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PRELIMINARY PERFORMANCE APPRAISAL OF NAVY V/STOL TRANSPORT AND SEARCH-TYPE AIRPLANES USING HYDROGEN FUEL

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ABSTRACT

First-cut estimates are given of the performance advantages of liquid-hydrogen-fueled, ejector wing, V/STOL aircraft designed for shipboard delivery and search-type missions (i.e., antisubmarine warfare, and search and rescue). Results indicate that the use of LH_2 could reduce gross weights 30 percent, empty weights 15 percent, and energy consumption 10 percent for a fixed payload and mission. If gross weight is fixed, the delivery range could be increased about 60 percent or the hover time during a search mission doubled. No analysis or discussion of the economic and operational disadvantages is presented.

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PRELIMINARY PERFORMANCE APPRAISAL OF NAVY V/STOL TRANSPORT AND SEARCH-TYPE AIRPLANES USING HYDROGEN FUEL

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SUMMARY

This study considers the possibility of using LH_2 in place of JP fuel for a subsonic, utility-type airplane of possible interest to the Navy. The extremely high heating value of LH_2 fuel (2.78 times that of JP) is offset to some degree by the weight, volume, and indirect structural penalties associated with cryogenic hydrogen tankage. By using semianalytic methods, the complete LH_2 tankage system (tank structure, insulation, boiloff, and unusable fuel) is estimated to weigh approximately 70 percent of the usable fuel. Also, the low density of LH_2 leads to additional fuel volume penalties that are estimated to be a 17 percent increase in fuselage shell weight plus a 25 percent increase in drag for a given gross weight airplane.

But even with these penalties, hydrogen fuel offers significant performance gains for ejector wing, V/STOL, military aircraft designed for subsonic missions. In the case of a transport airplane with a fixed payload and range, the TOGW (takeoff gross weight) could be reduced 30 percent, the empty weight 15 percent, and the energy consumed 10 percent, compared to JP fuel. If transport aircraft are compared on an equal TOGW basis, the payload increase due to LH_2 is about 100 percent for a VTOL (vertical takeoff and landing design) and 50 percent for a STO/VL (short takeoff/vertical landing design); however, the empty weight of the LH_2 airplane would be 20 percent greater. As with JP, a LH_2 transport airplane could also perform a 300-nautical-mile search and rescue mission with about 15 minutes of on-station hover time. Obtaining antisubmarine capability requires an airplane sized for this mission - a 50 000-pound VTOL antisubmarine aircraft could fly a 300-mile-radius mission with 1-hour on-station hover time using LH_2 fuel and less than half this time using JP fuel. In order to completely eliminate the LH_2 performance advantages, the estimated weight of the tankage system would have to be increased by a factor of 2.3.

The best engine cycle for the baseline turbine inlet temperature (2650° R, maximum continuous rotor inlet) is a turbofan with a bypass ratio near 1.5 and an overall pressure ratio near 20. A 500° R turbine temperature increase using additional bleed air for cooling could reduce TOGW 5 percent. Another 2 or 3 percent reduction is theoretically possible by raising the temperatures 1000° R and using the LH_2 fuel flow to cool the bleed air.

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INTRODUCTION

Liquid hydrogen as a fuel for typically long-range aircraft has been extensively studied for supersonic, hypersonic, and subsonic transports as well as for large bulk carriers (e.g., refs. 1 to 4). For such missions the high heating value and/or cooling capacity were attractive and the potential for decreased pollution and for saving fossil fuels are additional benefits. Principal disadvantages were related to the hazards, cryogenic problems, and low density of liquid hydrogen, which required large tankage volumes and related structural penalties. The common denominator for such missions was a large fuel weight fraction using JP fuel. Using hydrogen as a fuel usually resulted in a smaller and lighter airplane for the same payload/range.

Another class of aircraft having difficult missions, which might be aided by hydrogen fuel, is that referred to as VTOL (vertical takeoff and landing). Because of the extra weight required by the VTOL propulsion system, the weight allotted to payload and fuel is smaller than that for conventional-takeoff-and-landing airplanes. Hence, the performance of VTOL aircraft should be quite sensitive to fuel savings even when their fuel fractions are not unusually large. Therefore, the present brief study assesses the gross weight reductions that might occur by designing hydrogen-fueled VTOL aircraft. A typical Navy COD (carrier-onboard-delivery) mission was selected that could operate from Sea Control Ships as well as carriers and land bases. In addition to this basic supply type of mission, a secondary role was assigned for this airplane - ASW/SAR (antisubmarine warfare and search and rescue). Specifically, an airplane designed for COD missions was sought that would also provide reasonable ASW/SAR capability.

Sometimes these types of aircraft are envisioned to use a short takeoff run instead of a VTO (vertical takeoff) while still landing vertically. Hence results are given for this mode (STO/VL (short takeoff/vertical landing)) of operation, too.

The ejector wing propulsion concept was chosen since the results of reference 5 showed its general competitiveness with the usual VTOL propulsion systems providing that a thrust augmentation ratio of about 1.6 and low ejector system weight are achieved. Also, the low downwash should aid erosion and noise problems.

In order to afford a systematic basis of comparison, fuselage volumetric efficiency constraints were established for the JP and hydrogen-fueled aircraft. Fuselage size for a fixed length-to-width ratio was varied until the volumetric constraint was satisfied.

For each fuel and mission the best mixed-flow turbofan engine cycle (bypass ratio and overall pressure ratio) is identified. Also, the concept of using the LH_2 fuel flow to cool the bleed air for turbine cooling is explored. The hope is to significantly reduce the bleed flow required and thereby improve engine performance.

In this quick-scan study no judgment is made of the important military operational problems involved with using liquid hydrogen as a fuel.

ANALYSIS

The general approach to studying the advantages of hydrogen as a fuel for logistic aircraft involved the use of a computerized synthesis of aircraft aerodynamics, weight, and propulsion. Thus, for a chosen mission flight path and payload the aircraft and engine sizes are iterated until the fuel required equals that available and the volume utilization constraint is also satisfied. This step gives an aircraft gross weight for a selected configuration, engine cycle, and fuel type.

Airplane Configurations and Missions

The aircraft configuration used in this study was selected from previous applicable studies since the primary purpose herein is to demonstrate the performance benefits of hydrogen fuel. Accordingly, the ejector wing COD arrangement of reference 5 was selected and is shown schematically in figure 1 with pertinent parameters given in table I. The COD airplane has a high wing with two nacelles and a conventional logistic-type fuselage. Vertical thrust is achieved by the ejector flap system with an assumed 1.6 augmentation ratio (lift forces divided by ideal thrust at ejector primary inlet conditions).

The COD primary and ASW/SAR secondary mission profiles are shown in figure 2. The COD mission is a one-way trip with cruise at Mach 0.7, 36 000 feet. The ASW/SAR mission is a two-way trip to a sea-level station 300 nautical miles from the base with the same cruising conditions as the COD mission. On-station, the aircraft loiters for 30 minutes and hovers 5 to 15 minutes if it is a SAR mission or considerably longer, on the order of an hour, if it is an ASW mission. Vertical takeoffs and transitions are assumed to require a total of 2 minutes at a net vertical thrust level of 1.1 times the gross weight. Short takeoffs are assumed to require 2 minutes of idling plus a 300 foot deck run into zero wind. Vertical landing is assumed in either case. The payload for either mission is arbitrarily assumed to weigh 5700 pounds (700 lb of mission related avionics and 5000 lb of delivered payload in the case of a COD mission).

Airframe Weight and Aerodynamics

Major airframe component weights such as wings, tails, and fuselages were estimated with the statistical method of reference 6, and modified where necessary by semianalytic corrections to account for the ejector flap system. Statistical correlations were also used for the conventional subsystems such as surface controls, electronics, inlets, air conditioning, and so forth. The drag coefficients of all airframes were computed as a function of Mach number and geometry using modeling techniques similar to those discussed in reference 7. In this technique the individual component drags are summed to give the total zero-lift drag. The individual drags are based on geometrical properties such as surface area, thickness, sweep angle, length, width, and so forth. The induced drag and compressibility drag rise terms are then added to the zero-lift drag to obtain the total drag.

Propulsion Systems

JP-fueled engines. - The JP propulsion engines are assumed to be two-spool mixed-flow turbofans designed at the current level of technology (e.g., F401). Standard day performance data for these engines were generated with the GENENG computer program (ref. 8) assuming a 0.975 inlet pressure recovery and a maximum continuous turbine-rotor inlet temperature of 2650° R. The VTO thrust-to-weight ratio was set at 1.1 to allow for reingestion, "suck-down," and control losses. Bare engine weights and dimensions were calculated with the statistical correlation method of Gerend (ref. 9) and the following items were added to the bare weight:

Remote gear box, lb.	135
Diverter valve, lb	.150 × (airflow/265)
Exhaust nozzle, lb	80 × (airflow/265)
Duct system.	ref. 10 data

LH_2-fueled engines. - The baseline LH_2 engines were assumed to differ from the JP engines in only one respect - their fuel consumption rates were a factor of 2.78 less than for JP to account for the difference in heating values. This assumption avoids recomputing vast quantities of engine performance data (e.g., to reflect changes in fuel-air ratios and thermodynamic properties) and has been shown (ref. 11) to involve insignificant errors. Similarly, the engine weights and dimensions are also assumed to be independent of fuel type in accordance with the results of reference 11.

As a departure from this standard set of LH_2 assumptions, however, a probe was also made of the potential cooling benefits afforded by this cryogenic fuel. In particular, the turbine inlet temperature was allowed to rise from 2650° to 3150° R and then to 3650° R. These 500° R increments were assumed to involve absolutely no penalties (e.g., for heat exchanger weight, increased compressor bleed flow, etc.) in order to determine the potential effect of this change alone - without clouding the effect with necessarily crude estimates of such penalties. This sweeping assumption, of course, ignores the difficulty of implementing such high temperatures without penalties, but does answer the question of whether such an effort is even warranted.

Fuselage Volume Constraint

Due to the low density of LH_2 (4.43 lb/ft^3), it is necessary to enlarge the fuselage significantly in order to accommodate the large volume of fuel. A simple, first-order approach was used that involves setting a constraint on the total enclosed fuselage volume so as to ensure enough volume to house the LH_2 tank. The basis of the method is a side study which revealed that for existing JP fueled aircraft the average fuselage volume was allocated as follows:

Payload and void space, percent	55
Aircraft subsystems, percent.	45

The aircraft subsystem category includes the cockpit, fuel tankage, avionics, propulsion subsystem, power actuators, ductwork, and miscellaneous. This breakdown is, of course, a reflection of how efficiently the various items are packaged. For the purposes of comparing LH_2 airplanes with JP airplanes, the fuselage volume was constrained such that

$$E \equiv \frac{\text{aircraft subsystems volume}}{\text{total fuselage volume}} \begin{cases} \leqq 0.5 \text{ (for JP)} \\ \leqq 0.7 \text{ (for } LH_2\text{)} \end{cases}$$

In essence we are arbitrarily specifying that the volumetric efficiency of the JP airplanes must be nearly as good as that of typical existing airplanes and that LH_2 airplanes must provide about the same usable space as the JP airplanes. A higher limit ($E \leqq 0.7$) was set for LH_2 aircraft in recognition of the fact that the LH_2 fuel tank can be packaged very efficiently within typical fuselages. A spot check indicated that a LH_2 airplane constrained by $E \leqq 0.7$ yields the same usable volume as a JP airplane constrained by $E \leq 0.5$. This methodology is admittedly rather crude and a more detailed study is suggested to improve its accuracy. In any case, the JP constraint was never binding and the LH_2 constraint was binding for only a few of the more difficult missions as will be shown later in the RESULTS section.

