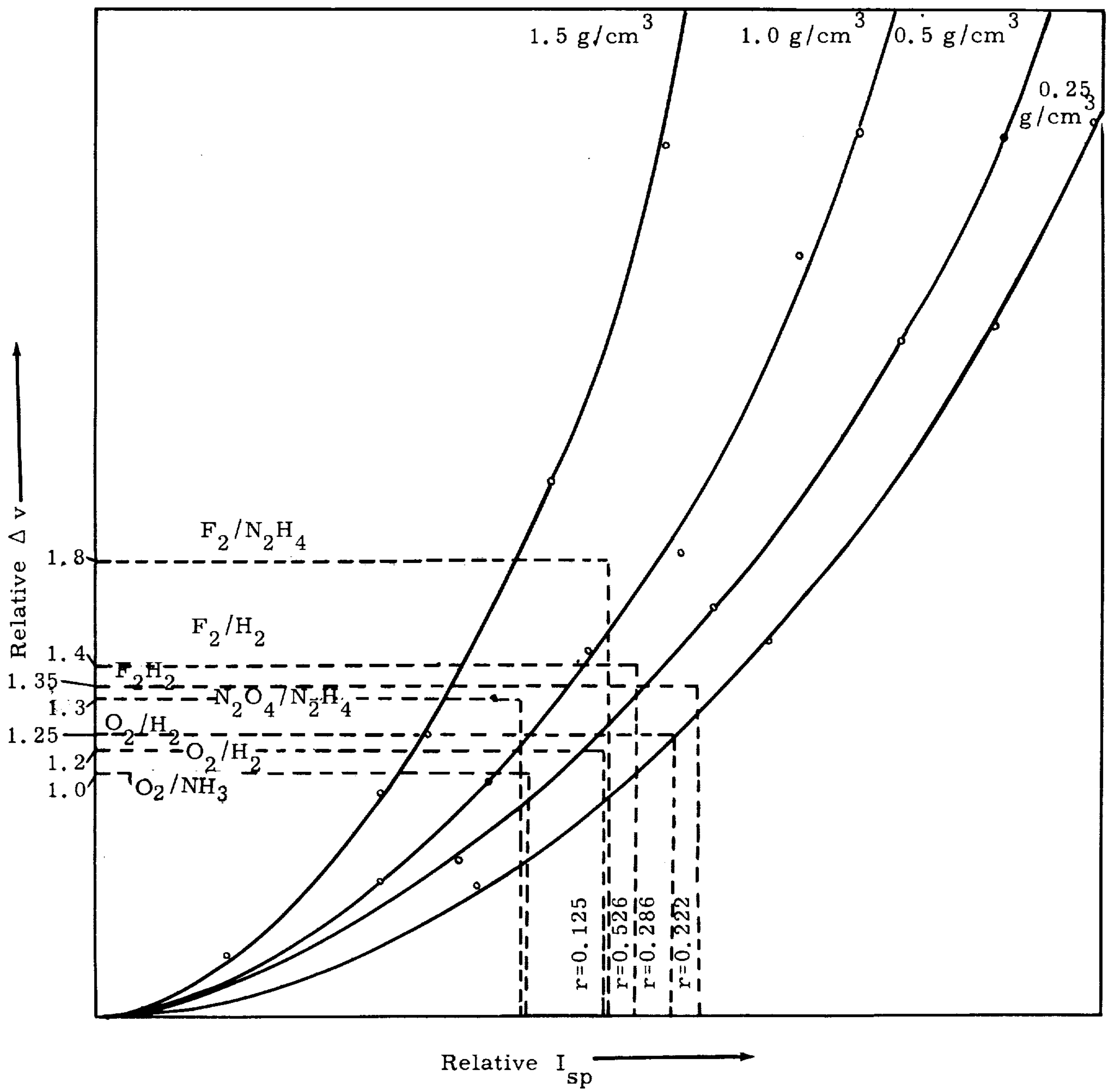

Fig. 3 RELATIONSHIP OF RELATIVE BURN-OUT VELOCITY AND RELATIVE SPECIFIC IMPULSE

Entering the Δv factor values in Table 2 from Fig. 3, we see that F_2/N_2H_4, with a 23% Δv (1.8 over 1.4) improvement of F_2H_2 at $r = 0.526$ or a 50% improvement over O_2H_2 at $r = 0.286$, should be the preferred propellant combination for the particular example of a lunar propulsion system with a restricted weight and volume. This situation is true only when vehicle volumes are fixed and constant.

