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APPLICATION OF CRYOGENIC PROPULSION TO SPACE MISSIONS

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While I must admit to some personal bias in the matter, it seems fair to claim that no scientific or technological area could be of greater importance to our country's space program than that of propulsion. Throughout the first 4 years of this program, as, indeed, throughout the half century of aviation that preceded it, advancements in flight capability have always been paced and determined by advancements in propulsion.

Further, it is clear that all of our future space missions stand at the very frontiers of our technology. What can be done in space and when it can be done is strongly dependent on how fast new knowledge can be accumulated and how effectively this knowledge can be translated into hardware that works. The press of time schedules and the urgency with which we reach toward ever more difficult missions has also imposed a high degree of overlapping, and even concurrency, among applied research on new problems, on hardware development, and on qualification of this hardware for flight. It is therefore both fitting and necessary to survey future mission requirements periodically, to define optimum or most useful systems for various types of missions, and to isolate the critical problem areas requiring timely effort.

Propulsion systems currently under development are aimed at providing larger payload weights in Earth orbit, sending scientific probes to the nearby planets, and sending a brief expedition of men to the Moon and return. Building upon this base, it is pertinent to examine the types of upper stages that might be added to our large boosters to carry out more energetic and sophisticated scientific missions and to extend the payload capabilities of manned lunar missions.

Possible scientific missions of importance in the near future and that are beyond our present capabilities are those of solar probes and planetary orbiters. Flights out of the plane of the ecliptic might also be added. Both the solar probe and the flight out of the ecliptic are characterized by very high total-energy requirements, while a space propulsion system for a planetary orbiter must be capable of living and operating in the unique conditions of a space environment for extended periods of time. Meteoroids, ionizing radiation, extreme thermal loads, and zero gravity are among these unique conditions of a space environment.

A brief survey of the energy requirements for such scientific missions is presented in figure 1. Total mission velocity increment is plotted against the perihelion distance of a solar probe with the two horizontal dashed lines indicating the requirements of a planetary probe and a planetary orbiter. Gravity and drag losses are included in these numbers. The energy requirements of a solar probe are, of course, strongly dependent upon how closely one wishes to approach the sun, with primary attention here being directed to a distance of around 0.2 astronomical unit. Such a distance is in the region of large scientific interest and near the limits imposed by thermal loads.

At this point, a total mission velocity increment of 59,000 feet per second is indicated, which would, incidentally, exceed the energy requirements for escape from the solar system; this total energy would also be capable of propelling a probe at an angle of 25° out of the ecliptic. The energy requirements of the planetary probes are considerably less, with the orbiter requiring a velocity increment of about 7400 feet per second above the 42,000 feet per second necessary for a 220-day trip to Mars. An upper stage of possible interest in a solar probe is that of a kick stage atop an Atlas Centaur. For this event, the stage velocity increment is about 23,000 feet per second. We might therefore consider one stage of this velocity increment that could be mated to Atlas Centaur and another stage of about one-third this velocity increment, which must be capable of "living" in the space environment.

The high velocity increment of the solar probe and the absence of the requirement to live in the space environment for extended periods of time clearly suggests the use of high-energy propellants. What might such stages look like? A schematic inboard profile of two such stages is presented in figure 2. To the left is shown a pumped hydrogen-oxygen stage and to the right, a pressurized hydrogen-fluorine stage. Principal performance assumptions are a chamber pressure of 300 pounds per square inch and an expansion ratio of 60 for the hydrogen-oxygen stage, which can produce an impulse of 430 seconds. For the pressurized hydrogen-fluorine stage, the chamber pressure is assumed to be 60 pounds per square inch and the expansion ratio is reduced to 40 for obvious reasons. The resulting impulse of this hydrogen-fluorine stage is about 442 seconds. For both configurations, nonintegrated, spherical tanks are assumed. Although toroidal or clustered tankage arrangements would permit reductions in stage length, the single spherical tanks provide important reductions in structural weight and avoid difficult sumping problems.

Principal configuration differences between these two stages are the size of the fuel tank and the larger engine of the low-pressure hydrogen-fluorine system. Because the fluorine engine operates at a mixture ratio of 11.5 as compared with 5.0 for the hydrogen-oxygen engine, only about half as much hydrogen fuel volume is required. Interestingly enough, the dry weight of both stages turns out to be about the same, with the heavier pressurization system, tanks, and engine of the hydrogen-fluorine system being just about offset by the heavier structure, greater insulation, and, additional fuel necessary to replace boiloff losses of the hydrogen-oxygen stage. If one assumes aluminum fuel and oxidant tanks, a Hylas-type pressurization system, and a 25-percent weight contingency, a dry weight of slightly less than 500 pounds is obtained. The hydrogen-oxygen stage requires a little more propellant because of the slightly lower impulse, but both type of stages are in the 4200-pound-gross-weight class and it is possible to specify an engine thrust of a little over 2000 pounds for both systems.

The payload that may be placed at various perihelion distances is shown for four different stages in figure 3: the pressurized hydrogen-fluorine and pumped hydrogen-oxygen stages previously illustrated and, in addition, a pumped hydrogen-fluorine and a pressurized Earth storable stage. Clearly, the storable propellant stage, with an impulse of about 300 seconds, is not suitable for such an energetic mission as the solar probe. The pumped hydrogen-fluorine stage, with an impulse of 455 seconds and a somewhat better mass fraction, is better than either of the other two stages, but not enough better, in my opinion, to justify the very considerable technological problems of pumping fluorine. We may thus reduce our consideration for this mission to either the pumped hydrogen-oxygen stage or the mechanically simpler but technologically more advanced pressurized hydrogen-fluorine stage. In either case, a payload of approximately 400 pounds is possible on an Atlas-Centaur booster, certainly in the region of practical scientific interest.

A selection between these two stages must therefore be made on some basis other than performance if this is the only mission considered. Let us therefore consider the suitability of these stages for the planetary orbiter mission, in which the energy requirements are much less, but in which survivability in the space environment is more important. A basic concept here would be to try to avoid the development of different, optimum stages for each mission, but to modify the stages appropriately as required. Accordingly, some 2 inches of foil insulation was added to the propellant tanks and a meteoroid bumper was added outside the structural frames and inside the inner stage. It was assumed that the meteoroid bumper would consist of a honeycomb structure with a bumper factor of 5 and that 90 percent of its weight would be jettisoned prior to insertion into the planetary orbit.

It is possible to place either of these stages into the planetary transfer trajectory by a Saturn C-1B Centaur booster combination. Although the smaller fuel tank of the hydrogen-fluorine system reduces the total weight chargeable to insulation, boiloff, and meteoroid protection to about half that of the hydrogen-oxygen stage, the fact that the meteoroid shield is jettisoned prior to the orbital maneuver reduces the effect of this weight difference on the payload. As a result, as shown in figure 4, the total payload that is placed into orbit is only about 10 percent greater for the hydrogen-fluorine system than for the pumped hydrogen-oxygen system.

Also included in this figure is the possible performance of a pressurized Earth-storable stage designed specifically for this mission. The quite low velocity increment of this mission clearly does not capture the basic advantage of the high-energy propellants, with the result that the simpler Earth-storable system is admittedly competitive. It would, of course, have to be specifically designed for this mission and the important advantage of developing essentially one stage for both this and the more energetic missions would be lost.

All of these results are, as indicated, for a Whipple meteoroid flux density and a survival probability of 90 percent. It is possibly of interest, then, to look at the effects of varying this meteoroid protection criterion on the payload characteristics of the two high-energy stages; this is done in figure 5, where the survivability criterion is increased to 99 percent. Although this more conservative assumption penalizes the hydrogen-oxygen system more than it does the denser fluorine stage, the payload results are still within 20 percent of each other. Also included in this figure is the performance of a much higher density and moderate impulse system, here typified by hydrazine and fluorine with an impulse of about 390 seconds. For this particular mission, density and impulse pay off on nearly equal terms with the result that the two fluorine oxidized systems have about equal payload capabilities. Of course, the lower impulse of the hydrazine-fluorine system would place it at a significant disadvantage for the more energetic missions previously mentioned.