In application, a statistical model is first used to predict the fuselage dimensions based on gross weight and length-to-diameter ratio. If the constraint is violated the fuselage is enlarged so as to satisfy the constraint while keeping the length-to-diameter ratio fixed.

Hydrogen Tankage System Weight and Size

The LH_2 tankage weight estimates are based on the Convair Division of General Dynamics Corporation (GDC) study, performed under contract to the Air Force Flight Dynamics Laboratory, of a Mach 6 manned hypersonic cruise vehicle's LH_2 tankage (refs. 12 and 13). The GDC study involved a demonstration LH_2 tank of approximately the same size (3000 lb of usable fuel) as required by the present study. The GDC tank is a nonintegral, insulated tank of "Siamese" configuration (cross section of two intersecting circles, 64 in. in diam.) 8 feet wide and 24 feet in overall length. The structure is a pressure membrane with frame stiffening designed for 30-psig maximum operating pressure and 3 g ultimate loads. The skin structural material is a thin gage (0.016-in.) nickel-based superalloy (alloy 718). The insulation system is an all-microquartz layer in a helium environment.

Using the GDC tank as a reference, several adjustments were made to account for the different missions involved. First, the load sensitive structure was strengthened to withstand 7 g's instead of 3 g's. This was done by grouping the structural components according to whether or not they were sized by inertia loads, and adjusting the inertia load sensitive group with a scaling law. In particular, the actual GDC tankage system (exclusive of insulation) weighed almost 1000 pounds of which 444 pounds was inertia load sensitive. Hence,

$$W_{tank} = \left[555. + 444. \left(\frac{U_N}{3}\right)^{0.6}\right] \frac{W_{fuel}}{3000} \tag{1}$$

where U_N is the ultimate flight load factor and it has been assumed that tank weight scales linearly with fuel weight W_{fuel} and the factor $(U_N/3)^{0.6}$ is based on the relations given in reference 13. Since $U_N = 7$ in this study, equation (1) reduces to

$$W_{tank} = 0.432\ W_{fuel} \tag{2}$$

Second, the insulation material was changed from high-temperature microquartz to a PVC foam called Klegecell H 917 (ref. 14) since the skin temperature near the tank would seldom exceed 100° F. Its thickness was computed with the aid of an analytical approximation (ref. 15) for steady-state, wet-wall conditions and based on minimizing the sum of the insulation plus boiloff weight. The insulation thickness at the bottom of the tank L_{bot} is

$$L_{bot} = \sqrt{\frac{Kt_f(T_S - T_H)}{\rho h_{fg}}},\ \text{ft} \tag{3}$$

where K is the average insulation conductivity (0.013 Btu/hr-ft-°F), t_f is flight time plus an hour to account for holds, $T_S - T_H$ is the temperature difference between the aircraft skin and hydrogen (560° F), ρ is the insulation density (3.0 lb/ft^3), and h_{fg} is the hydrogen heat of evaporation (193. Btu/lb). For a typical 4-hour flight this yields an insulation thickness of 3 inches at the bottom of the tank (wet wall condition). Since the top thickness (dry wall condition) was one-fourth of the bottom thickness in the GDC study, an average thickness of 5/8 L_{bot} was used to compute the insulation weight:

$$W_{ins} = [\rho(5/8\ L_{bot}) + 0.15]A_{tank},\ \text{lb} \tag{4}$$

The first term in this expression represents the bare insulation weight, while the second term represents the weight of the adhesive and fiberglass covering (ref. 15). The tank surface area A_{tank}, assuming a constant diameter cylindrical tank of circular cross section with hemisphere ends, is

$$A_{tank} = \pi(\text{tank length})(\text{tank diameter}),\ \text{ft}^2 \qquad (5)$$

These approximate insulation weight and thickness equations are based on the minimization of the sum of the insulation and boiloff weights. As a result of this criterion (ref. 15), the hydrogen boiloff weight is equal to the bare insulation weight and is added to the usable fuel to compute the total fuel load. When computing the tank volume, an additional 10 percent is added to the fuel load volume to account for unusable fuel and ullage needs. The diameter of the LH_2 tank is determined such that a 4-inch clearance is allowed between the fuselage skin and the insulation outer surface as illustrated in figure 1. A detailed tankage weight breakdown is given in table II for both the reference GDC hypersonic design and an adjusted COD design for a 4-hour flight.

No allowance was made for cryopumping. This assumption is based on the tentative selection of Klegecell H 917, a relatively new rigid plastic insulation material (ref. 14), which is impermeable to air at all temperatures. This material has approximately the same thermal performance as sealed-foam installations and has good strength properties.

RESULTS AND DISCUSSION

The estimated sizes for both JP- and LH_2-fueled COD airplanes are presented in figure 3 for a fixed payload size of 5700 pounds. Since the effects of mission range, takeoff mode, and fuselage volume constraint are also shown in figure 3, the discussion of this figure is separated into two parts - one dealing with the advantages of LH_2 and the other dealing with these other aspects.

Advantages of LH_2 Fuel

Figure 3(a) shows that the TOGW (takeoff gross weight) of LH_2-fueled COD airplanes is 25 to 35 percent less than that for JP airplanes, depending on the range. This TOGW reduction would yield the same percentage reduction in engine size and cost, which is especially significant for a VTOL or STO/VL aircraft with its high propulsion system weight fraction. Lower TOGW would also yield a physically smaller airplane that would offer important space and handling advantages for operations onboard ships.

Figure 3(b) shows that the OEW (overall empty weight) of LH_2 airplanes is also less than that for JP airplanes. Again the advantage is range-dependent, varying from 10 to 20 percent when range increases from 1500 to 2500 nautical miles. Although the OEW reduction for LH_2 is only about one-half of the TOGW reduction, it is still significant since it implies a proportionate reduction in airframe cost. On the other hand, a LH_2 tankage system is quite expensive and its cost might completely overshadow the basic cost savings due to lower OEW.

Another cost consideration is the fuel cost which injects even more uncertainty into the comparison since the potential LH_2 price for a large-scale market is not easily predicted. Nonetheless, it is worth mentioning that a LH_2 airplane would consume 5 to 15 percent less energy than its JP counterpart due to its smaller size and hence drag. This energy savings may not be particularly important for a small fleet of special purpose Navy aircraft, but the implications for a whole LH_2 military inventory could be very significant. Naturally the energy cost required to produce LH_2 and JP fuel must also be accounted for when viewing the overall energy consumption differences between these two fuels.

Effects of Range, Takeoff Mode, and Volume Constraints

These other effects are also shown in figure 3(a). The STO/VL curves are presented for a 300-foot takeoff ground roll into zero wind. Designing for STO/VL instead of VTOL reduces the TOGW 15 to 25 percent for a fixed range; or, for a fixed TOGW, it increases the range 30 to 40 percent. Range itself has a very strong effect on TOGW. A range increase from 1000 to 2200 nautical miles increases the design TOGW of a VTOL JP airplane from 30 000 to 60 000 pounds.

The effect of imposing the fuselage volume constraint is also illustrated in figure 3(a) with the dashed curves. The dashed curves represent solutions ignoring the $E \leqq 0.7$ volumetric efficiency factor constraint discussed in the section ANALYSIS. No dashed curves appear for JP since the constraint ($E \leqq 0.5$) is not binding. Even for LH_2 the constraint is not binding unless the range is over 2000 miles - and even then, the effect of the constraint is not very severe. The remaining results are presented assuming the constraint is imposed.

Comparison of LH_2 and JP Airplanes Having Equal Design TOGW

Tables III and IV are presented to illustrate the weight breakdowns of a JP and LH_2 airplane, each with a design TOGW of 40 000 pounds and 2000 mile range. Both a VTOL design and a STO/VL design are listed for each fuel. The constant design gross weight basis was selected to clearly identify the component weight differences. For example, due to its large volume of fuel, the LH_2 airplane's fuselage shell is 45 percent wider than the JP fuselage shell, is 20 percent longer, weighs 17 percent more, and increases the airplane drag 25 percent. Also, the LH_2 tankage plus insulation weight (3341 lb for the VTOL design) represents a sizable 14 percent of the empty weight. The LH_2 fuel weight is less than half of the JP fuel weight, however, and the net effect of these and the other weight differences is a 98 percent increase in payload weight for a LH_2 VTOL or a 47 percent increase for the STO/VL design.

Figure 4 complements these tables by showing a payload summary with different takeoff modes. Results for the VTOL designs are shown both when operating VTOL and when overloaded and operating STO/VL. The gross payloads of the STO/VL designs are considerably greater (about 3000 lb) than the VTOL designs. But the overloaded VTOL designs operating STO/VL have the greatest payload of all - more than twice their design payloads for VTOL operation. However, the VTOL-designed airplane operating in the STO/VL mode must operate with a reduced ultimate flight load factor of 5.6 instead of the 7.0 design value due to the increased TOGW of 50 150 pounds (extra fuel as well as payload is needed to maintain fixed range). This is the reason the STO/VL-designed airplanes (with $U_N = 7.0$) cannot carry as much payload as the overloaded VTOL designs. In any case, the payload advantages of LH_2 are quite apparent.

On the other hand, the ground rules could have been set up differently, such that the LH_2 benefits would be markedly reduced. For example, if an LH_2 airplane is designed and built for VTOL and then placed into STO operation, it would be very difficult to acquire additional range capability. This is because the usual practice of adding supplementary fuel tanks would be a very complicated and messy task for liquid hydrogen. In such a case relaxing the VTO constraint would not offer a choice of either increased range or payload - just payload. This places a high priority on designing the LH_2 tankage volume for the maximum fuel load envisioned - an important consideration if the airplane is intended for multipurpose usage.

Engine Cycle Selection

The propulsion system consists of a pair of wing-mounted mixed-flow turbofans connected to an ejector wing system with an augmentation ratio ϕ of 1.6. The turbofans were assumed to have a BPR (bypass ratio) of 1.5, an OPR (overall pressure ratio) of 20, and a turbine-inlet temperature (maximum continuous rotor) of 2650° R. The basis for selecting this particular engine cycle as a baseline is discussed in this section, along with the possibility of substantially raising the turbine-inlet temperature by using the liquid hydrogen fuel flow to cool the compressor bleed air.

Increased turbine temperatures with LH_2 cooling. - Theoretically, the heat sink capacity of LH_2 could be used to help cool the turbine section of an engine. If enough cooling capacity is available, this advantage could be exploited to substantially raise the turbine-inlet temperature and lower the quantity of compressor bleed air. Several calculations and assumptions were made to assess the potential of LH_2 cooling without attempting to define the associated hardware.

