Table 5 Results of payload "optimization"

Expt. or Inst.	Run number 1	2	3	4	5	6	7
Experiment value judgments, X_i							
1[b]	4	2	4	4	4	4	4
2	3	4	2	1	0	2	1
3	2	4	2	1	0	2	1
4	3	2	4	4	4	4	4
5	1	1	1	1	1	1	1
6	1	1	1	1	1	1	1
Instrument ranking in descending order, α							
1[b]	10	4	10	10	10	10	10
2	9	3	9	9	9	9	9
3	1	2	12	12	12	12	12
4	12	15	1	1	1	1	1
5	11	10	11	11	11	11	11
6	8	9	8	8	8	8	8
7	4	12	5	5	5	4	5
8	3	11	4	4	7	3	4
9	2	1	3	7	4	5	3
10	15	8	2	3	3	2	7
11	5	7	15	2	6	15	2
12	7	6	7	15	2	7	15
13	6	5	6	6	15	6	6
14	13	13	13	13	13	13	13
15	16	16	16	16	16	16	16
16	14	14	14	14	14	14	14
Cumulative wt, lb, necessary to include experiments with preceding ranking							
1[a]	88	285	88	88	88	88	88
4	289	301	92	92	92	137	92
2	258	275[c]	263	268	161	149	154
3	285	197	289	295	187	175	181
5	301	297	301	301	148	187	187
6	321	321	321	321	207	207	207

[a] Experiment number. [b] Instrument number. [c] 170 without TV camera.

(experiment) is 1700(197), and the spacecraft total is 2400; for run numbers 5–7, the Orbiter (experiment) is 700(124), the Lander (experiment) is 1000(83), and the spacecraft total is 1700.

Optimization of Probe Weight vs Launch Vehicle and Spacecraft Cost

The design value for Earth departure hyperbolic excess speed $V_{\infty\oplus}$ equal to 3 km/sec requires a ΔV of approximately 11.58 km/sec at perigee of the geocentric escape path. For this ΔV, three possible launch vehicle candidates are given in Table 6.

The third column in Table 6 is based on a design value of $\Delta V = 2.18$ km/sec for the Mars orbital capture maneuver. A stage mass ratio for the capture retropropulsion unit of $\lambda = 0.72$ and an $I_{sp} = 310$ sec have been assumed.

It is important to observe that the present booster inventory does not include a continuum of launch vehicle capabilities; a specific capability is associated with each launch vehicle for the trajectory chosen. TAT/40K/7K can deliver the orbiter alone in either runs 1–4 or 5–7, and then Titan III C can deliver the lander in runs 5–7. Or, Titan III C/40K can simultaneously deliver *both* orbiter and lander in all the runs. This would, of course, require increased cost to develop a 40K upper stage for the Titan III C. In cases of experiment deletions, a cumulative mission value less than unity is achieved. The particular weighting (run) of Table 5 will suggest the least valuable experiments for deletion.

The decrease in mission value must be compared with the cost saving, if any, through use of smaller boosters and spacecraft. This must be done in the context of the entire space probe program.[5] This is necessary because the cost of adding one more of a given booster depends on the total number of such boosters used in the over-all space program because of amortization of fixed costs and decrease of unit costs due to manufacturing learning curves. The ratio of value-to-cost is to be optimized.

Table 6 Payload capabilities (lb) of three launch vehicles

Launch vehicle	Spacecraft gross weight leaving Earth orbit	Approx. weight in circular Mars orbit
TAT/40K/7K (thrust augmented Thor plus stages)	2600	740
Titan III C	4000	1140
Titan III C/40K	8500	2430

The foregoing points are often overlooked by analysts when they "optimize" around a single mission. Unless vehicles are selected for a mission in the context of all missions to be performed in a program, one could project a need for a large collection of optimum launch vehicles, each requiring a separate booster research and development effort, at great total cost.

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Gemini Rendezvous

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THE NASA Gemini spacecraft is a two-man spacecraft designed for extensive mission capabilities during Earth orbital flights. One of the greatest challenges during the design and development of this spacecraft was in providing the capability to rendezvous and dock with another orbital craft. This note describes concepts and equipment that will perform these tasks, with emphasis on the practical application to space rendezvous as opposed to the theoretical basis of system design.

Rendezvous missions begin with the target vehicle's launch into a nearly circular orbit from Cape Kennedy. The

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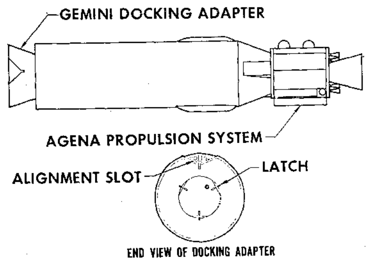

Fig. 1 Agena target vehicle.

characteristics of the target orbit and the requirements for the time of launch and insertion accuracy of the subsequent Gemini launch are very important, and a fairly complete discussion of this subject may be found in Ref. 1. For the first Gemini-Agena rendezvous mission, the Gemini will nominally be launched into a coplanar orbit with a trailing displacement at insertion on the order of 1000 naut miles. The Gemini orbit altitude will be lower than that of the target, thereby producing a natural catch-up rate that allows the crew about three orbits for making orbital maneuvers before the beginning of the rendezvous sequence. These orbital maneuvers are designed to correct for insertion dispersions and minor launch delays as well as to adjust the catch-up rate, so that the desired target lighting conditions will be achieved at rendezvous. Even in the absence of any launch dispersions there will be midcourse maneuvers, so that the resulting Gemini orbit is nearly circular and has a constant differential altitude of about 15 naut miles below the target. It is from this orbital configuration that the rendezvous sequence will be initiated.