A choice among these propellant combinations, at least for this range of missions, therefore depends upon how one evaluates the importance of a 10 to 20 percent performance gain and compares the mechanically and operationally simpler pressurized fluorine stage with the more readily available and growing technology available for hydrogen-oxygen systems. While many years of research on the hydrogen-fluorine system have clearly established its fine performance, cooling, ignition, and stability characteristics, many new problems and hazards related to toxicity, logistics, and materials compatibility are introduced. On the other hand, if a hydrogen-fluorine technology is deemed desirable for other future missions that are not presently defined, this seems a good place to start - that is, with a small pressurized system - rather than to initiate a fluorine application for a manned space flight system, for larger stages, or for a more difficult pumped system.

Of possible interest to other system or mission applications is the use of fluorine as an additive to the oxygen of a hydrogen-oxygen system. Such an additive, or mixed oxidizer, may prove beneficial in situations in which a performance gain of a few percent is critical; fluorine additions to oxygen may also be expected to improve combustion stability, particularly in throttleable systems, will improve propellant bulk density, and will be shown later herein to render hydrogen and oxygen hypergolic. Further work to define both the difficulties and the performance and operational benefits of fluorine additions to oxygen seems justified.

Let us now turn our attention to the application of cryogenic propellants to some possible near-future manned missions. A possible area of interest here is the growth of mission capability that hydrogen-oxygen propellants may afford to the present lunar orbit rendezvous type of Apollo mission. With the selection of this mission mode being based largely on the judgment that the advantages of a small landing vehicle and the use of Earth-storable propellants on the moon will outweigh the operational complexities of lunar rendezvous, a most appropriate method of increasing this mission capability is to substitute cryogenic propellants for the present storable propellant system in the service module.

A general idea of what such a stage might look like is presented in figure 6. Two arrangements are shown, one for a pumped system and one for a pressurized system. As far as outward appearances are concerned, the major difference between the two systems is the much larger engine required for the low-pressure, pressurized system. The factors influencing the choice of pressure level will be illustrated later; it will suffice to note here that an expansion ratio of 40 is shown for the pressurized system and 60 for the pumped system. In both systems an engine thrust level of about 15,000 pounds was chosen. Although the overall length of the pressurized system could certainly be reduced by the use of toroidal or clustered tanks grouped around the large engine, the arrangement shown was chosen for preliminary performance analyses because of structural weight advantages and much simpler sumping problems. The fuel is thus contained in a sphere approximately 13 feet in diameter, which fits easily within the chosen envelope. The loads are carried through an external honeycomb structure, which was found to be no heavier than an arrangement in which the loads are carried through the tank; this external structure also provides adequate meteoroid protection for the assumed 10-day exposure time.

The importance of the pressure level assumed for the pressurized system on its payload capabilities is shown in figure 7, with the payload being considered as the useful weight placed in lunar orbit. Also included is the payload capability of the pumped hydrogen-oxygen system. Both systems are relative to the value computed on a comparable base for a pressurized Earth-storable propellant service module. Because the engine weight savings at higher pressures are almost negligible, the payload varies almost linearly with chamber pressure through its effects on pressurization system and tank weights. It is therefore necessary to use rather low chamber pressures for the pressurized system if it is to compete on a performance basis with a pumped system. A suitable choice might be a chamber pressure of 60 pounds per square inch (this system does not have to be throttleable) and a tank pressure of about 110 pounds per square inch, which should be adequate for a regeneratively cooled engine. The pumped system is superior in payload capability to the pressurized system even at the lowest chamber pressures because of its slightly higher impulse and most particularly because of the much smaller weight of pressurization equipment and propellant tanks. It should also be noted that aluminum tanks were assumed because of their better known characteristics at liquid-hydrogen temperatures and the pressurization systems assumed were a topping system for the pumped engine and a Hylas system for the pressurized case.

With either system, the use of cryogenic propellants in this service module may provide a payload gain of 20 to 25 percent. One is, perhaps, strongly tempted to prefer the pumped engine system because of its somewhat superior performance, shorter overall length, and the well-developed status of the present RL-10 engine. However, a rapid start capability of this system is also required for some abort maneuvers that may prove difficult to achieve with a pumped system.

An additional factor of some importance, particularly to a pressurized system, is whether the combustion chamber should be regeneratively or ablatively cooled. Regeneratively cooled chambers may be considered superior to ablatively cooled chambers, particularly for hydrogen-oxygen propellants, because of their extensive technological background, excellent cooling characteristics, and ability to properly condition the propellant for stable combustion. Ablatively cooled chambers, on the other hand, have several advantages in terms of their simplicity of construction, ruggedness, reduced pressure requirements, and possibly reduced startup and shutdown transients. Much work remains to be done, however, to achieve a high performance level and long burning duration in the same chamber. Because a high combustion efficiency implies a high bulk gas temperature, a high impulse level is basically incompatible with a long firing duration. This situation is illustrated in figure 8, in which the characteristic velocity efficiency is plotted against firing duration. The curve of this figure represents the result of an analytical model of the ablative process. Although this calculation is idealized in the sense that it does not account for uneven ablation, it is clear that a long firing duration, such as the 800 seconds indicated for our present application, can be obtained only at fairly low chamber pressures and at a relatively low level of performance. The shaded area to the left of this curve indicates the general region of current experience. The upper dashed curve illustrates the region of performance and duration reported in the unclassified literature for small rockets utilizing tungsten throat inserts. These results, though promising, are for very small engines and must be considered tentative. Obviously, much work remains to be done on special throat inserts, in the search for better materials, on new fabrication processes, and in the accumulation of adequate statistical data before ablative chambers can be effectively utilized in long-burning space propulsion systems.

Space propulsion systems must operate not only for extended firing durations but also over several cycles of operation between which they are exposed to the space environment. An area of possible concern with ablating materials is therefore the tendency for the resins in the ablative material to decompose and vaporize at low pressure while still hot following a firing period. The resins in many proposed ablative materials start to decompose at 300° or 400° F and could possibly volatilize at high rates until cooled below their decomposition temperatures. Some decomposition gases from the ablative resins could also be trapped beneath the char layer and cause additional spalling of material when exposed to vacuum.

In order to provide some preliminary answers to these questions, similar ablative rocket engines composed of phenolic high-purity silical fibers were alternatively exposed after firing either to a low-pressure environment of about 10 millimeters of mercury or to atmospheric pressure. The conditions of the tests were also such that the ablative material cooled to approximately 1700° F prior to exposure to the low-pressure environment. Although this temperature is well above the decomposition temperature of the ablator, it is not certain that similar results would be obtained under more severe, and more realistic, conditions of temperature and pressure. The results, plotted in figure 9 in terms of weight loss against accumulated running time, are presented in this preliminary form to indicate current status of the work and to encourage further investigation. As noted in figure 9, very similar results were obtained for both conditions, which indicates that there was no significant effect on the ablative material due to the low-pressure cooling. X-ray photographs of the char depths and measurements of the extent of eroded material also were very similar for the two conditions of exposure.

An essential requirement of any space propulsion system is reliable ignition. In this respect, the hydrogen-oxygen propellant combination is at a disadvantage compared with the hypergolic Earth-storable or fluorine combinations. Considerable effort has therefore been devoted either to achieving reliable ignition systems or to finding additives or catalysts that may render the hydrogen-oxygen combination hypergolic. With regard to the latter method, some evidence exists that palladium alumina may be a suitable catalyst, but present information is too fragmentary to permit useful conclusions. Additives so far investigated relate to the addition of either ozone bifluoride or fluorine to the oxygen. To date, it has been found that the addition of ozone bifluoride up to nearly its solubility limit of 0.1 percent can provide hypergolic ignition, but with delay times that are too erratic to be considered satisfactory. The addition of sufficient fluorine to the oxygen can, of course, provide reliable hypergolic ignition, but just what is a sufficient quantity is, of course, the central question.