For comparison purposes, TIT (turbine-inlet temperature) was varied for a typical LH_2 COD airplane without any cooling assistance from the

LH_2. The result of this variation on TOGW is shown in figure 5 along with the assumed bleed schedule (the dashed curves) as calculated from the data given in reference 16. Full-film coverage cooling is assumed using advanced fabrication techniques with an average bulk metal temperature of 1850° F for the stators and 1750° F for the blades. The bleed is shown to increase from the baseline value of 2.6 to 23 percent as TIT is raised from 2650° to 3650° R. The engine weight is assumed to be explicitly independent of TIT here. However, the increasingly large bleed flows eventually cause the engine thrust-to-weight ratio to decrease and an optimum TIT is formed at about 3250° R. At this point the TOGW is $5\frac{1}{2}$ percent lower than at the baseline TIT of 2650° R.

The solid lines in figure 5 indicate the potential improvement in this situation if the LH_2 fuel flow is passed through an LH_2-bleed air heat exchanger before entering the combustion chamber. The heat transferred between the bleed air and LH_2 fuel is assumed to be one-half of that possible with an infinite sized heat exchanger. This assumption was used in lieu of an accurate heat exchanger weight model and is accompanied by a parametric representation of the total cooling apparatus weight as shown in the figure. The net effect of using one-half of the LH_2 cooling capacity is to lower the bleed flow by about 50 percent. This, in turn, reshapes the TOGW curves such that an optimum TIT below 3650° R does not exist. However, the improvement over the best uncooled bleed case is rather limited - at 3650° R the TOGW could be reduced an additional $2\frac{1}{2}$ percent if the cooling apparatus weighed nothing. This advantage disappears completely if the cooling weight penalty is 15 percent of the installed engine weight.

Additional details concerning five key cases shown in figure 5 (numbered circles) are given in table V where the columns are labeled with the corresponding numbered circles. As turbine-inlet temperature was increased, the engine cycle variables (bypass ratio and pressure ratio) were reoptimized as reflected by the increases in bypass ratio from 1.5 to 2.25 and overall pressure ratio from 20 to 25. This expected trend also produced a secondary benefit - the turbine-exit pressure and temperature increased but the temperature remained low enough to continue the use of titanium as the ejector ductwork material. Hence, the higher energy density of the hot gas permitted smaller duct sizes and weight. The engine and ductwork weight reductions, in turn, permit a smaller airframe and less fuel. The overall empty weight reductions are about the same percentage as the TOGW reductions. For example, the OEW of the LH_2 cooled bleed case at 3650° R (case 5) is 2.8 percent lower than the OEW of the best uncooled case (case 2).

From these results it appears that increased TIT would save moderate amounts of empty weight and fuel, but that the cooling capacity of LH_2, although helpful, is not a major source of improvement.

Bypass ratio and pressure ratio. - The baseline engine cycle selection of 1.5 for the bypass ratio and 20 for the overall pressure ratio was based on the results displayed in figure 6. Two figures of merit are used in this figure to evaluate engine cycles. Each cycle is used to size an airplane in terms of VTOGW that is capable of performing the primary COD mission (1500-n mi range and 5700-lb payload); then the on-station hover time is computed for the same airplane flying the secondary ASW/SAR mission (300-n mi radius and 5700-lb payload). For each bypass ratio (0, 0.75, 1.5, and 2.25) a solid line is drawn that connects the four overall pressure ratio points (10, 15, 20, and 25). A dashed line is also drawn that connects the minimums of each bypass ratio line. The most desirable engine cycle is one that produces low VTOGW and large on-station hover duration. Hence, points at the bottom of the dashed curve or slightly to the right of it are the best choices to satisfy this duality of criteria. The solid symbols satisfy this condition and denote the chosen cycle - a bypass ratio of 1.5 and a pressure ratio of 20 for both JP and LH_2.

Note that LH_2 offers about 15 percent more hover time than JP but that neither fuel offers more than 17 minutes - a meager amount of hover duration that would probably suffice for a SAR mission but not for an ASW mission. To provide adequate ASW capability, it would be necessary to consider a much longer range COD with more payload or to size the aircraft for the ASW mission rather than the COD mission. It might be imagined that the 15 percent hover time advantage of LH_2 ought to be larger in view of LH_2's much lower sfc (specific fuel consumption). However, because of its low sfc the LH_2 airplane requires considerably less fuel for the design COD mission and therefore has less fuel available for SAR hovering. Thus even though the LH_2 airplane burns its fuel very sparingly (relatively) during hover, it has much less fuel aboard to allot to hovering and this offsetting effect results in the rather modest increase in hover duration.

Secondary ASW Mission Considerations

From the preceding discussion it is obvious that the original notion of using a small COD designed VTOL airplane to fulfill a secondary ASW mission involving a large amount of hover duration is in error regardless of fuel type. Hence, it is of interest to see (1) just how large an ASW-sized-airplane would need to be in order to obtain reasonable hover time capability and (2) how much hover time can be increased through the use of LH_2. These questions may be answered with the data of figure 7 which shows how TOGW varies with on-station hover time for an ASW-designed aircraft. TOGW depends strongly on hover time. Due to the asymptotic nature of these curves the JP airplanes are limited to hover times less than 60 minutes and LH_2 airplanes to less than 100 minutes, regardless of TOGW. At any given TOGW, a LH_2 airplane could hover twice as long as a JP airplane. Only the LH_2 airplanes offer hover times long enough to be termed reasonable ASW candidates. At TOGW equal to 50 000 pounds, for example, the JP hover time is 28 minutes and the LH_2 hover time is 60 minutes for VTOL airplanes. An extra 13 minutes could be gained in either case by using a STO/VL design with a 300-foot ground run in zero wind instead of a VTOL. A nonzero wind would not allow more hover time since the STO engine size with zero wind is just barely sufficient to permit on-station hovering. Thus the only possible benefit of a wind would be to reduce the ground run required to takeoff.

Effect of LH_2 Tankage Weight Assumption

The greatest uncertainty in this study is the weight penalty associated with the LH_2 tankage system. The effect of changing this weight assumption is shown in figure 8 for a typical COD mission. Note that the LH_2 airplane would be just as heavy as the JP airplane if the assumed LH_2 tankage weight were increased by a factor of 2.35. In this particular case the tankage weight would increase from nearly 2000 to 6000 pounds as a result of the combined effects of increasing the relative tankage weight factor K and increasing the VTOGW (to maintain the same range and payload as K rises). Thus, doubling the tankage weight would effectively erase the advantages of LH_2; however, weight increases of up to 50 percent could be tolerated to the extent that the VTOGW advantage would still be a respectable 17 percent (instead of 24 percent for $K = 1.0$).

Effect of Body Fineness Ratio and Wing Loading

As pointed out in the ANALYSIS section of this report, most of the airplane geometry parameters were arbitrarily set at representative values since the goal of the report is simply to compare two different fuels. It was decided, nonetheless, to select the optimum value of body fineness ratio L/D since, conceivably, the large LH_2 tank could influence this choice markedly. The design wing loading's influence on TOGW was also scanned in order to estimate the "best" value of wing loading W/S for different design takeoff modes (i.e., VTO against STO).

Body fineness ratio. - The effect of this parameter is shown in figure 9 for a typical COD mission. There exists a shallow minimum in VTOGW for both JP and LH_2 aircraft. The optimum L/D for JP is about 7 while it is about 6 for LH_2; these values were selected as baselines. The lower value for LH_2 occurs because its large volume requirements emphasize the need to minimize surface area (reduce drag and tankage insulation weight) and better the structural efficiency. However, the reduction in VTOGW is only 500 pounds for the LH_2 airplane when shifting from $L/D = 7$ to $L/D = 6$.

Wing loading. - The effect of design wing loading W/S is displayed in figure 10 for a typical COD mission. The top half of the figure shows that the optimum W/S is 90 pounds per square foot for VTOL designs and slightly less than 70 for STO/VL designs (the values used as baselines for this report). The STO mode lowers the optimum W/S because of the influence W/S exerts on engine sizing when constraining the takeoff run to 300 feet. Reducing W/S lowers the required thrust for takeoff, which lowers the propulsion system weight since the engines are sized by the takeoff constraint.

The lower half of this figure (for VTOL only) shows how the wing weight and fuel weight are affected by W/S and thereby produce the optimum W/S. Wing weight declines with increasing wing loading while the fuel weight required for the design mission generally rises. The tradeoff produces a minimum at the intersection of the two curves - just shy of 90 pounds per square foot. Of secondary interest is the eventual rise in fuel weight if W/S becomes too small because of the increased induced drag during cruise.

The type of fuel does not materially influence the optimum W/S as shown in figure 11. Curves for JP and LH_2 fueled VTOL designed airplanes are shown both at the fixed cruising altitude (36 000 ft) assumed in this report and also at the optimum altitude (varies with W/S). Note that, for the fixed altitude, the optimum W/S is 90 pounds per square foot for both fuels. If the cruise altitude is optimized, however, the optimum W/S drops to 75 pounds per square foot for the JP airplane and 80 pounds per square foot for the LH_2 airplane. The savings in TOGW that results from removing the fixed cruising altitude constraint and reoptimizing the wing loading is small - 1 percent for the JP airplane and 1/2 percent for the LH_2 airplane. The optimum cruise altitude is about 40 000 feet using either fuel.

CONCLUDING REMARKS

The results of this brief study indicate that hydrogen fuel could save 15 percent in airplane empty weight and 30 percent in gross weight for moderately difficult V/STOL subsonic missions. Even greater gains are possible for long range/large payload or long hover time requirements. These results are quite dependent on the volume and weight penalties associated with LH_2. Doubling these penalties, for example, would nearly eliminate the advantages of LH_2. Hence, a more accurate assessment of the advantages of LH_2 for these missions requires an in-depth study. Nonetheless, it is difficult to imagine the estimates being that far in error. A more refined analysis in the Convair Division of General Dynamics Corporation study indicates that an optimized tank structure would weigh 20 percent less than that assumed here. Such a theoretical weight savings could be used to offset installation penalties not accounted for in this analysis.

Beyond these performance advantages lies a whole multitude of operational and cost disadvantages that need to be considered before a true picture of LH_2's attractiveness is brought into focus. A wide-scope study is needed that would include all of these aspects. Such factors as fuel cost and availability, fleet size, development cost, refueling delays, and so forth have an important bearing on the question of using LH_2 in aircraft. A complete systems analysis such as this, although much more difficult than the present study, is needed to properly assess LH_2's competitiveness.