We note that N_2O_4/N_2H_4, with a Δv factor of $\simeq 1.3$ is almost as good as the F_2/H_2 combination at $r = 0.526$, and better than O_2/H_2 at $r = 0.286$ or $r = 0.125$. Obviously, a propellant combination cannot, for lunar conditions, be chosen entirely on the basis of these parameters. There are modifying factors, such as toxicity, corrosivity, storability in space, sensitivity to combustion instability, cavitation sensitivity, cooling capacity, and ignition characteristics. Cost per kilogram can be ignored here since, in an expensive lunar landing program, the relatively small amount of propellant required for the landing and takeoff maneuver is not likely to influence overall costs significantly.

Table 3 is a comparison of some physical properties of the two "best" propellants (oxidizers)for combination with N_2H_4 (or a mixture of hydrazines) as fuels. We can conclude then that the F_2 oxidizer is superior to N_2O_4 in all but two areas:

- space storage (assume solar heating)
- cavitation sensitivity.

(It is not a good coolant; but the fuel can be used if it is a mixed hydrazine.)

3. SELF-IGNITION AND LOW IGNITION LAG

The ignition lags of N_2O_4 and F_2, which are hypergolic with hydrazines, are listed in Table 3. But O_2 is not hypergolic with any of the fuels considered in this report, so an external ignition system or a hypergolic "slug" start is required for initial and repeated ignitions. Some experimental evidence exists that H_2/O_2 can be ignited in a low pressure (near-space) environment, but tests in space vacuum are necessary. Such an ignition probably will be no great problem. (The X-1, X-2, and X-15 have had a few, not critical, very high altitude start and restart problems.)

However, other factors being equal, a hypergolic start, eliminating (as it does) an external ignition system, is preferable. Coupled with this is the need for the shortest attainable ignition lag (or delay) to prevent "hard" starts from propellant accumulation. The F_2 system has the lowest ignition delay and, therefore, has the advantage over the other systems (Table 3).

4. STORABILITY IN SPACE

The space storage factor cannot be judged on the heat-source-sink basis alone. If we ignored the fluid's specific heat, nonconvection in 0 g, solar heat flux, nonsteady-state heat flow, and convective heat transfer from the lunar surface through the structure, the F_2 would appear better than the N_2O_4 on the basis of

$$E_b = K\,(T^4 - T_o{}^4)$$

where

E_b = energy radiated by a <u>black</u> body

T = temperature of emitter (absolute)

T_o = temperature of surroundings

K = Stephan-Boltzmann constant;

if

T_o = 0 (in space),

E_b = KT^4.

Now for $F_2, T \simeq 50^\circ A$, and for $N_2O_4, T \simeq 290^\circ A$. Therefore,

$$\left[\frac{(dE_b)}{(dt)}\right]_{N_2O_4} \gg \left[\frac{(dE_b)}{(dt)}\right]_{F_2}.$$

Table 3

COMPARISON OF FLUORINE AND NITROGEN TETROXIDE AS OXIDIZERS USED WITH HYDRAZINE(S)

Property	F_2	N_2O_4
Toxicity: A problem only if leakage into cabin is possible -- a remote possibility with good tank and cabin design.	Serious at over 4 ppm when H_2O is present.	Serious at over 7 ppm. Presence of H_2O is influential.
Corrosivity with Al alloys	Negligible in dry state. Can be very serious when temperature rises as in pump seals or in presence of traces of organic or inorganic impurities.	Negligible.
Storability in space (Considering only heat transfer)	Poor → Fair FP: ~ $-218^\circ C$ BP: ~ $-189^\circ C$	Good FP: ~ $-10^\circ C$ BP: ~ $+21^\circ C$ Experiments with rocket payloads and aircraft indicate 0-g behavior also favors use of a noncryogenic.
Combustion instability sensitivity	Apparently quite insensitive; F_2/N_2H_4 reaction is characterized by a low combustion time lag (τ) -- lower than that of N_2O_4. Experience indicates that this benefits stability.	High frequency instabilities have been noted, but no serious terminating instabilities.
Cavitation sensitivity: Coupled to the 0-g state and aggravated by it. Cavitation can be serious in injector domes and feed lines, as well as in pump inlets.	Vapor pressure: 760 mm at $-189^\circ C$ 1 mm at $-218^\circ C$ Rises 759 mm for a $20^\circ C$ temperature rise.	Vapor pressure: 760 mm at $21^\circ C$ 810 mm at $50^\circ C$ Rises 50 mm for a $20^\circ C$ temperature rise.
Cooling capacity	Fair only beyond critical pressure. Would require heavy cooling passage pressures; ablative thrust chambers would eliminate this need.	Good up to saturation.
Ignition characteristic: Liquid-liquid contact assumed. Lag tends to increase as ambient pressure (inside thrust chamber) decreases.	Measured at 760 mm initial chamber pressure: 1.2 msec. Hypergolic, much more reactive than N_2O_4.	Measured at 760 mm: 4.5 msec. Hypergolic.