Some test results that define this quantity of fluorine required for hypergolic ignition are presented in figure 10. Ignition delay time is plotted against the percentage of fluorine in the oxygen for two different injectors and two hydrogen temperatures. With a swirl cup injector and 500° R hydrogen, short and reproducible ignition delay times were possible only for fluorine additions of 30 percent or more. With colder hydrogen or with a showerhead injector, significantly larger proportions of fluorine were required. In any event, while fluorine additives will render the hydrogen-oxygen system hypergolic, the amounts of fluorine required are really too large to justify the term "additive."

A last and most important topic that I would like to mention briefly is reliability. While much attention has been given to this subject over the past few years, it must be recognized at the outset that our future reliability requirements, particularly for manned space flight, exceed anything demonstrated so far or obtainable by our current development processes. In a very real sense, this matter of reliability is of overriding importance for space-mission equipment; further because only a few models of each type are built and used, the usual statistical approach to reliability prediction becomes meaningless.

Reliability goals are, of course, established early in the planning of any program with representative values being of the order of 90 percent for mission success and perhaps 99 percent for crew survival - and thus starts the numbers game. If this goal of 90 percent reliability is to be divided among the several stages of a complex system, a required value of perhaps 98 percent for each stage is obtained. Dividing this stage reliability among the thousands of components produces a required component reliability of a decimal point followed by a long string of nines. And yet, what meaning is there to the statement that a pump, a valve, or a thrust chamber must have a reliability of, for example, 0.999? It would have to be tested a thousand times to demonstrate this reliability and then it would have been done only once.

Although these reliability analyses are of importance in identifying the most critical features of a system, in laying out the initial design, and in establishing proper redundancy networks, other elements of reliability must now be given increased emphasis. One such element is dramatically illustrated by the public record of NASA experience in the last 3 years of launching satellites and space probes. A gross summary of this reliability record, divided into two broad categories of spacecraft and launch vehicles, is presented in figure 11. The distinctly superior record of the payloads as compared with the launch vehicles is at once apparent.

This difference in reliability between the payloads and their launch vehicles cannot be attributed to a greater complexity of the launch vehicles. In fact, the payloads can be shown to be from 1 to 2 orders of magnitude more complex than a launch vehicle on either an "active element" or a "functions performed" basis. An additional point of interest is that most of the vehicles chosen to carry expensive payloads have had a relatively long history of prior flight testing, whereas the payloads themselves were, in many cases, being exposed to their first flight.

In spite of this long flight history of the launch vehicles, the overall reliability improvement over the 4-year period has been small and appears to be approaching an asymptotic level of about 70 percent. Effective reliability improvement actions require an intimate knowledge of how and why failures occur, as well as a knowledge of performance parameter variation on successful trials. Here, the payloads clearly have had the advantage. Since they are of relatively small size and weight, they could be subjected to extensive qualification and development testing under simulated launch and space conditions. Each payload is committed to flight only after the most extensive series of shock and vibrating testing, of thermal cycling, of exposure to extreme vacuum, of pressurization tests, and of acceleration loads on the entire system.

We must now do the same for all of our future space propulsion systems. These complete space propulsion systems must be subjected to the most thorough and detailed development evaluation, operational checkout, and qualification testing under space conditions that can be devised. Although such a process and the major testing facilities that are required may seem expensive, it is, in truth, the most economical process. With the cost of each launch in the scores of millions of dollars, it is simply not possible to spend effectively enough money on the ground to offset the cost of a few launch failures. Also, beyond the purely economical argument is the overriding need for flight success to achieve the program continuity and the compressed time schedules associated with our nation's present commitment to space exploration.

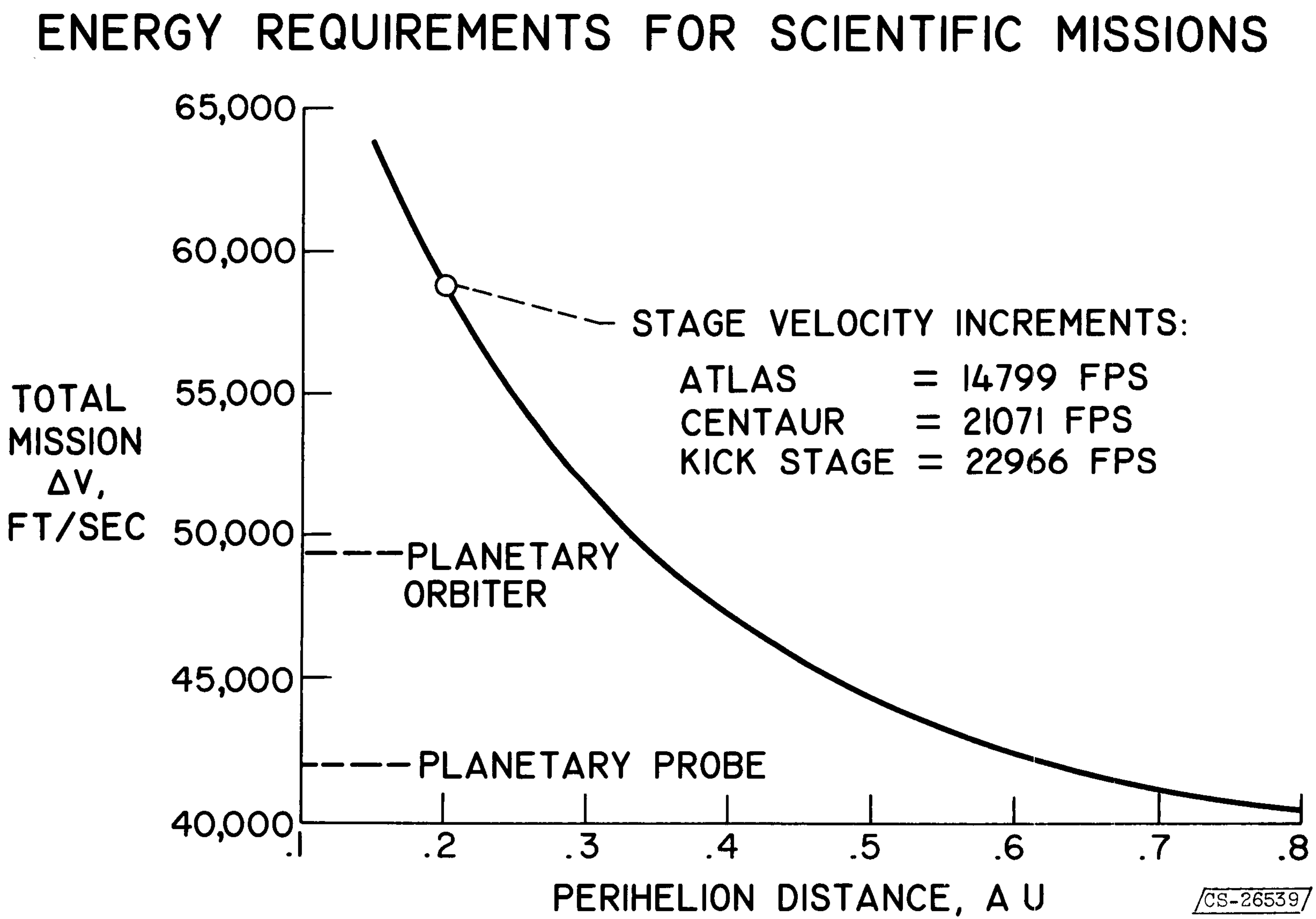
ENERGY REQUIREMENTS FOR SCIENTIFIC MISSIONS
TOTAL MISSION ΔV, FT/SEC
65,000
60,000
55,000
50,000
45,000
40,000
STAGE VELOCITY INCREMENTS:
ATLAS = 14799 FPS
CENTAUR = 21071 FPS
KICK STAGE = 22966 FPS
PLANETARY ORBITER
PLANETARY PROBE
.1
.2
.3
.4
.5
.6
.7
.8
PERIHELION DISTANCE, A U
CS-26539