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TABLE I. - AIRPLANE ASSUMPTIONS

Design wing loading, lb/ft^2	
VTO	90
STO	70
Aspect ratio	4
Taper ratio	0.35
Leading edge sweep	30
Thickness ratio (root/tip)	0.14/0.12
Fuselage length to diameter ratio	
JP	7
LH_2	6
Number of engines	2
Ultimate load factor	7
Thrust-to-weight ratio on standard day	1.1
Cruise Mach number	0.7
Cruise altitude, ft	36 000
Payload, lb	5700

TABLE II. - LH_2 TANKAGE WEIGHT ESTIMATES

	GDC Mach 6.0, 3g, $1\frac{1}{2}$ hr design		Adjusted for Mach 0.7, 7 g, 4 hr	
	Weight, lb			
Tank structure (Alloy 718)				
Cylinder skins	311	895	311	1186
Domes and hatches	143		143	
Main frames	60		100	
Intermediate frames	91		151	
Center beam	187		310	
Doublers and supports	103		171	
Subsystems (pump and vents		98		98
Insulation[a]		726		347
Liquid				
Usable	2980	3410	2980	3395
Unusable	152		152	
Boiloff	278		263	
Total tankage and fuel		4912		5026
$\frac{\text{Structure and subsystems}}{\text{Usable fuel}}$		0.261		0.432
$\frac{\text{Insulation weight}}{\text{Tank surface area (561 ft}^2}$		1.3 lb/ft^2		0.62 lb/ft^2

[a]Microquartz for GDC design, PVC foram for COD adjusted design.

TABLE III. - WEIGHT STATEMENTS FOR 2000-NAUTICAL MILE COD AIRPLANES

	JP fuel		LH_2 fuel	
	VTO design	STO design	VTO design	STO design
	Weight, lb			
Wing	2 632	2 967	2 746	3 086
Fuselage	5 003	5 003	5 848	5 800
Horizontal tail	636	591	638	594
Vertical tail	313	359	315	361
Landing gear	1 472	1 472	1 472	1 472
Nacelles (2)	1 260	924	1 260	924
Propulsion subsystems[a]	1 053	967	293	266
Surface controls	1 195	1 182	1 135	1 124
Furnishings, instruments, air conditioning, equipment	2 439	2 520	2 538	2 598
Engines (2)	3 836	2 677	3 836	2 677
Ductwork	1 630	1 152	1 684	1 195
LH_2 tankage	-----	-----	2 809	2 541
LH_2 insulation	-----	-----	532	490
Empty weight	21 190	20 276	25 106	23 128
Crew	800	800	800	800
Fuel	13 865	12 438	6 480	5 861
Payload	3 866	6 947	7 614	10 211
Standard day TOGW	40 000	40 000	40 000	40 000
STO overload condition				
Fuel	17 076	-----	7 920	-----
Payload	10 805	-----	16 404	-----
TOGW	50 150	-----	50 150	-----

[a]Includes fuel tanks for JP airplanes but not for LH_2 airplanes.

TABLE IV. - COD AIRPLANE DATA (2000-N-MI RANGE)

Parameter	JP fuel		LH_2 fuel	
	VTO design	STO design	VTO design	STO design
Wing planform area, ft^2	444	571	444	571
Wing span, ft	42.1	47.7	42.1	47.7
Fuselage diameter, ft	6.9	6.9	10.0	9.9
Fuselage length, ft	48	48	60	59
Minimum drag coefficient, $(C_D)_{min}$	0.0202	0.0175	0.0253	0.0214
Engine cycle (BPR/OPR)	1.5/20	1.5/20	1.5/20	1.5/20
Engine thrust (SLS, ea), lb	14 412	10 185	14 412	10 185
Engine airflow, lb/sec	282	199	282	199
Fuselage volume efficiency	0.478	0.451	0.687	0.655
Fuel fraction	0.347	0.311	0.163	0.147
Flight time, hr	6.2	6.2	6.2	6.2

TABLE V. - DETAILS OF WEIGHT CHANGES FOR INCREASED

TURBINE-INLET TEMPERATURE (TIT)[a]

Parameters	Normal bleed schedule			LH_2 cooled bleed[b]	
	Reference case, TIT = 2650° R (1)	Raise TIT 500° R (2)	Raise TIT 1000° R (3)	Raise TIT 500° R (4)	Raise TIT 1000° R (5)
Engine optimum cycle, BPR/OPR	1.5/20	1.5/25	2.25/25	1.5/25	2.25/25
Percent bleed	2.6	9.4	23.0	4.7	12.5
Cruise SFC	0.29	0.30	0.29	0.28	0.28
Airframe weight, lb[c]	14 751	-615	-311	-1079	-951
Engine weight, lb	2 737	-529	-309	-376	-614
Ductwork weight, lb	1 144	-327	-154	-428	-381
Overall empty weight, lb	18 632	-1471	-774	-1683	-1946
Fuel weight, lb	3 815	-55	-51	-284	-315
Crew weight, lb	800	0	0	0	0
Payload weight, lb	5 700	0	0	0	0
TOGW, lb	28 947	-1526	-825	-1967	-2261

[a] LH_2 fueled COD with 2000 n-mi range; circled numbers refer to points in figure 5.

[b] 50 percent of LH_2 cooling capacity transferred to bleed air, zero cooling apparatus weight.

[c] Reference case weights are absolute, all others are weight increments.

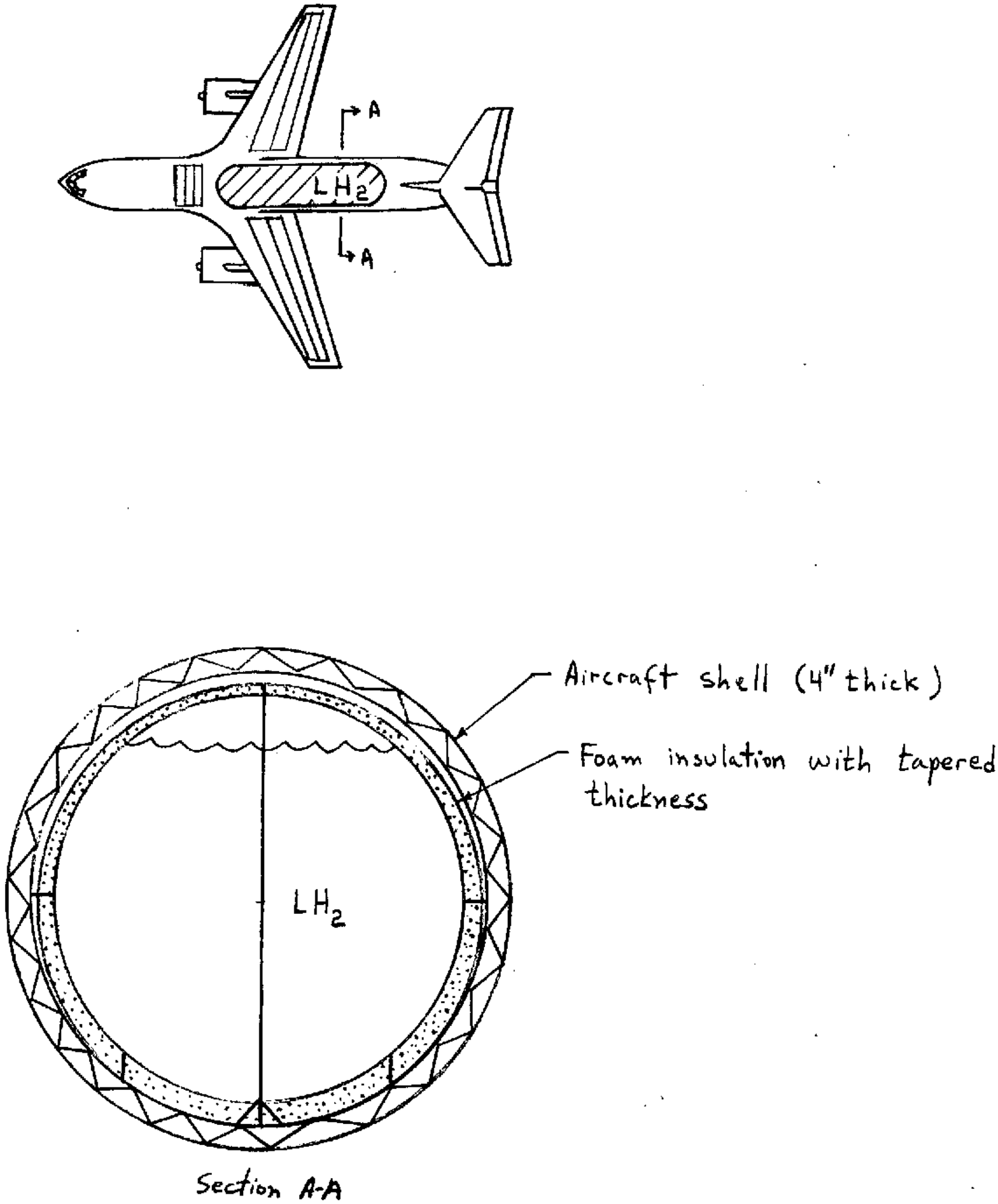
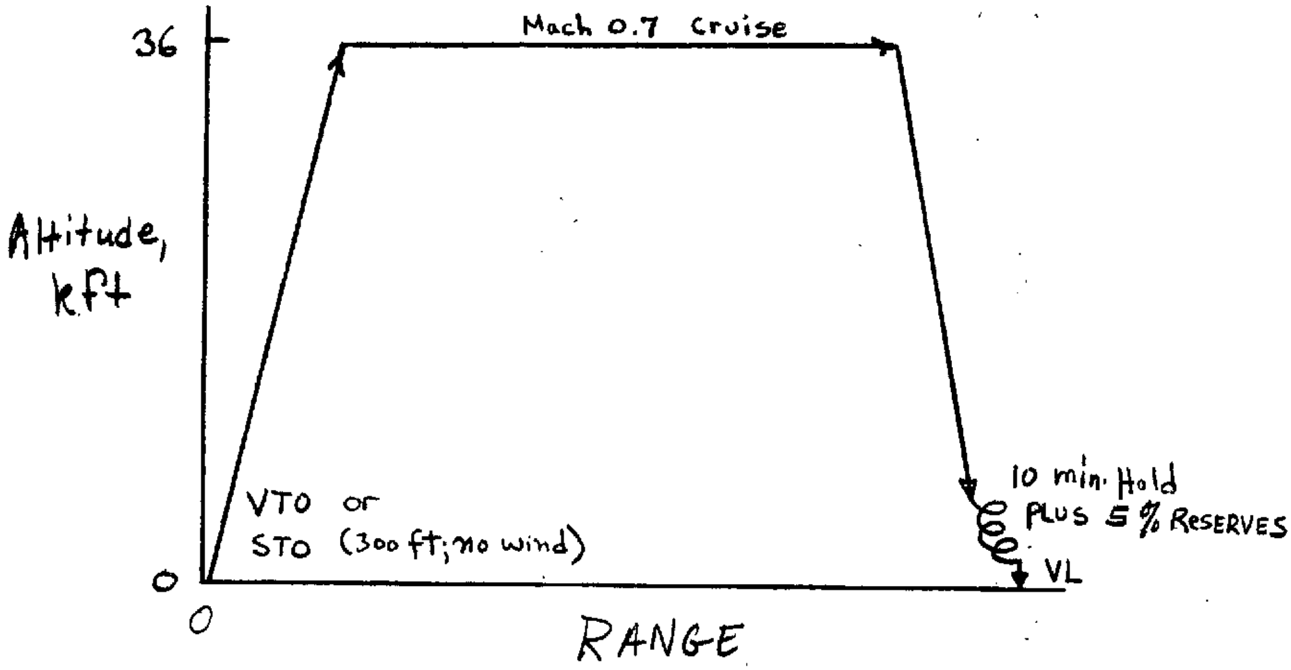
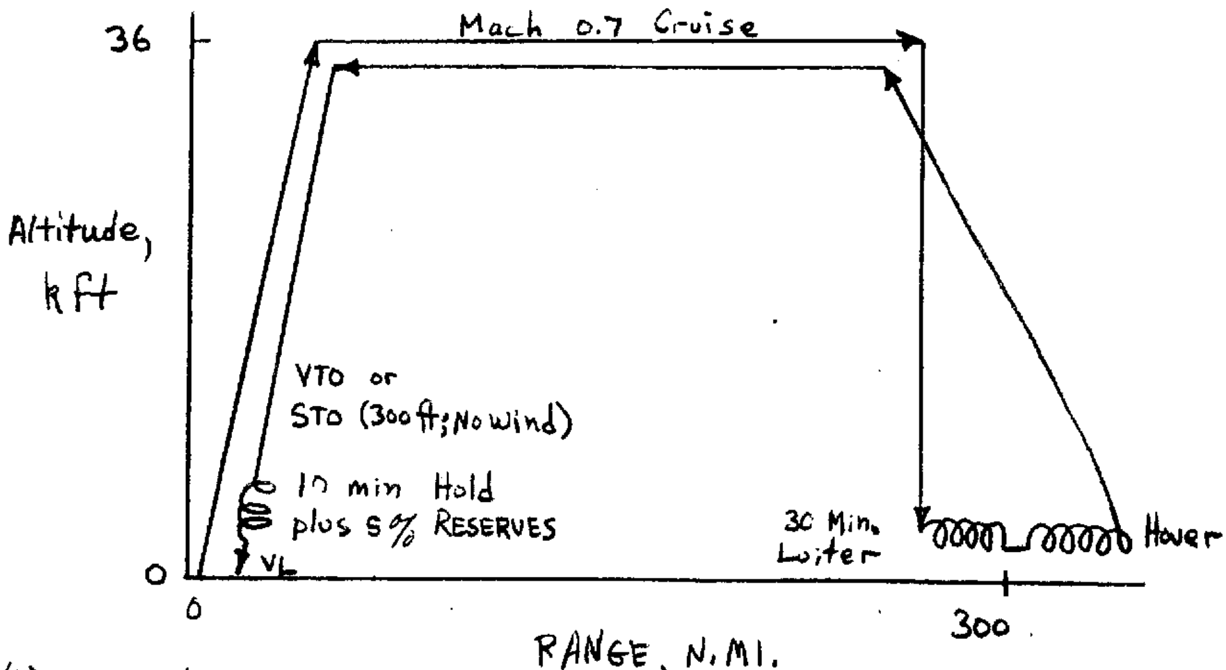