Obviously, the rate of heat transfer flux, d Q/dt, is reduced when the area, and the emissivity E of the tank wall are reduced ($E < 1$). Conversely the reflectivity of the wall can be made very high; e.g., polished aluminum has a reflection coefficient of 0.97 from 10,000°A to 2,700°A--the region of greatest solar heat flux.

The space storage picture is extremely complicated, and it may not be proper to state that F_2 is "less storable" than N_2O_4, on the basis of solar heat flux alone. In the complex lunar maneuvering this flux effect is variable and transient, so that heat transfer prognostications are almost impossible. The possibility of freezing propellants in space (in solar shadow) cannot be ignored for the noncryogenics.

5. OPERATIONAL RELIABILITY

The probability of cavitation in F_2 can be reduced by increasing feed pressures--but this would mean a weight penalty. The effect of 0 g on cavitation in cryogenics (with their high vapor pressures) is still being investigated.

If non-regenerative uncooled chambers are assumed, the cooling capacities of the propellants are not pertinent. However, if a sublimating ablative lining is used, with an organic filler and an inorganic (siliceous) binder, the F_2 probably will present a somewhat greater corrosion-erosion problem than N_2O_4. The affinity of F_2 for organics is greater than that of N_2O_4, and fluorine compounds, notably HF and F_2O (both combustion species), in the presence of N_2H (also a combustion species) could erode siliceous matter. The use of graphite, titanium, or coated chambers and nozzles might obviate this problem.

Leaving propellant considerations aside now, it can be said that an "ablative" chamber, having no jacket pressure drop or manifolding, leads to a somewhat simpler system design, lower tank pressures, and a chamber less critical for design or manufacturing tolerances and more rugged, with no cooling passages to deform.

6. SYSTEM SIMPLICITY AND LOW WEIGHT

Operational reliability and system simplicity are related. The classical statistical reliability relationship, when the possibility of a chance failure is given as the exponential function e^{-ut}, where u = number of parts and t = time, is a great simplification, but has enough substance to justify a designer's preoccupation with parts reduction. It may be of interest to review how a lunar propulsion system could be "simplified" for high reliability and serviceability. It must be remembered that the mechanical servicing of a system in the inimical environment of the lunar surface is attainable only on the most rudimentary level.

Without attempting to "design" a lunar rocket propulsion system, it is possible to suggest a few technical stratagems:

1. Ablation-cooled thrust chamber
2. Pressure feeding
3. Integral flow controls, such as rupture disks and/or valve-in-head propellant valves
4. Manual propellant shut-off, based on pilot's interpretation of accelerometer and attitude data from a display panel
5. Hand-pressure regulator control of a cold-gas-source pressure, or at least, manual override of automatic regulators and manual shut-off
6. Fixed bang-bang cold gas jet attitude control, or the use of a proven attitude-control system, such as that of the X-15 or Mercury capsule
7. The use of flexible 1,000% overrated pneumatic and liquid lines
8. Manual mixture ratio control with dual-pintle single-cam flow controls.

It is, of course, possible to list many other stratagems that would be desirable and perhaps even necessary; these are only a few of the more obvious ones.

Keeping the number and complexity of system parts to a minimum is also advantageous in keeping the weight of the overall system to a minimum.

The extensive use of such metals as beryllium and magnesium may be warranted because their use should yield a light-weight, sturdy structure capable of surviving a "hard" landing.

7. THROTTLING AND RESTART CAPABILITY

Throttling may be a desirable and necessary feature of a lunar system because of a need to tailor and to trim total impulse precisely in response to information derived from a radar altimeter and drift or approach indicator. The information could be fed to a display panel, for pilot's action, or to an autopilot-coupler propulsion controller.

At any rate, some form of throttling control will have to be devised. Now the thrust developed is a function of nozzle expansion ratio A_{ε} , chamber pressure P_c and specific heat ratio y. Only P_c and A_{ε} can be considered; P_c varies linearly with chamber pressure. Theoretically, the P_c can be reduced to a value just above the critical point--normally about 3 kg/cm^2--and the thrust would be reduced proportionally. Assuming that regenerative cooling (the severest limitation) does not apply, we are still faced with the problem of obtaining full nozzle flow at the low P_c and preventing combustion instability.

Fortunately, with the ambient pressure at 0 (P_a =0), giving a pressure ratio of $\left[(N\ kg/cm^2)\right] / \left[(0\ kg/cm^2)\right] = P_c/P_a = \infty$, we should have substantially full nozzle flow; indeed the nozzle is always under-expanded in vacuum. This circumstance is modified because of frictional, cross-flow and turbulence losses and enthalpy changes caused by re-combinations.