Figure 1

CRYOGENIC UPPER STAGES

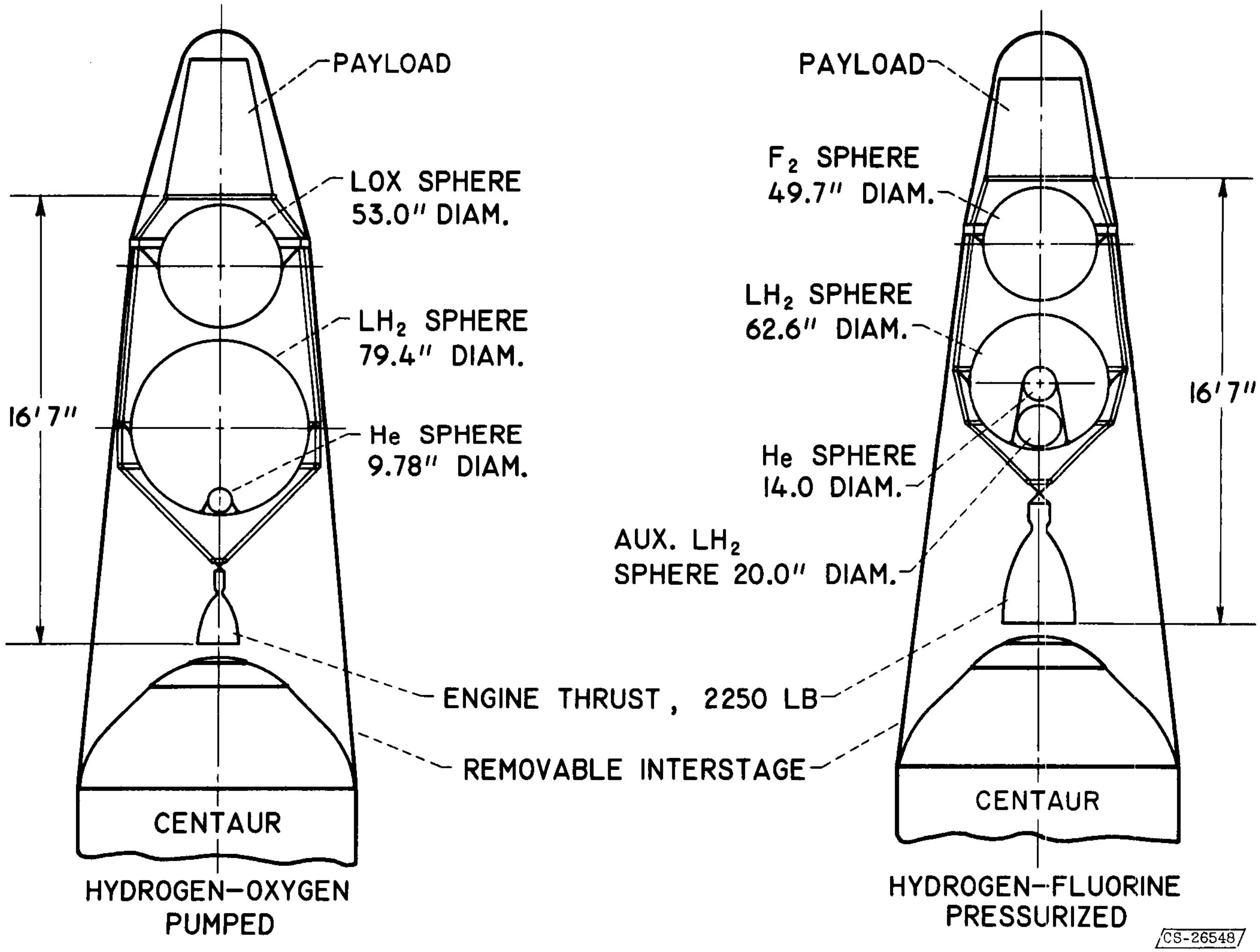

Figure 2

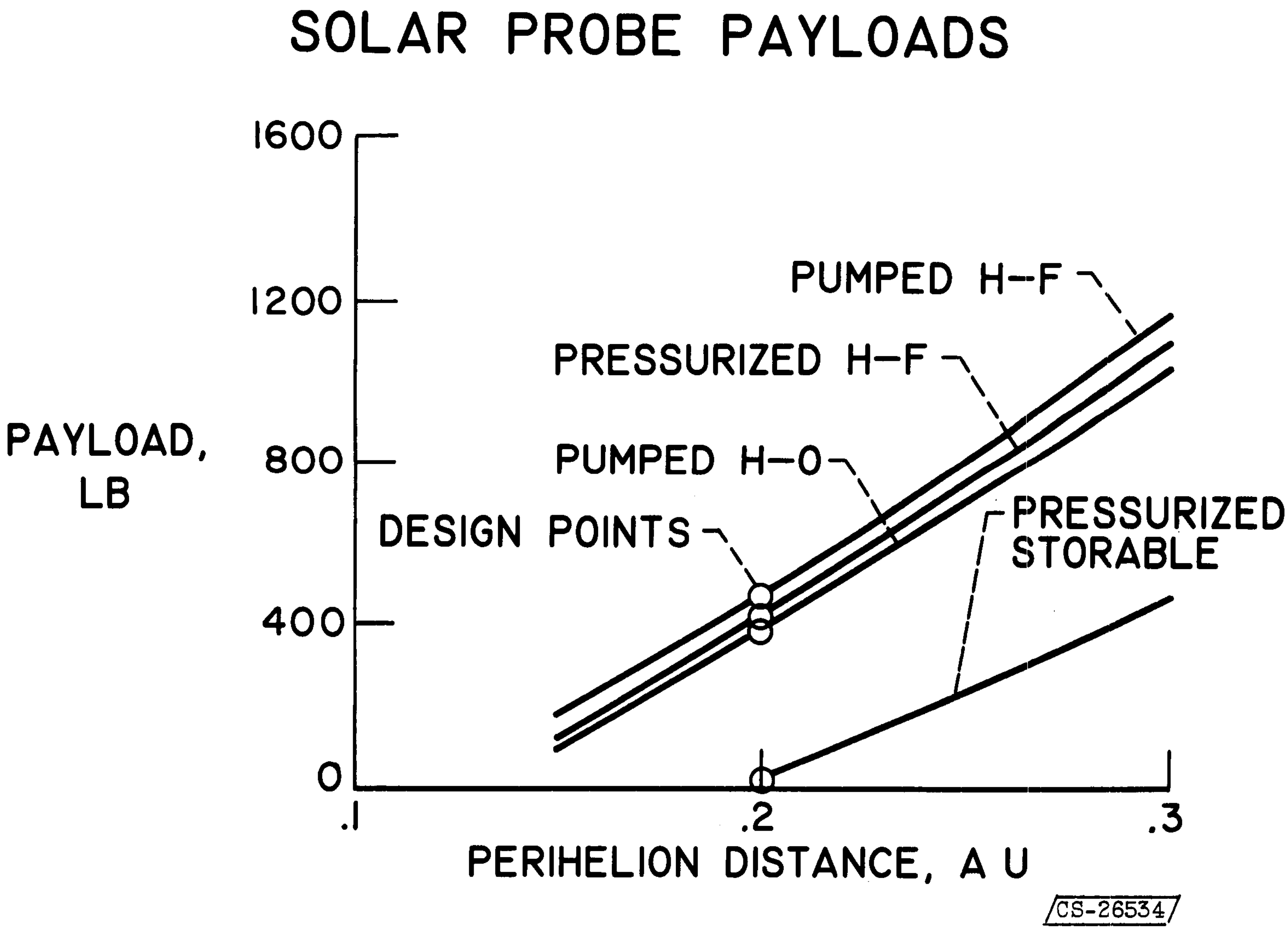
SOLAR PROBE PAYLOADS
PAYLOAD, LB
1600
1200
800
400
0
.1
.2
.3
PERIHELION DISTANCE, A U
PUMPED H–F
PRESSURIZED H–F
PUMPED H–O
DESIGN POINTS
PRESSURIZED STORABLE
CS-26534