The target for Gemini rendezvous is an Agena D vehicle modified with a docking adapter to provide Gemini with a highly cooperative target. The Agena vehicle (Fig. 1) has attitude and maneuvering control systems that can be operated via radar command link from the Gemini spacecraft or by UHF command link from ground. The inertial reference package and horizon sensors provide both inertial and local vertical attitude references. The docking adapter provides the capability for mechanically mating with the Gemini and, in addition, provides enhancement of both visual and radar observations through lighting and the use of a radar transponder. The docking cone, which is attached to the body of the adapter by an arrangement of shock absorbers and springs, has a V shaped slot that forces latch alignment as the Gemini nose enters the cone. This leaves the Gemini and Agena connected through a flexible linkage which is then rigidized by a powered device to complete docking. This mechanical design was one of several that were considered for the docking tasks and was selected over other, less direct designs such as trapezes only after impact velocities were determined to be small.

Optical enhancement of the target is provided at large ranges by the Agena acquisition lights. It was determined that visibility at ranges up to about 20 naut miles was needed for visual rendezvous guidance, and the acquisition lights were sized to provide an intensity equivalent to a third magnitude star at this range. This intensity was chosen as a result of Mercury experiments that have shown that dimmer sources are not consistently visible. The flashing rate of 65 flashes/min was chosen after some tests showed that significantly lower rates make the target difficult to track, whereas higher rates require greater electrical power. At short ranges, visibility is aided by submerged floodlights, which illuminate the docking cone, and by phosphorescent markings on the cone and vehicle body.

Radar enhancement comes from the use of a transponder, which receives signals from the Gemini transmitter and replies with a greatly amplified return. A combination of a dipole and two spiral antennae provides nearly spherical coverage, which permits approaches to the target from a wide range of directions.

Figure 2 shows the aspects of the Gemini spacecraft that are pertinent to the rendezvous mission. The rendezvous radar system is an interferometer type, and when used in conjunction with the target transponder, it provides range and angle data to both the digital computer and the astronauts. The Inertial Guidance System (IGS), composed basically of an inertial platform and a digital computer, provides a stable coordinate reference and the computational means to process the radar data through the guidance equations. The attitude and maneuvering thrusters are located in the equipment adapter and the retrograde adapter modules, which are jettisoned prior to re-entry. The attitude thrusters are arranged so that pairs fire together to create a pitch, yaw, or roll moment. Also, thrust can be applied in either direction along all three body axes by means of the maneuvering thrusters.

The principal guidance scheme uses the radar and the IGS to measure the required parameters and to compute the necessary velocity change to cause rendezvous at a specific time in the future. The equations used are the linearized relative motion equations that are fairly common in the literature.[2] For this closed-loop rendezvous guidance scheme, the radar measures range and relative bearing to the target, and this information is transferred through the inertial platform attitude angles into computer coordinates. The inertial platform for Gemini may be used in either a free inertial mode or torqued to maintain a local vertical reference. Since the spacecraft computer uses local vertical as one of the computational coordinates, the platform is maintained in the latter mode during rendezvous to simplify computations. The maximum platform accuracy for this mode of operation is achieved through the use of gyrocompassing with horizon sensors providing local vertical information. However, this restricts the spacecraft attitudes to a narrow band and would force the radar to track the target at a large look angle. Tracking errors increase at these large angles, and for some approach trajectories, tracking would be lost. An open-loop torquing signal to the platform is used to cause it to follow local vertical and thereby allow the astronaut to maintain a spacecraft attitude for favorable radar (and visual) tracking. System errors will then increase with time if the open-loop input is not exact but the magnitude is small, and the astronaut has the option of returning to the horizon sensor reference for periodic alignments.

Another problem associated with the closed-loop guidance scheme results from the fact that the closed-form solution for the required rendezvous maneuver was derived with the assumption of impulsive velocity changes. With an acceleration level of about 1 ft/sec^2, it is obvious that a velocity change of about 50 fps cannot be accomplished and still satisfy the assumption. Examination of the equations showed that any appreciable compensation in that area effectively defeats the simplification provided by the closed-form solution. The solution to the problem was achieved by a relatively simple extrapolation of the equations to determine the velocity impulse at a future time and then, with a

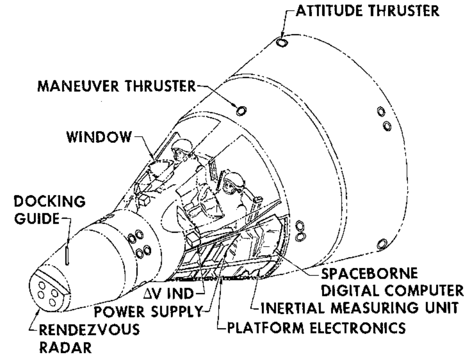

Fig 2 Gemini spacecraft.

knowledge of available accelerations, by the thrust starting one-half of the thrusting period prior to the computed time.