Figure 1.- Aircraft Configuration using LH_2 Fuel.

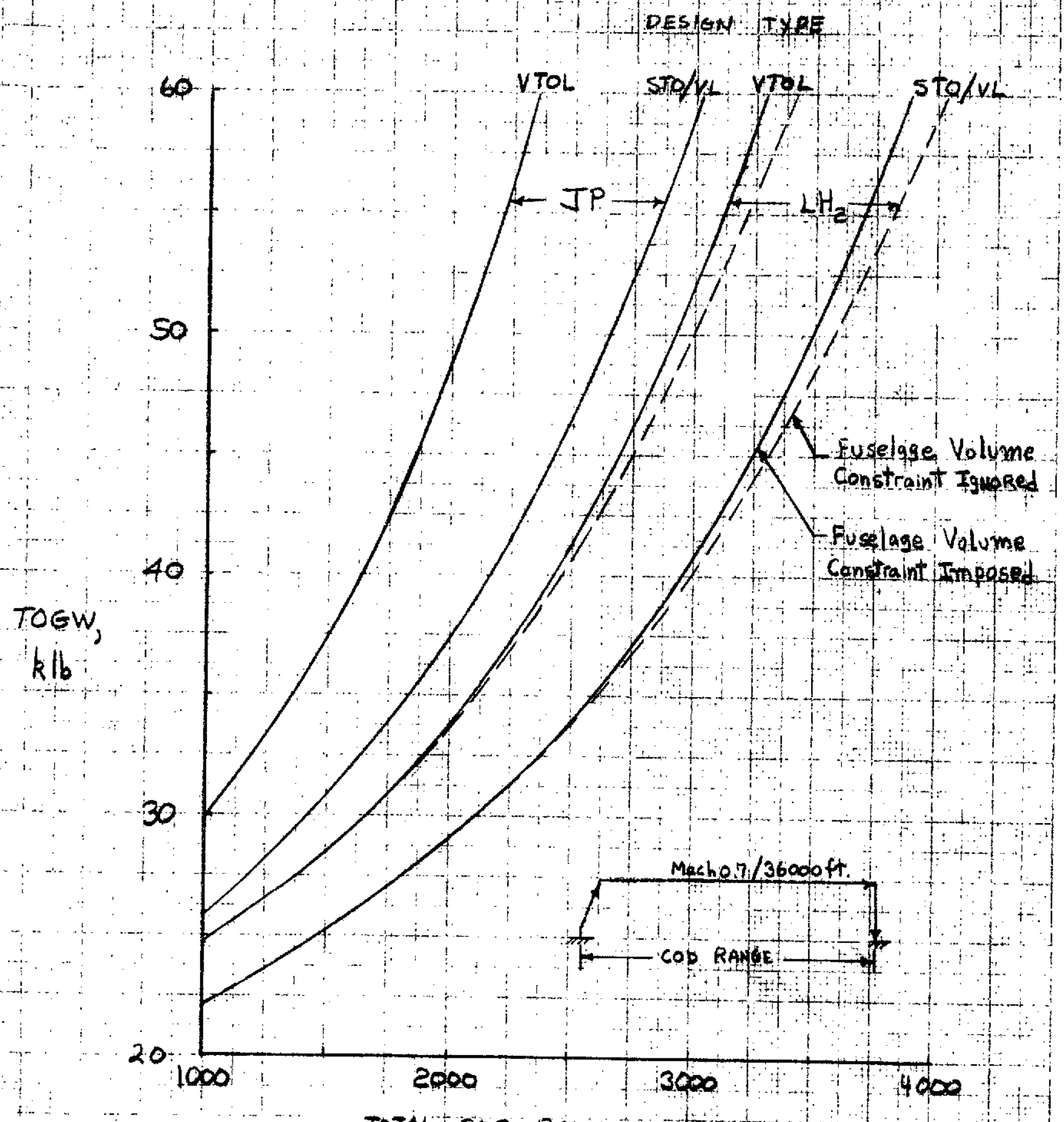


(a) COD.



(b) ASW/SAR.

Figure 2.- Mission PROFILES.



(a) Gross WEIGHT.

FIGURE 3 - EFFECTS OF RANGE, FUEL, AND TAKEOFF MODE ON COD MISSION AIRPLANE SIZE. PAYLOAD, 5700 lb; EJECTOR WING (φ = 1.6); ENGINE BPR/OPR, 1.5/20; STO: 300 FT./ZERO WIND.

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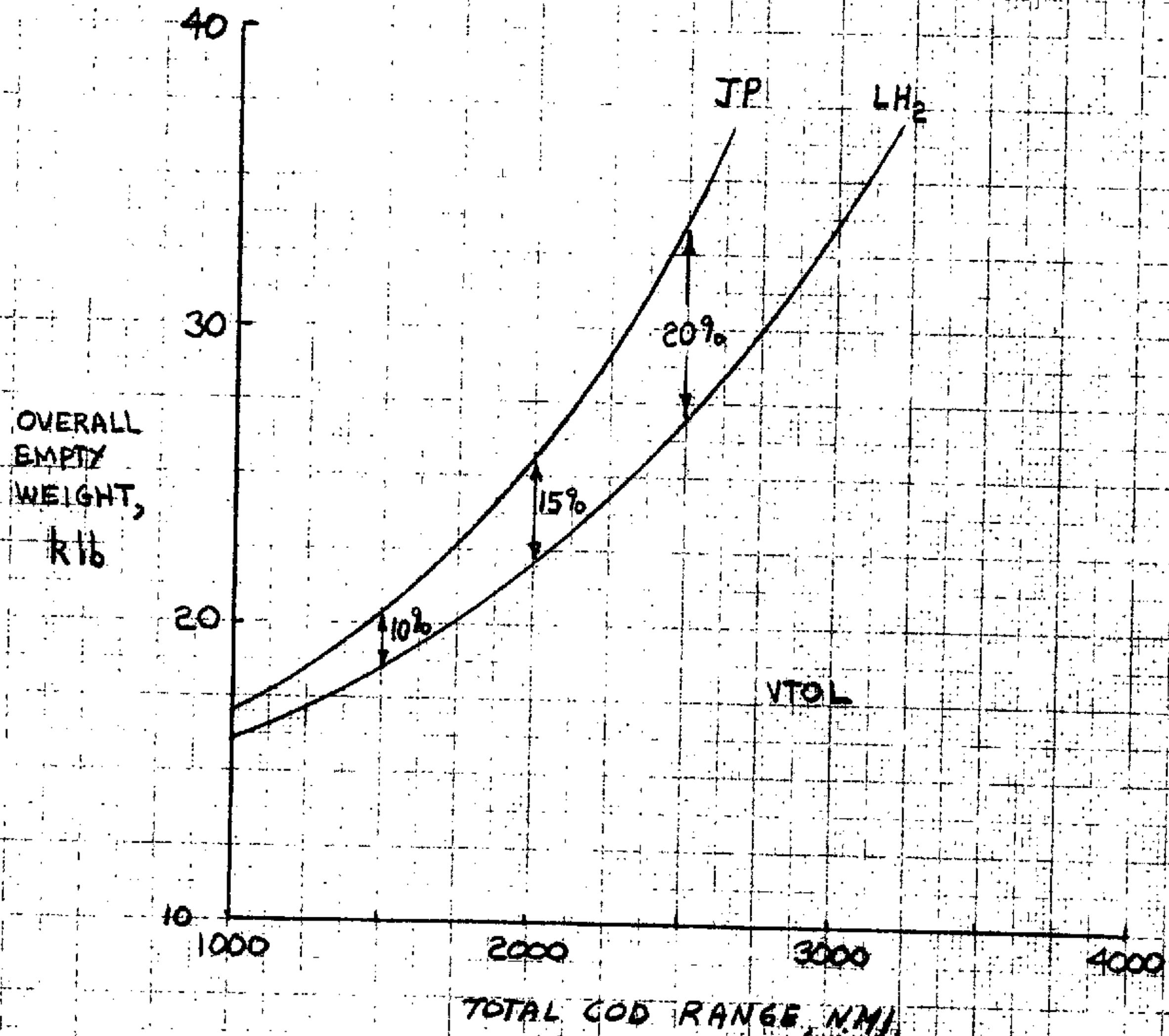

(b) OVERALL EMPTY WEIGHT.