Figure 3

MARS ORBIT PAYLOADS FOR VARIOUS INSERTION STAGES

WHIPPLE METEOROID FLUX
BOOSTER–C–IB AND CENTAUR
TRIP TIME, 220 DAYS
INSERTION ΔV, 7400 FT/SEC

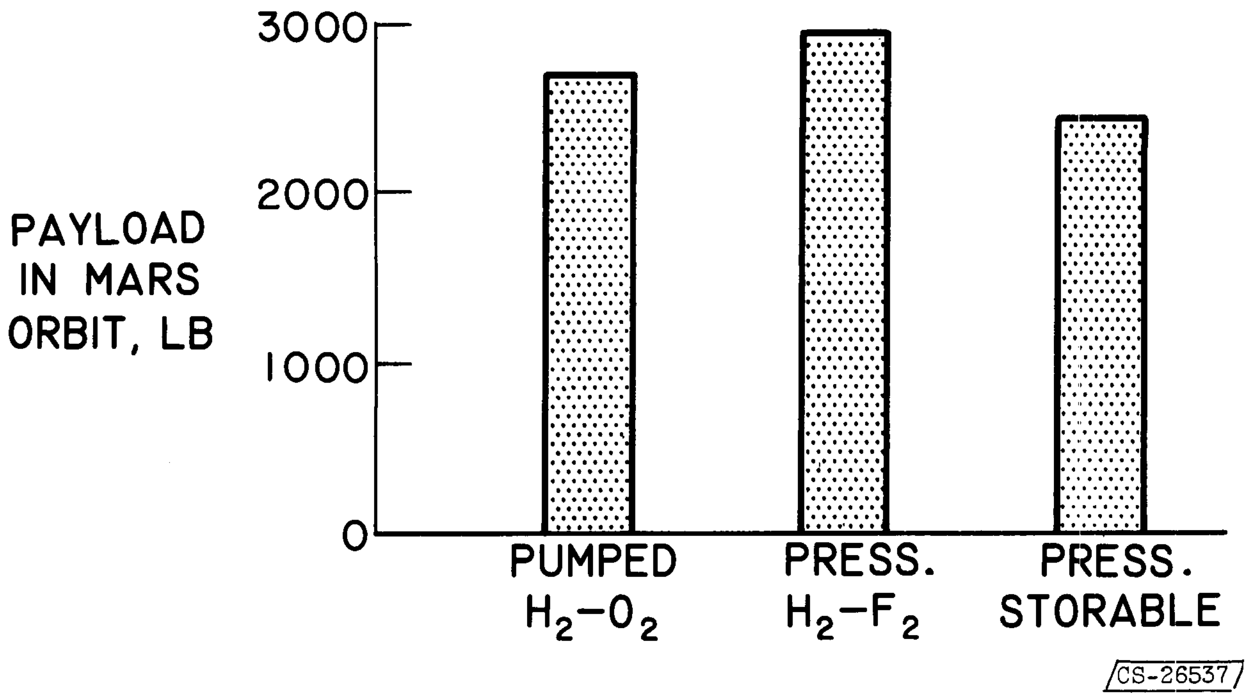

Figure 4

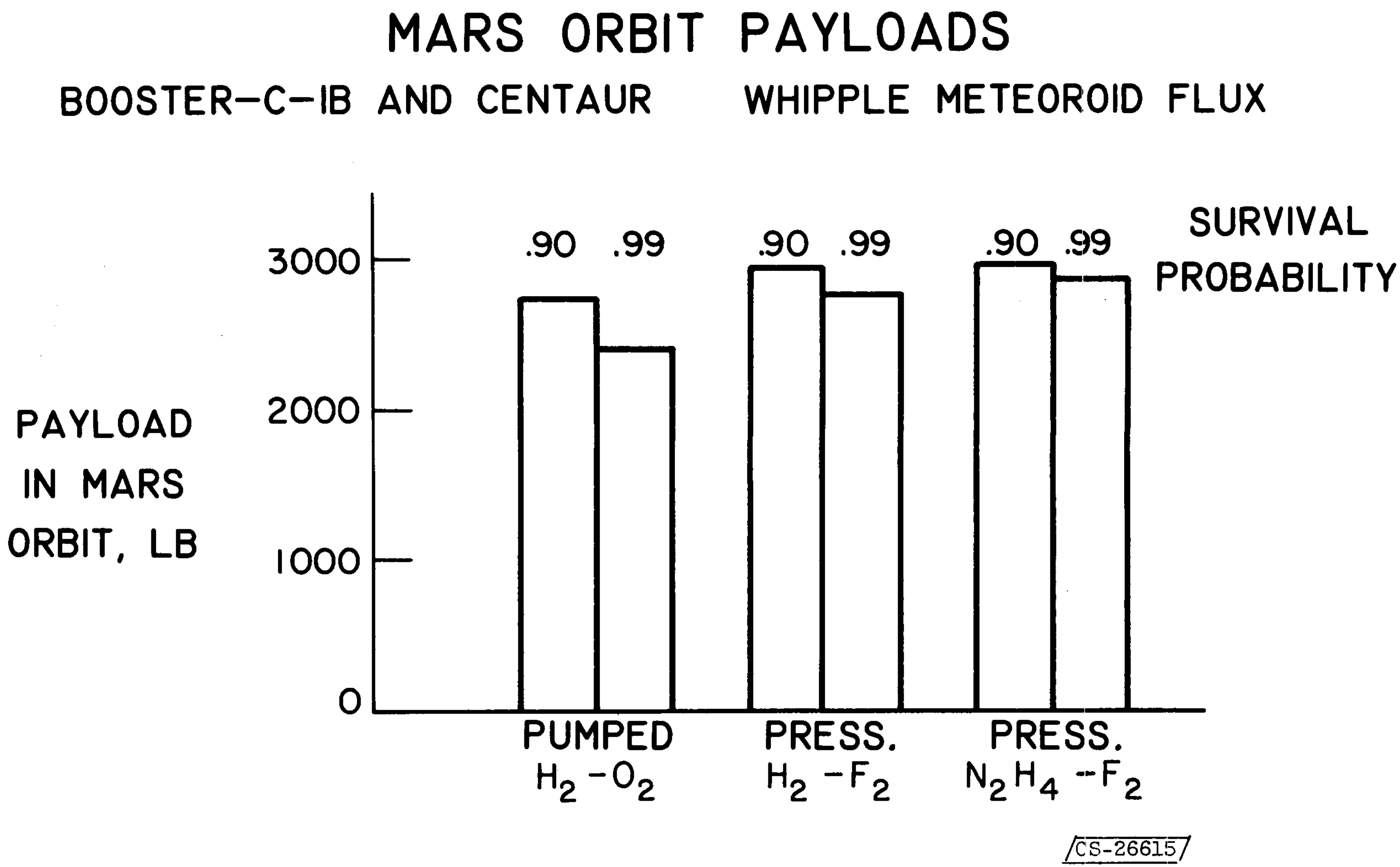
MARS ORBIT PAYLOADS
BOOSTER—C—IB AND CENTAUR
WHIPPLE METEOROID FLUX
.90
.99
.90
.99
.90
.99
SURVIVAL PROBABILITY
PAYLOAD IN MARS ORBIT, LB
3000
2000
1000
0
PUMPED H_2-O_2
PRESS. H_2-F_2
PRESS. $N_2H_4-F_2$
CS-26615