Various schemes for obtaining changes in thrust ΔF have been attemped, including a variable nozzle throat area, propellant feed throttling, segmented injector heads (as in the X-2), multichambers (as in the X-1 aircraft 6,000 C_4 engine), and multistage combustion (as in the X-15 aircraft engine). The smoothest P_c control has been obtained with the multistage combustion, with a combustor-injector firing into a thrust chamber, and main propellant injection into a sonic gas stream. The preferable scheme would be simple flow-control propellant throttling with injector configuration and thrust chamber L* designed for stability at reduced P_c. Such stratagems as "aerated" injector propellant sprays may add undesirable complications to the flow control circuit.

8. CONCLUSIONS

Table 4 summarizes very briefly the situation on propulsion system characteristics as affected by lunar conditions, considering that all weights and volumes are restricted for the landing and takeoff phases.

The propellants ranked according to the Δ v factor are:

1. F_2/N_2H_4
2. F_2/H_2
3. N_2O_4/N_2H_4
4. H_2/O_2.

Ignoring all other considerations, cryogenic propellants always pose some problems of handling and storage, not posed by a noncryogenic, such as the N_2O_4 oxidizer. From this standpoint the N_2O_4 is preferable, if the lower (than F_2) Δ v is acceptable. The N_2O_4 combination is, at any rate, superior to H_2/O_2 which combination in turn, besides raising the problem of H_2 handling, is not hypergolic. The storage problem is of less importance to the lander because of shorter use-time. Here the F_2 would probably be desirable.

However, it must be emphasized that the Δv factor relationships of the propellants discussed in this report are mainly valid only for low volume, high payload-to-vehicle-weight ratio vehicles such as a lunar lander or takeoff stage or a planetary orbiter or lander vehicle.

9. OXYGEN FLUORIDE-DIBORANE PROPELLANT COMBINATION

The special case of the oxygen fluoride-diborane (OF_2/B_2H_6) propellant combination was not reviewed previously because performance data was not at first available during the first writing of this paper.

Table 4

REQUIREMENTS FOR LUNAR VEHICLES

Characteristic	Conclusion	Comments
		Propellant
High I_{sp}	Desirable	
High I_d	Necessary	
Self-ignition	Very desirable	The X-1, X-2 and (to a lesser extent) the X-15 vehicles had some difficulties with re-lighting engines.
Low ignition lag: $<$ 10 msec	Necessary	To prevent severe starting transients and reduce dangers of non-ignition.
Storability in space	Necessary	Although this criterion is not yet well understood.
		Overall System
Operational reliability	Necessary	Can be achieved by reducing number of engine elements, by making extensive ground tests, and by providing a manned link in engine control.
Simplicity	Necessary	
Low weight	Necessary	Possibly achieved by extensive use of such metals as beryllium and magnesium.
Throttling	Necessary	A tried and proven approach should be used.
Restarting	Desirable	For descent (and possibly ascent) phase.

This situation may be summed up as follows:

1. There is a relative paucity of experimental (rocket) data at hand.

2. The available data tend to be inconsistent in such areas as C*, r, bulk density, and flame temperature. However, the disparities are small and should be resolved with further experimentation.

3. Thermodynamic computations and the limited experimental data show relatively high specific and density impulses:

$I_{sp} \simeq 432$ sec. (vacuum)

$I_d \simeq 431$ sec.

4. OF_2 has a boiling point of $128^{o}K$ ($-145^{o}C$) and B_2H_6 of $180^{o}K$ ($-93^{o}C$). Other factors remaining equal, this makes this combination more "space storable" than the combinations with two constituents that are "deep" cryogenic liquids, such as F_2 or O_2, but less storable than the N_2O_4/N_2H_4 combination.

5. The combination is hypergolic; but no consistent ignition delay or "lag" figures are available.

6. The combination has negligible corrosion rates with materials commonly used for tanks and plumbing.

7. The coolant qualities of the two substances are poorly understood and should be determined from an experimental heat transfer analysis.

8. About 2-3 years of thorough combustion and fluid mechanics research and development work is in order before this combination could be expected to be available for operational propulsion systems.

9. With an approximate bulk density of 1.04 g/cc at $-145^{o}C$ (for both propellants) and an estimated I_{sp} of 432 sec (shifting equilibrium and same conditions as other propellants in this paper), the OF_2/B_2H_6 combination has an apparent "quality factor" (obtained from Fig. 3) for Δ v, <u>superior</u> to F_2/N_2H_4 by a margin of 5 to 10%.

BIBLIOGRAPHY

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1. Stosick, A. J., Performance Characteristics and Limitations of Chemical Propellants, Ind. and Eng. Chem., v 48, No. 4, p 722-4, April 1956.
2. Stehling, K. R., Injector Spray and Hydraulic Factors in Rocket Analysis, Journal of the Am. Rocket Soc., v 22, No. 3, p 132-8, May-June 1952.
3. Rothrock, A., "Spacecraft Propulsion", in Current Research in Astronautical Sciences, L. B. Giori, (ed.). Pergammon Press, London, 1961. v 6, p 322-372.