FIGURE 3. - Concluded.

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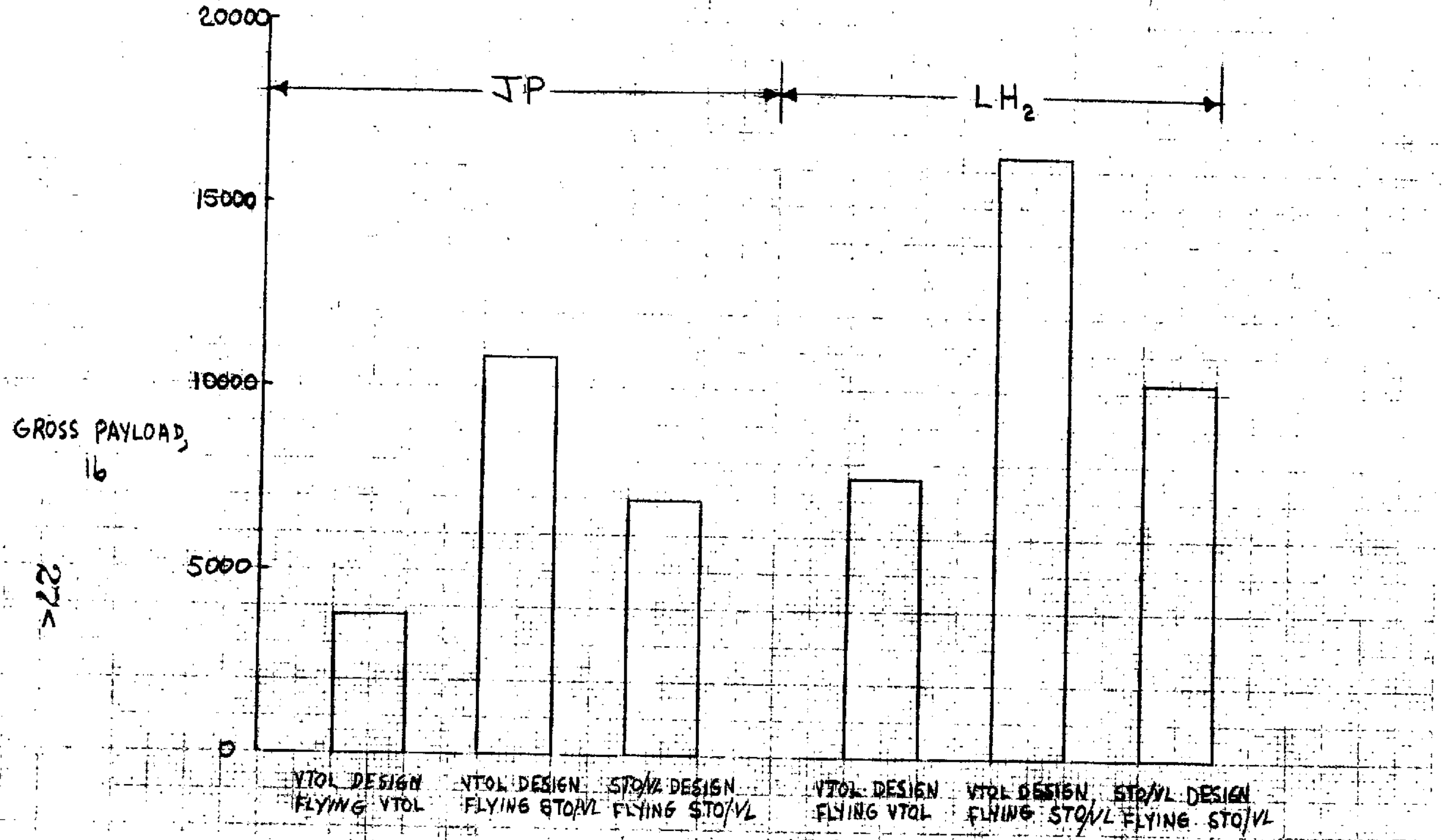

FIGURE 4. - EFFECT OF MODE OF TAKEOFF OPERATION. DESIGN GROSS WEIGHT, 40000 lb; RANGE, 2000 NMI; STO GROUND RUN, 300 FT / ZERO WIND; EJECTOR WING ($\phi = 1.6$).

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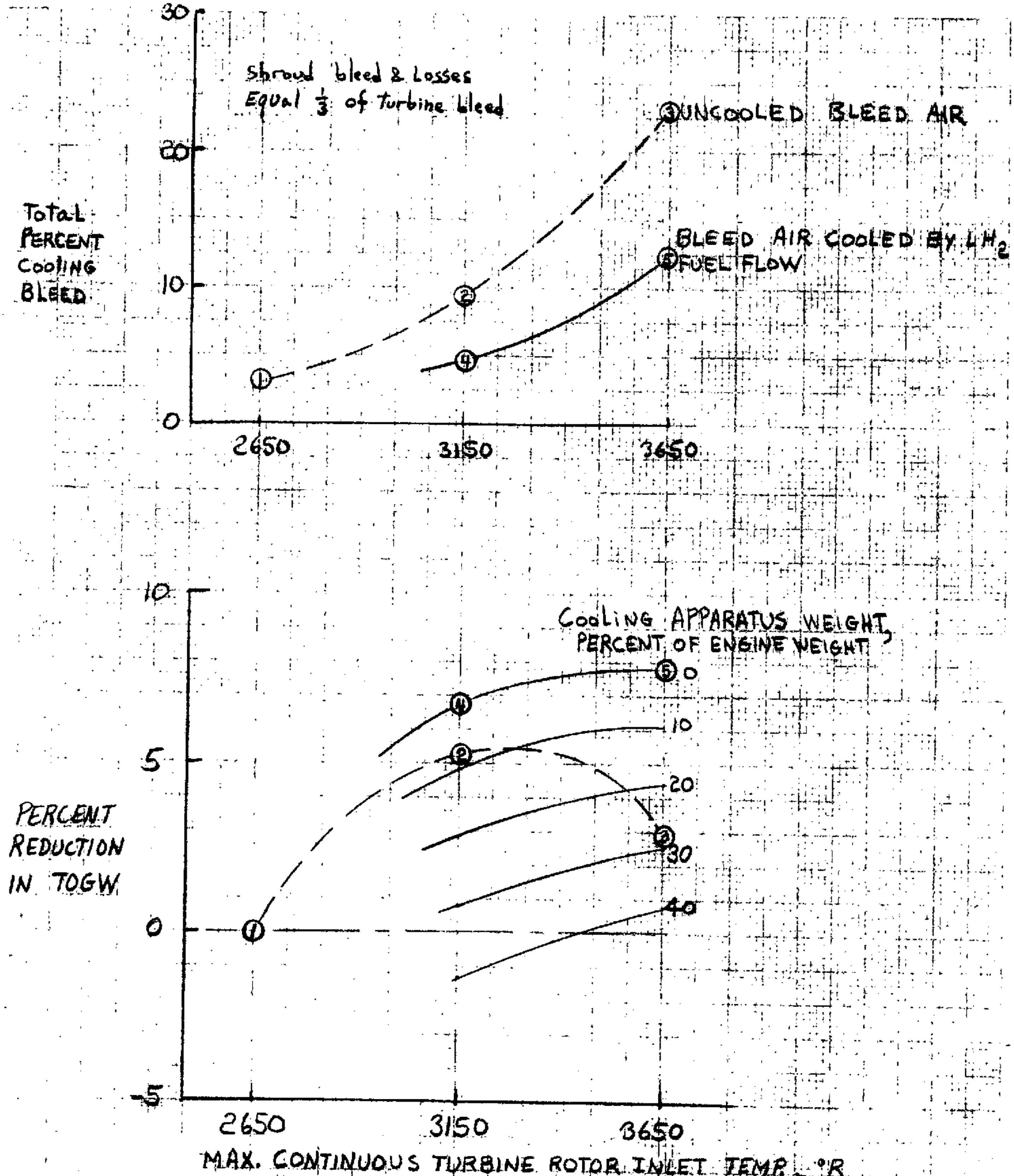

FIGURE 5 .- POTENTIAL IMPROVMENT DUE TO COOLING CAPACITY OF LH_2. RANGE, 2000 N.MI. BLEED AIR IS COOLED BY LH_2 FUEL FLOW ASSUMING 50% EFFECTIVENESS.

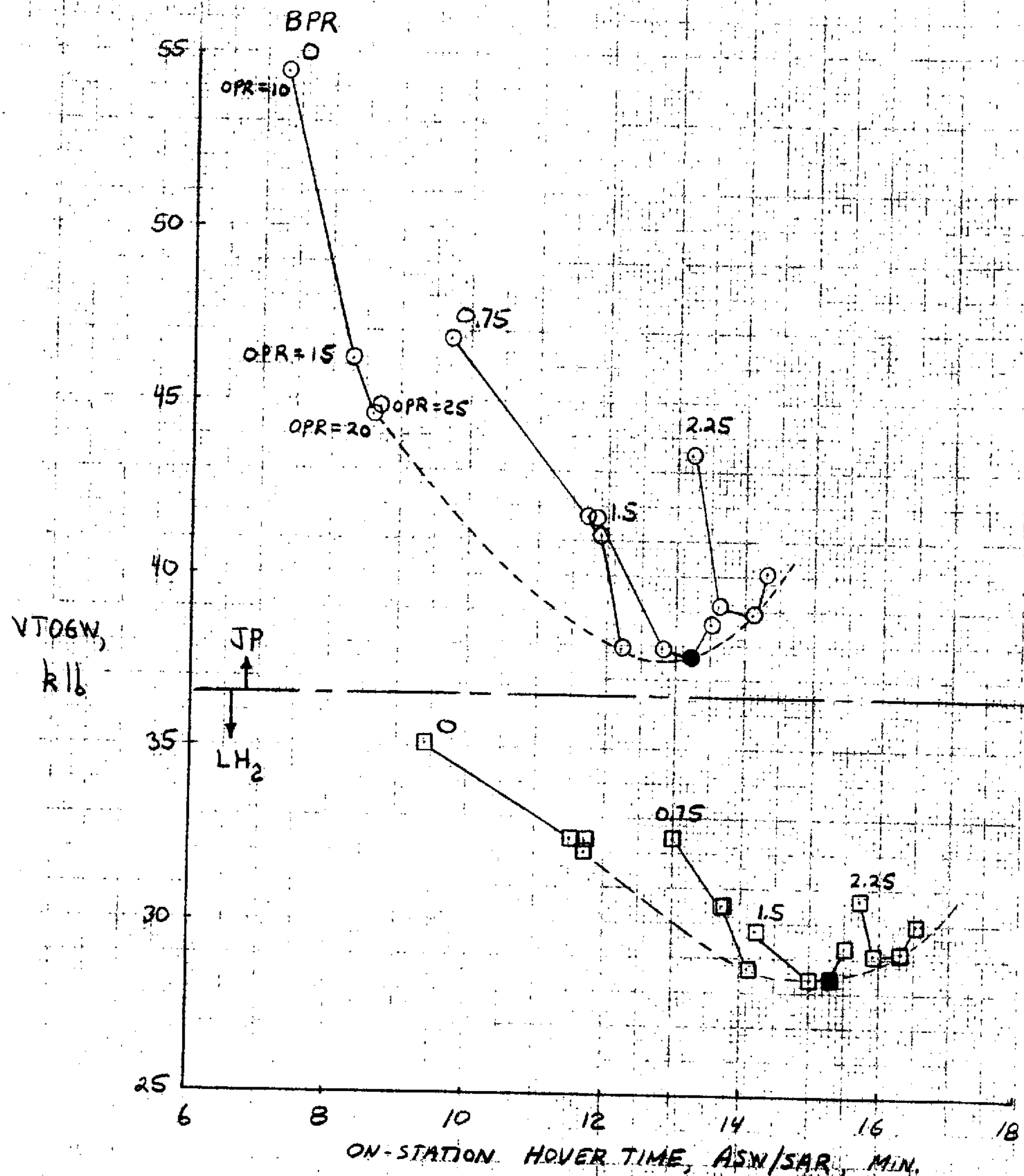

FIGURE 6 .- ENGINE CYCLE SELECTION. DESIGN IS FOR COD MISSION WITH 1500 N.M. RANGE & 5700 lb. PAYLOAD. SECONDARY MISSION IS ASW/SAR WITH 300 N.MI. RADIUS. EJECTOR WING (ϕ=1.6). Baselines denoted by solid symbols.