Figure 5

CRYOGENIC SERVICE MODULE

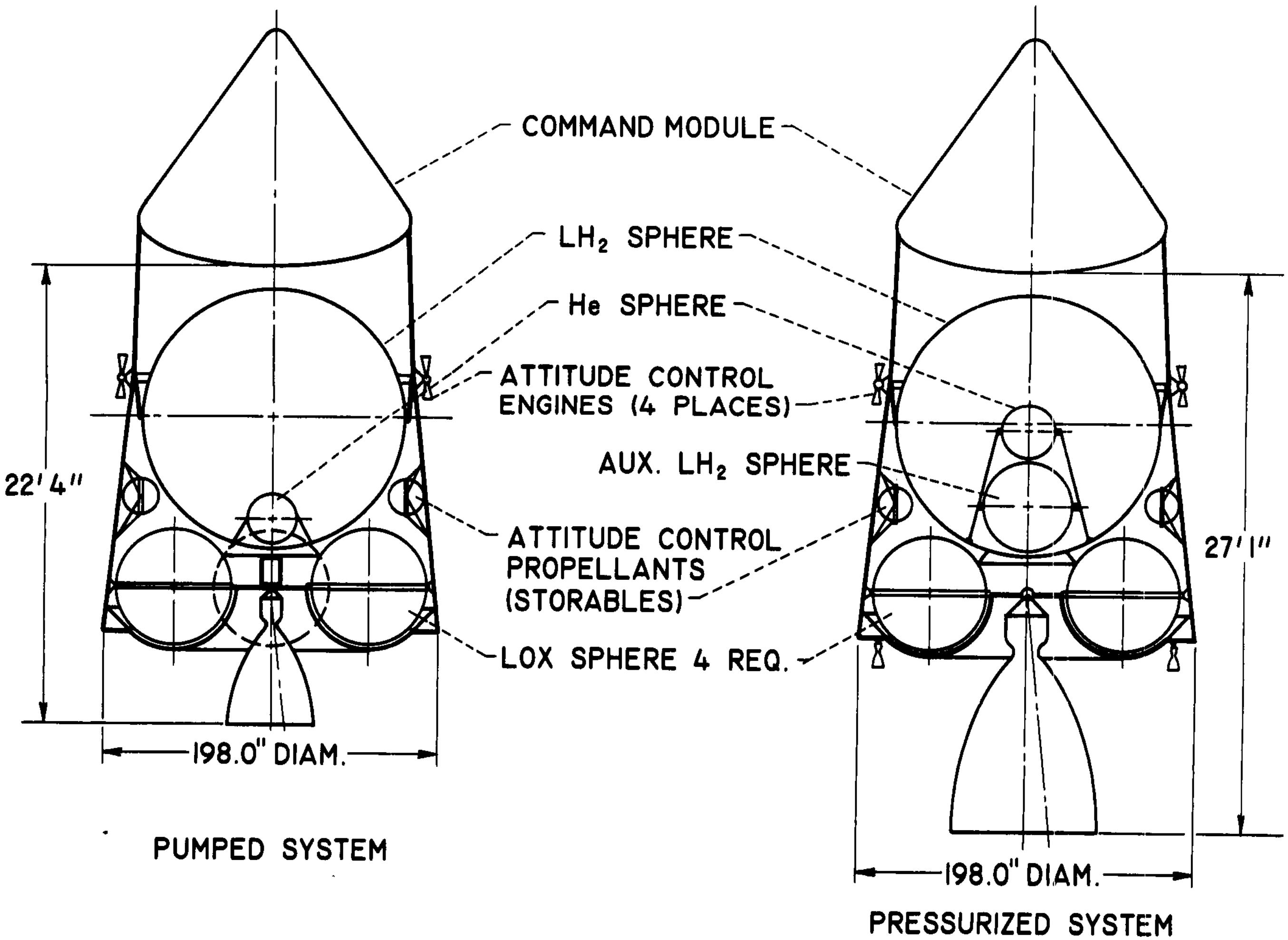

Figure 6

CS-26549

EFFECT OF PROPULSION SYSTEM ON LUNAR SERVICE MODULE PAYLOAD CAPABILITY

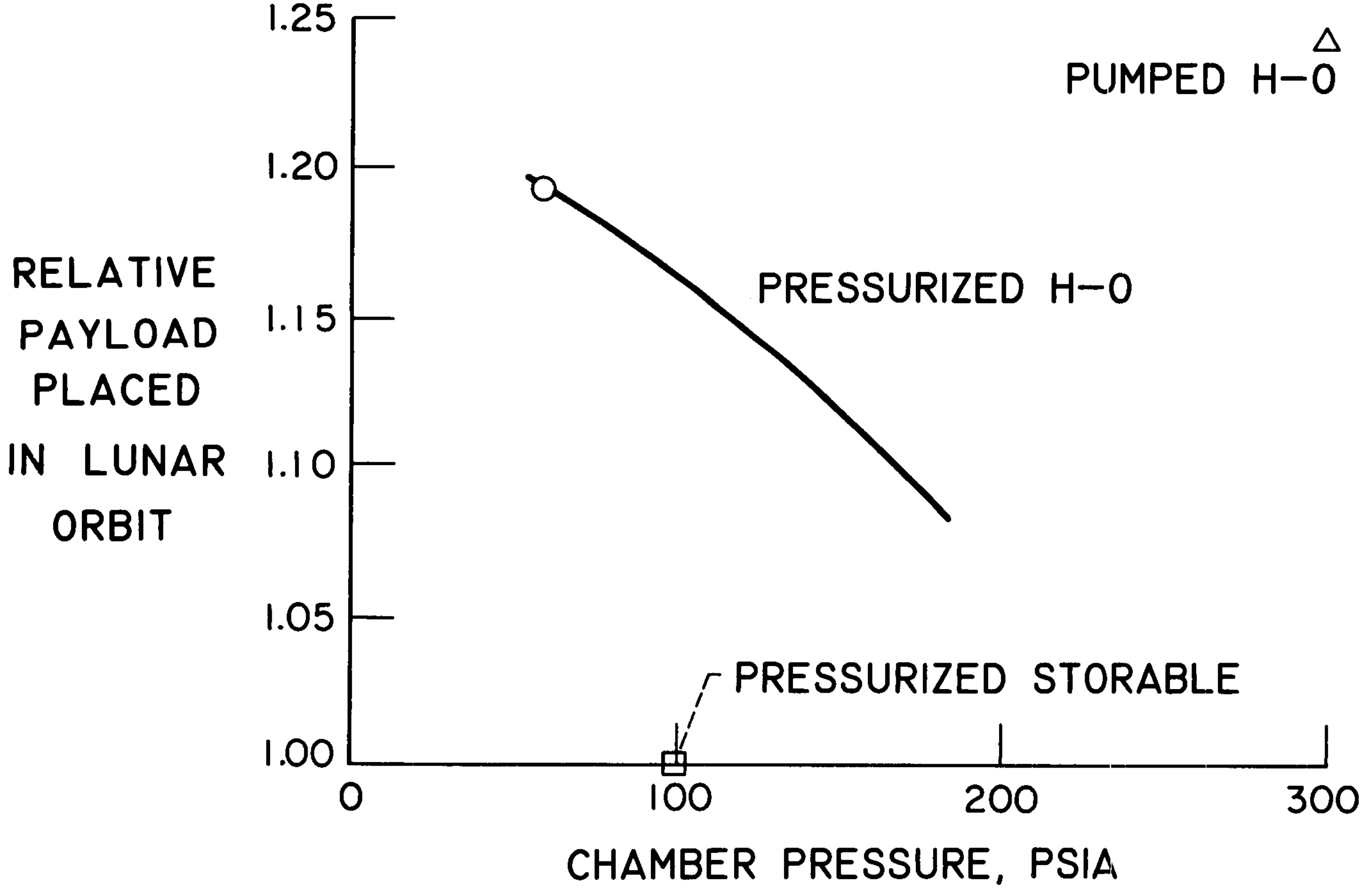