An area of closed-loop rendezvous, which received significant design effort, involves the processing of input data for noise and error rejection. Details of this effort are beyond the scope of this note, but the subject is mentioned because it represents a major engineering effort to achieve a practical noise rejection system. Although there are excellent theoretical optimizations available from the literature, direct application was difficult because of such constraints as limited capacity in the digital computer for data processing and operational requirements that present timing needs contrary to these dictated by filter optimization. The problem is further complicated because velocity information is derived from position measurements, and the error sensitivity becomes a function of the specific trajectory involved. Although optimization programs may be possible, which describe this type constraint mathematically, the solution considered most expeditious for Gemini rendezvous was basically a cut-and-try application of fundamental filter principles to achieve acceptable performance.

A second rendezvous scheme planned for use in Gemini missions is the semioptical technique. This method requires the flight crew to maneuver the spacecraft based on range and range rate information from the radar and visual observations of target motion with respect to star patterns. Until recently, the semioptical method was one of maneuvering laterally to drive the apparent target motion to zero relative to the star background and, at the same time, maintaining a closing range rate. Although this scheme will still be used at small ranges (3 naut miles to a few hundred feet), recent simulations have shown hat guidance to a nominal transfer trajectory is considerably more efficient when guidance must be initiated at large ranges. In the case of the first Gemini-Agena rendezvous mission, the distance of closest approach is about 15 naut miles if no rendezvous maneuvers are made; from this trajectory, semioptical rendezvous by nulling the line-of-sight angular rate is very costly. Guidance to a nominal transfer trajectory is achieved by monitoring range rate as a function of range and adjusting the rate when it deviates markedly from the nominal value at that range. Fuel values for this scheme approach the low expenditure required for the closed-loop method.

The final few hundred feet of the rendezvous mission and the actual docking maneuvers are strictly manually controlled. The pilots, based on viewing the target through the Gemini window, control both the attitude and maneuvering systems of the spacecraft to accomplish docking with the target. This phase of the mission, involving a controlled joining of multiton spacecraft, has introduced several design features to Gemini. The maneuvering control system provides six-directional thrust so that attitude changes are not required for performing speed or directional changes. Hence, the pilots are never required to lose sight of the target. The thrust levels for maneuvering are set at an intermediate value between the desired high thrust for orbit changes and the desired low thrust for docking. These design requirements were confirmed, again using simulation to introduce the human aspects of the system. Conversely, other proposed design features were eliminated when simulation showed the docking task could be accomplished without added complication. It is interesting to note that, although simulations of increasing sophistication were developed during the program, the demonstrated performance remained fairly constant. Apparently, the added cues, provided with more realistic simulations, compensate for the additional disturbing factors and/or system limitations. From the very first simulations, where meter presentations of range were used, until the "ultimate" simulator, which provides full-scale models with six degrees of freedom, velocities at contact have been fractions of a foot per second, and displacement errors have been only a few inches.

During the course of the Gemini program, it is fully expected that new and improved rendezvous techniques will be demonstrated. It should be possible to readily incorporate these new techniques in view of the basic flexibility of the radar and IGS equipment. For the first rendezvous missions, however, the spacecraft will use the techniques described herein. These have been thoroughly examined and, as described in the preceding text, molded to the capabilities of the Gemini equipment and the mission requirements. It is apparent that the optimization of the system was not pursued, and, in fact, the term becomes scarcely definable as opposed to the practical improvement modifications that were necessary to mechanize the theory. Analysis and simulation of both the closed-loop and semioptical rendezvous schemes have demonstrated their feasibility, and the first Gemini rendezvous mission is anticipated with high confidence of its success.

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On Secondary Gas Injection in Supersonic Nozzles

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Nomenclature

A_s = throat area of injection slot, in.^{2}
A_t = throat area of primary nozzle, in.^{2}
AK = amplification factor $= F_p \dot{W}_s / F_s \dot{W}_p$
F = thrust, lb
M = Mach number
P = pressure, psia (unless indicated otherwise)
s = injection slot width
T = temperature, °R
$\dot{W}$ = weight flow rate, lb/sec
γ = ratio of specific heats, c_p/c_v
ϵ = angle of injection measured upstream from a normal to the nozzle axis, deg
$\mathcal{M}$ = molecular weight, lb/mole

Subscripts

0 = stagnation condition
p = primary stream property
s = secondary stream property

DURING the past few years, several investigations have been reported[1–7] on the interaction of a jet of gas with a supersonic stream for the purpose of thrust-vector control. The principal parameters of influence in such an interaction are the physical and flow properties of the main (primary) stream and the injected (secondary) stream, the characteristics of the injection slot, the mass flow rate of the secondary stream, and the orientation of the injected jet with respect to the primary stream. If an amplification factor AK is defined as the ratio of the effective specific impulse of the

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