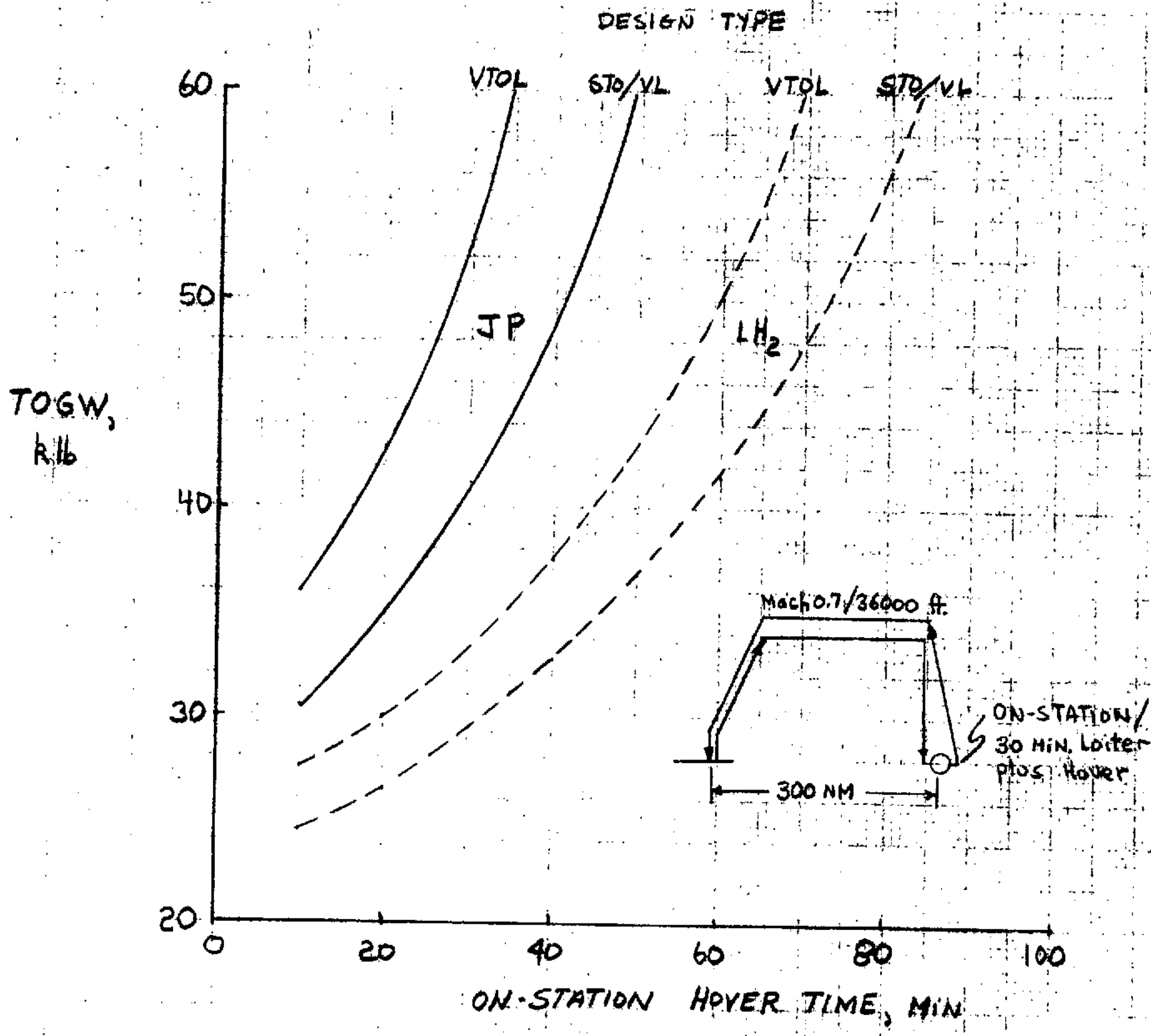

FIGURE 7.- EFFECTS OF HOVER TIME, FUEL, AND TAKEOFF MODE ON ASW MISSION AIRPLANE SIZE. MISSION RADIUS, 300 N. MI.; PAYLOAD, 5700 lb.; ENGINE BPR/OPR, 1.5/20; EJECTOR WING (ϕ = 1.6); STO: 300 ft/ZERO WIND.

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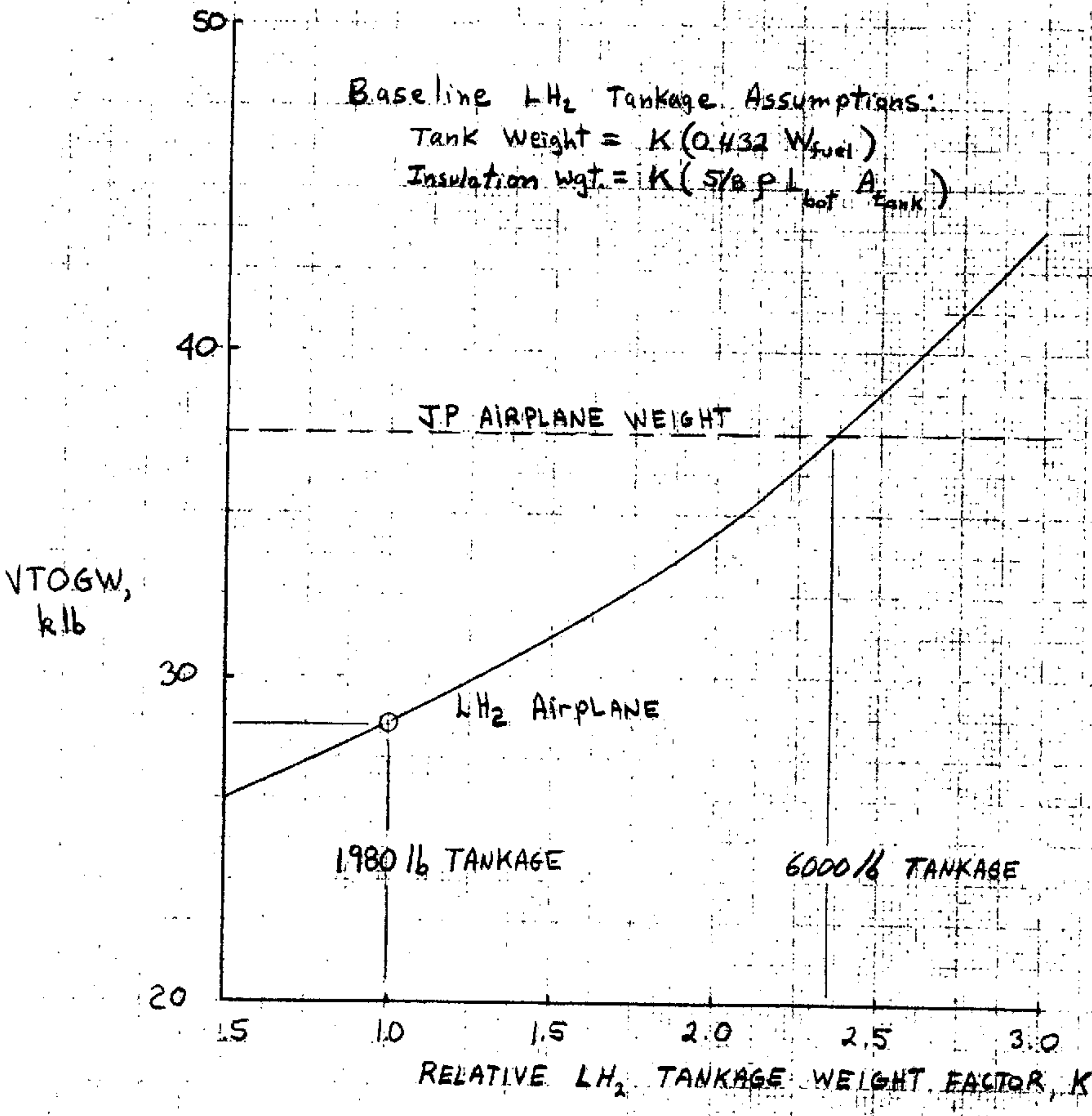

FIGURE 8 . - EFFECT OF CHANGING THE LH_2 TANKAGE WEIGHT ASSUMPTION. COD MISSION. RANGE, 1500 NMI; PAYLOAD, 5700 lb; VTOL; EJECTOR WING (ϕ=1.6); ENGINE BPR/OPR, 1.5/20.