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Figure 7

THEORETICAL DURATION CAPABILITY OF ABLATION CHAMBERS

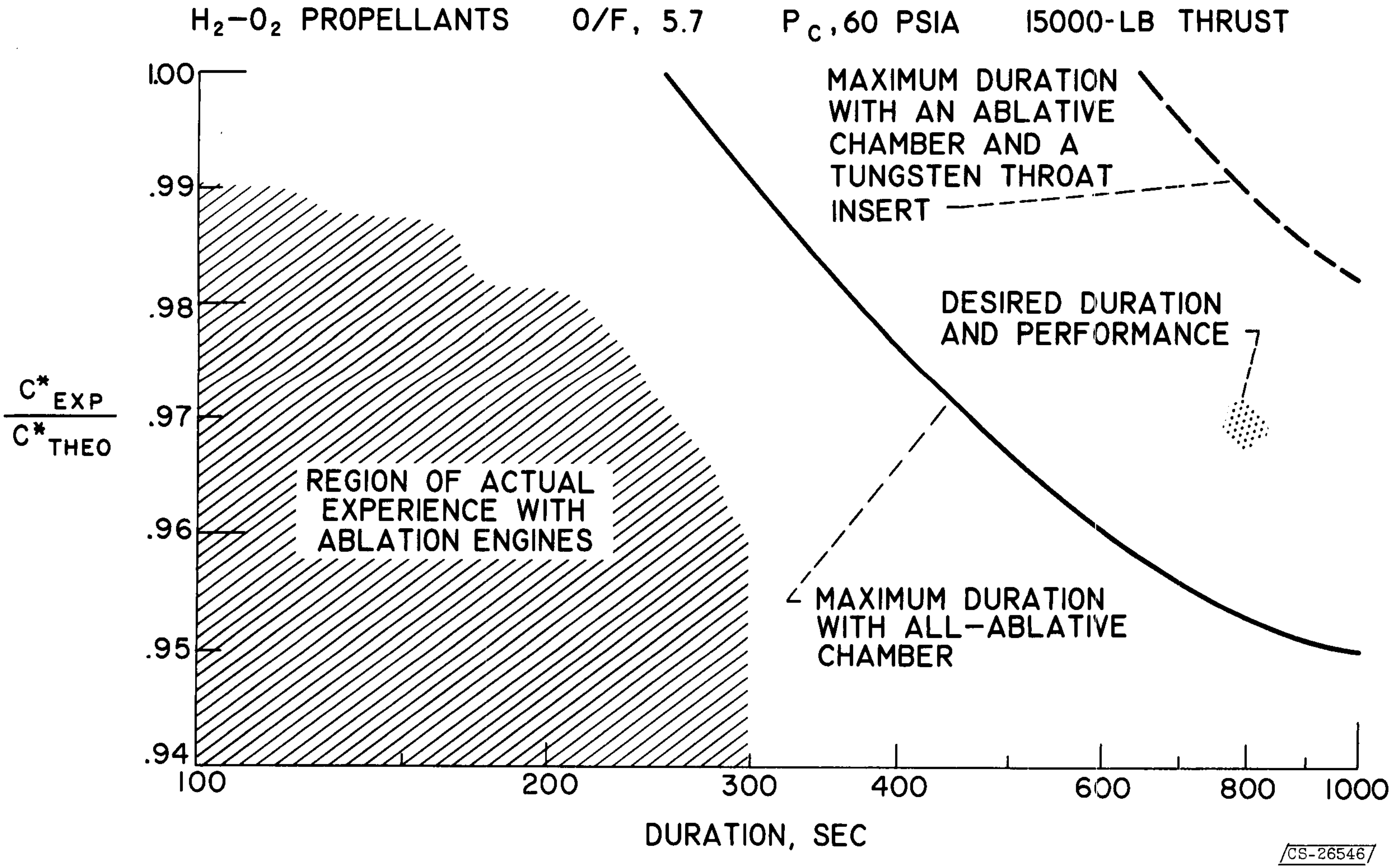
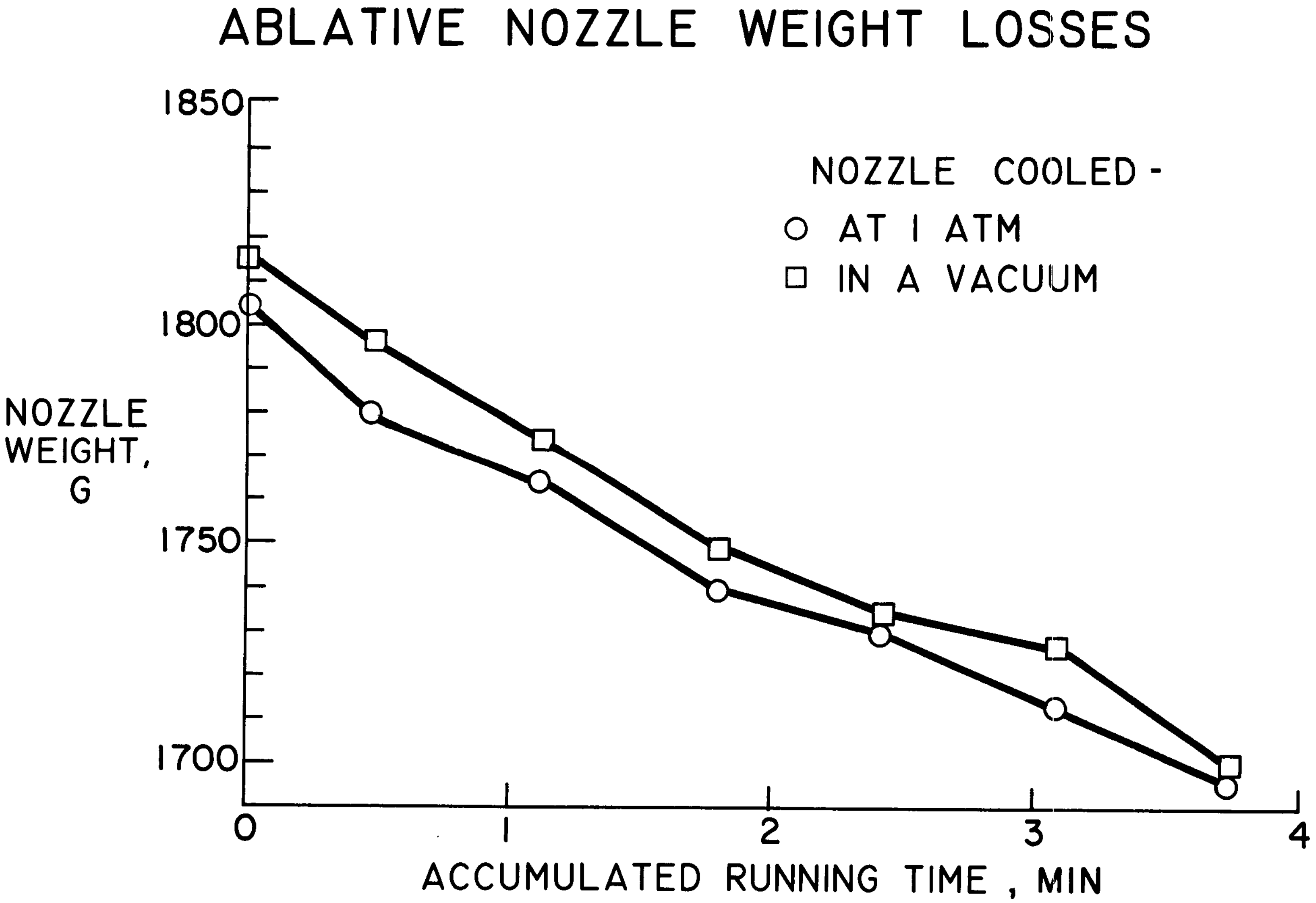


Figure 8



CS-25673

Figure 9

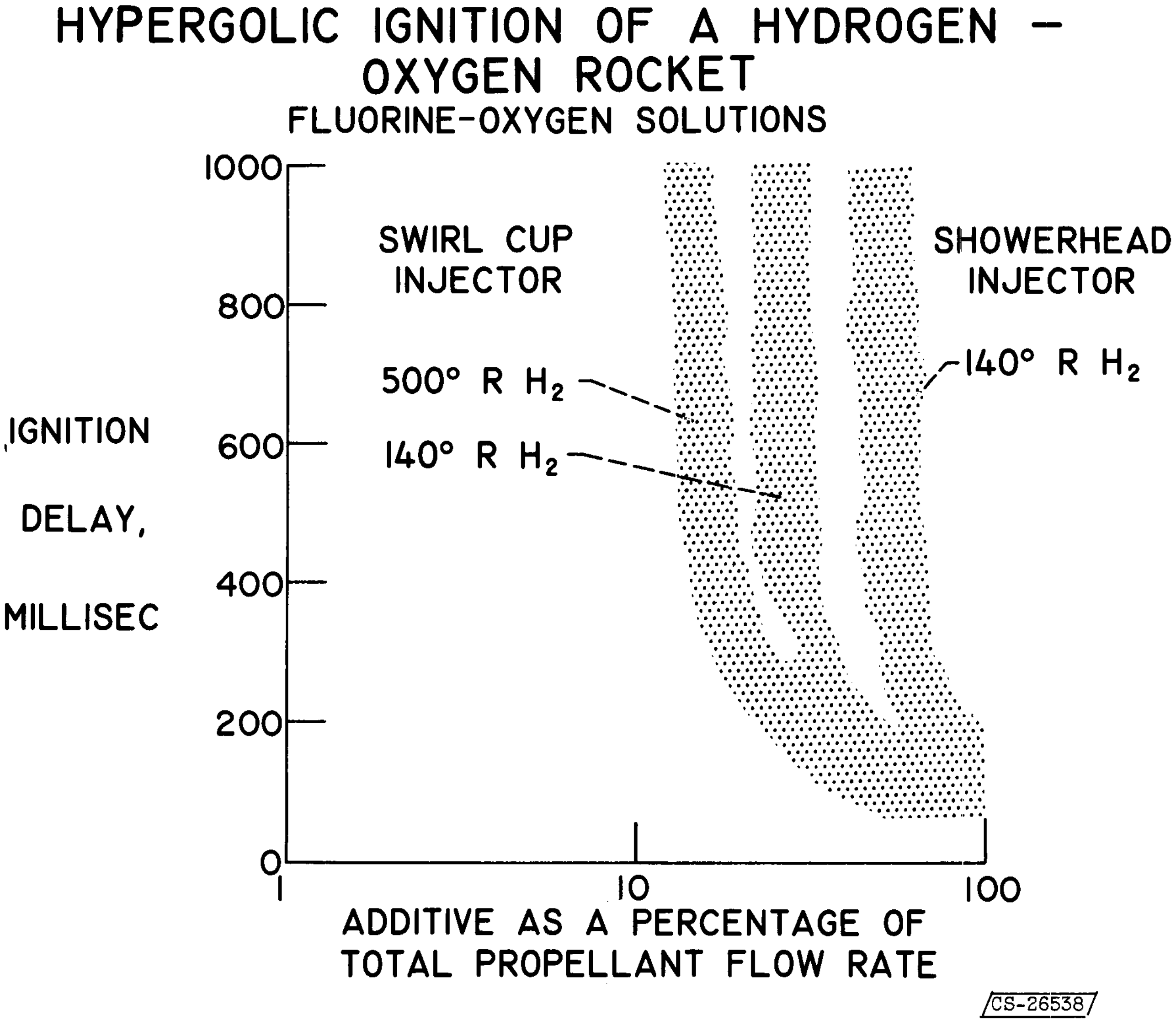

HYPERGOLIC IGNITION OF A HYDROGEN - OXYGEN ROCKET
FLUORINE-OXYGEN SOLUTIONS
SWIRL CUP INJECTOR
SHOWERHEAD INJECTOR
500° R H_2
140° R H_2
140° R H_2
IGNITION DELAY, MILLISEC
1000
800
600
400
200
0
1
10
100
ADDITIVE AS A PERCENTAGE OF TOTAL PROPELLANT FLOW RATE
CS-26538

Figure 10

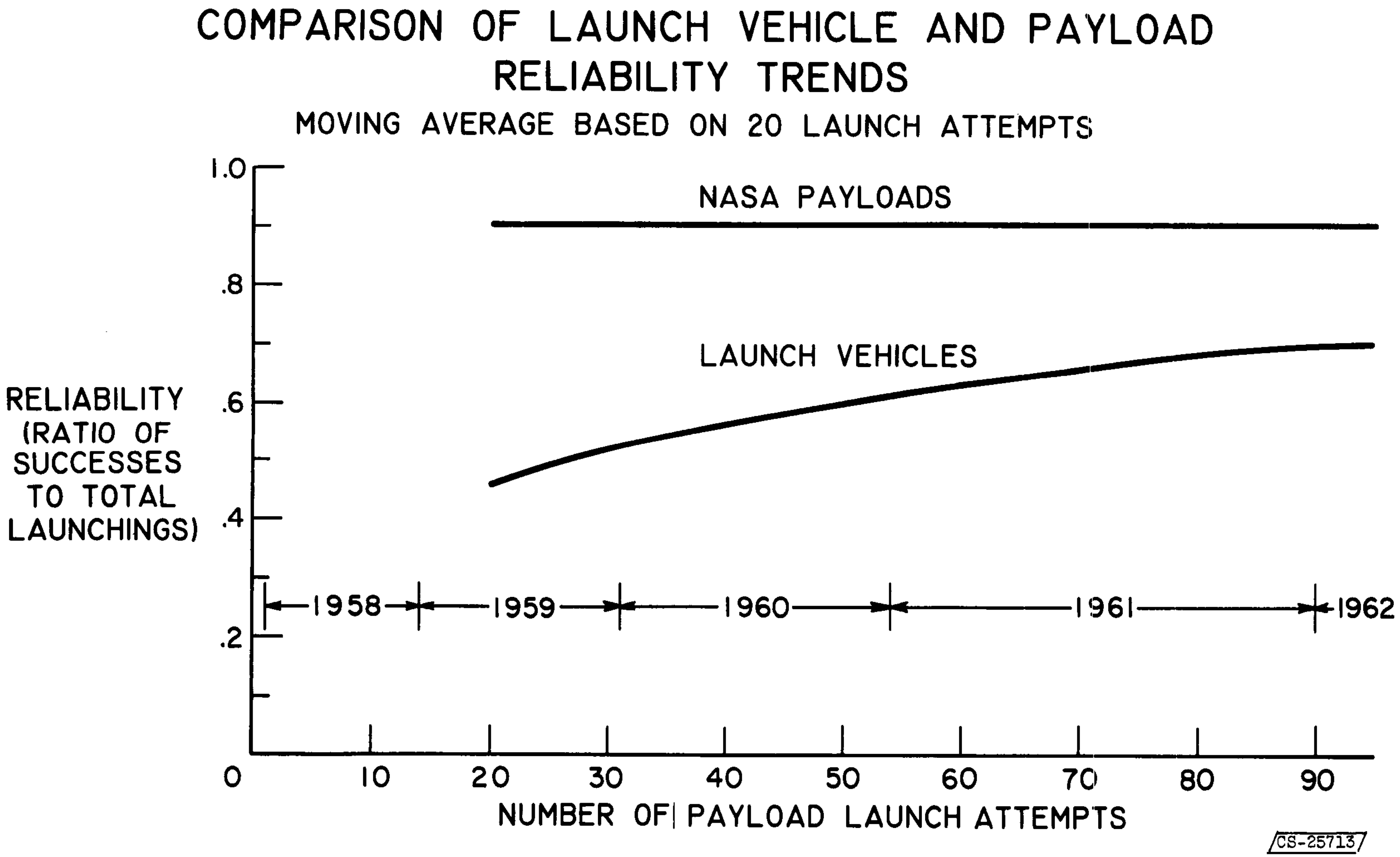

COMPARISON OF LAUNCH VEHICLE AND PAYLOAD RELIABILITY TRENDS
MOVING AVERAGE BASED ON 20 LAUNCH ATTEMPTS
NASA PAYLOADS
LAUNCH VEHICLES
RELIABILITY (RATIO OF SUCCESSES TO TOTAL LAUNCHINGS)
1.0
.8
.6
.4
.2
1958
1959
1960
1961
1962
0
10
20
30
40
50
60
70
80
90
NUMBER OF PAYLOAD LAUNCH ATTEMPTS
CS-25713

Figure 11