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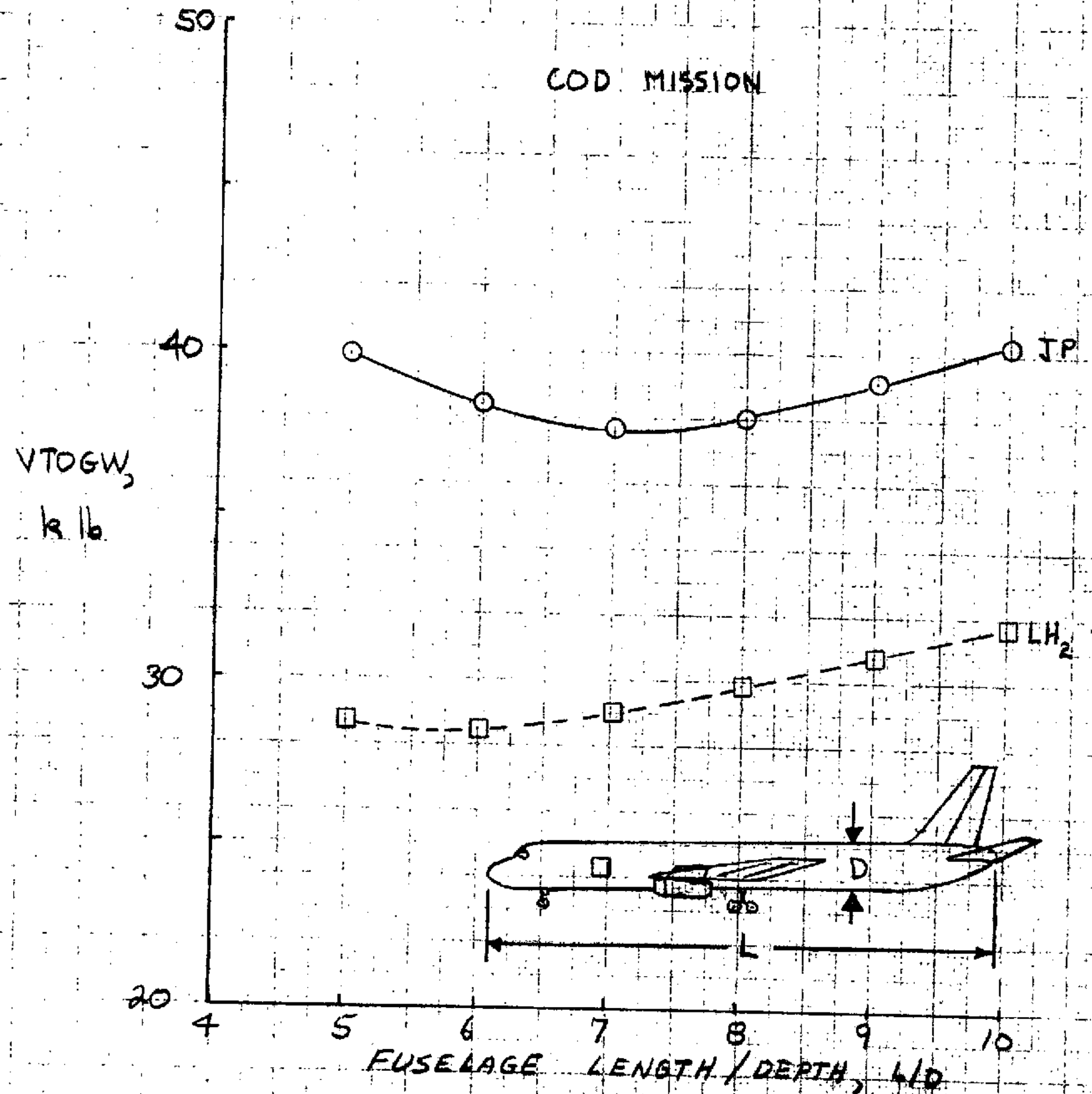

FIGURE 9.- EFFECT OF FUSELAGE LENGTH/DEPTH RATIO. RANGE, 1500 N.MI; PAYLOAD, 5700 lb.; EJECTOR WING ($\phi = 1.6$).

WCS 10-22-73

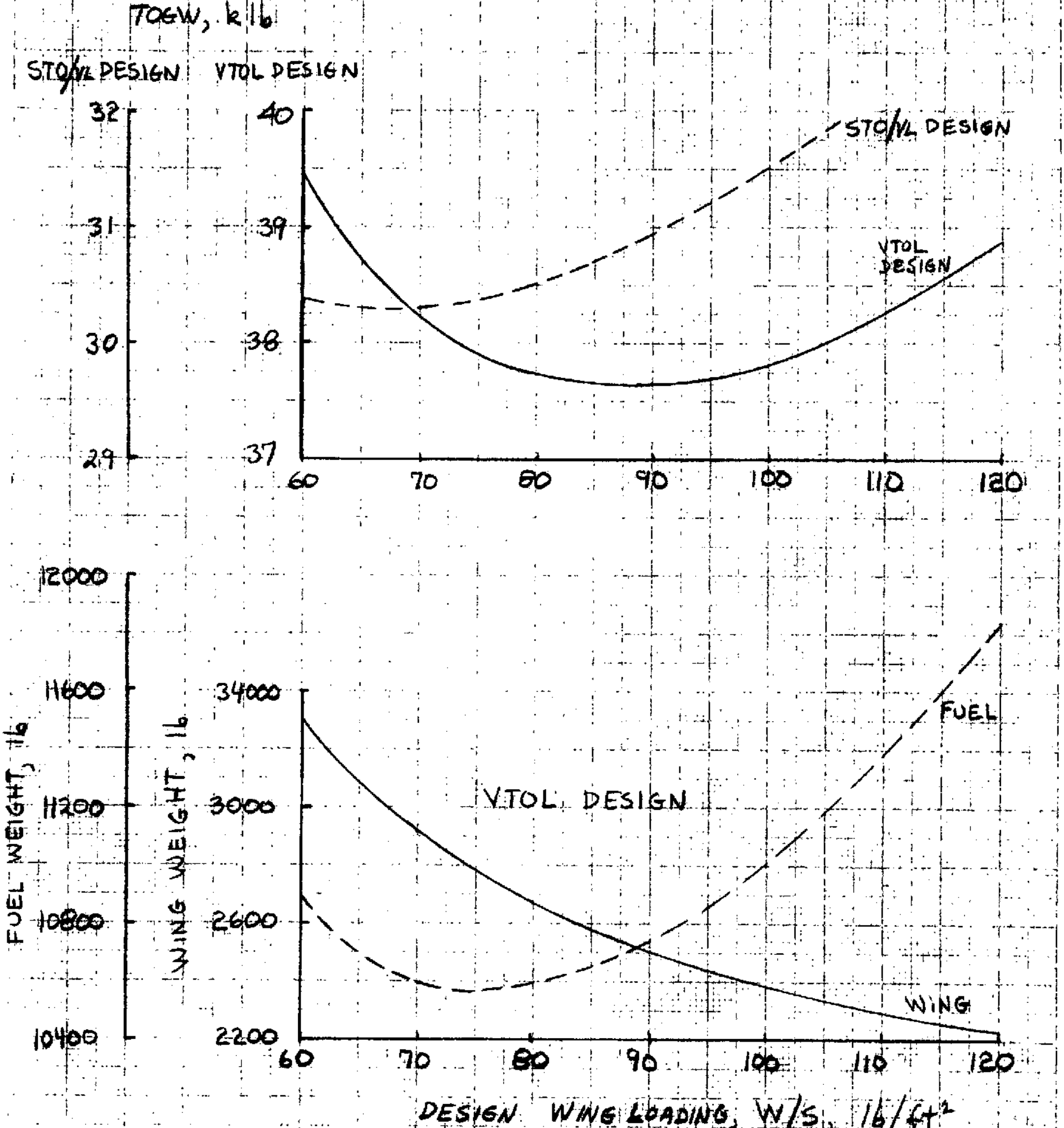

FIGURE 10.- WING LOADING SELECTION FOR COD MISSION. RANGE, 1500 N.MI.; PAYLOAD, 5700 lb.; EJECTOR WING (φ=1.6); JP FUEL.

WCS 11-5-77

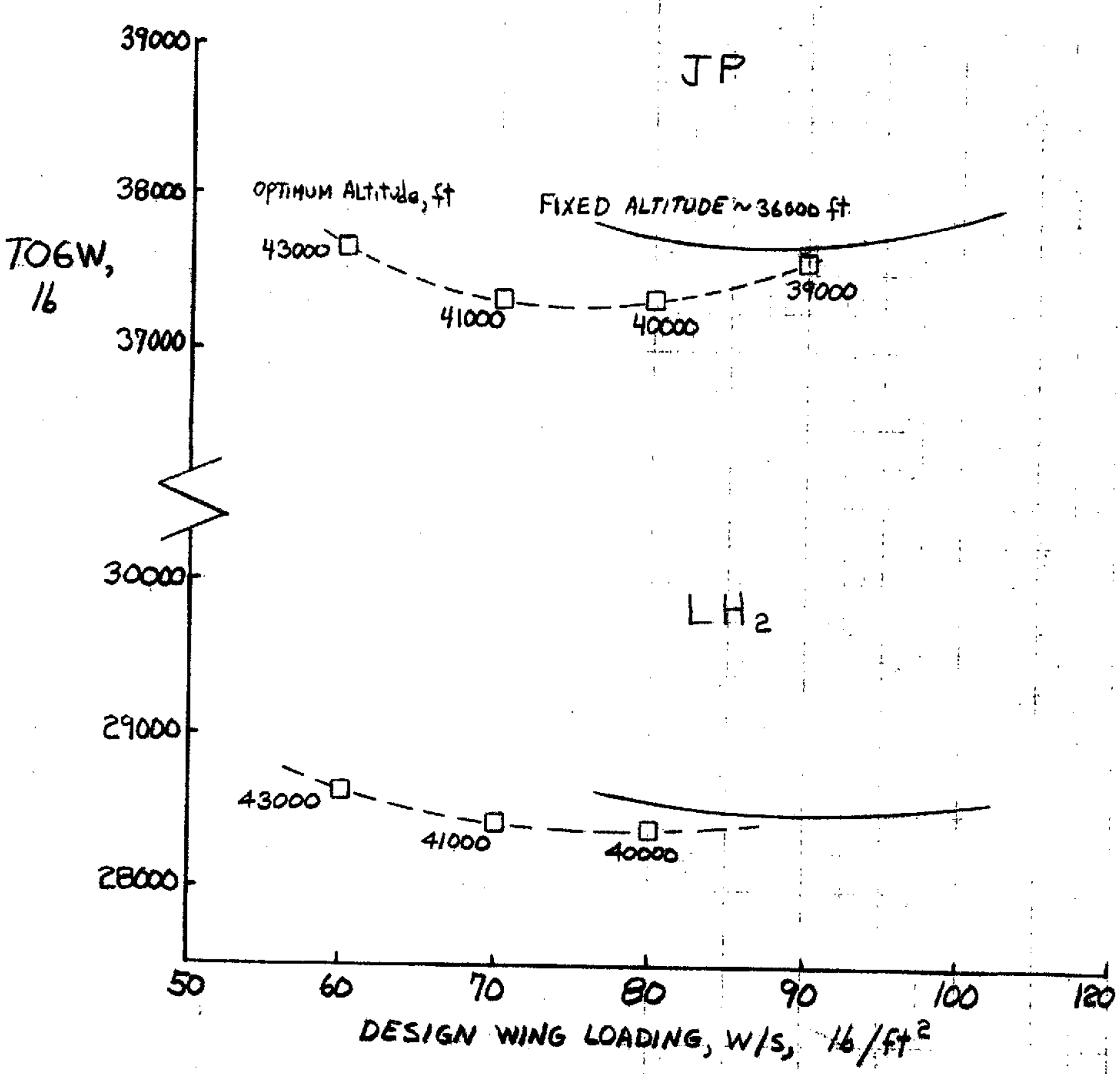

FIGURE 11.- EFFECT OF WING LOADING AND CRUISE ALTITUDE CONSTRAINT. RANGE, 1500 N. MI.; PAYLOAD, 5700 lb.; EJECTOR WING (ϕ = 1.6); VTOL.

WCS 1-17-74