Sparrow in the Falcon’s Nest

- “Born as a civilian sparrow in a nest of warbirds, NASA grew up and flew.”
- Why have a civilian space agency?
- Infighting within DoD: Army, USAF, OSD, ARPA
  - Favored pliant NACA model for NASA
- Finding a suitable administrator
  - T. Keith Glennan
- ABMA clearly the leading agency for large rocket development
- USAF sought much expanded role
- NASA raid on military capabilities and facilities
- Critical role of ARPA
- Military benefits as side effects of civilian space program
Sparrow in the Falcon’s Nest

- Need for two space programs with distinct objectives
- NASA and USAF the winners
- Army and Navy the losers
- Problematical relationship: Who owns **manned** space flight?
- Reasons for Eisenhower’s posture
- Soviet space program largely military
- Marxism vs. capitalism
- Prestige in third world
- Traditional and modern conservatives
- NASA’s budget: large or small?
- Justification for space program: science or space race?
- Embracing technocracy

**TIROS, Transit, and Echo**
Sparrow in the Falcon’s Nest

- Development of the F-1 engine
- Influence of “committee of outsiders”
- Vice President Nixon’s input
- *Newsweek*: “How to Lose the Space Race!”
  1) Start Late
  2) Downgrade Russian Feats
  3) Fragment Authority
  4) Pinch Pennies
  5) Think Small
  6) Shirk Decisions
- Inaccuracy of *New York Times* reportage
- NSC-5918, Jan 12, 1960, “U.S. Policy on Outer Space”
- Laymen viewed “true conquest of space” as **manned spaceflight**

Sparrow in the Falcon’s Nest

- International cooperation: help or hindrance?
- Price of cooperation: bring technology, knowledge, and funding to the table
- Global tracking networks, international relations
- Goodwill and positive image
- British interests; French interests
- Role for the United Nations
- Views of Eisenhower:
  - Tight-fisted
  - However, he secured NASA’s role in growing technocratic enterprise
  - Root principles, and dualities of space policies
- Ironies behind decisions
- “Honest” space policy: good or bad?
The Shape of Things to Come

- Jefferson: State and society natural adversaries
- Eisenhower Republicans: Government intervention as a necessary evil
- Democrats: Stevenson doves and Symington hawks
- Warfare had become politicized and democratized
- Nuclear weapons changed the nature of warfare forever – did they?
- “Mutual assured destruction” vs. survivable nuclear war
- Proliferation of “Think Tanks”
- Space, a “new frontier” for strategy

The Shape of Things to Come

- “Intensified exploration of brainpower”
  - Seaborg Report
- Consensus among many on need for vastly increased federal spending and power
- Ike didn’t buy it
- Impacts of the 1960 Presidential election
- U-2 flights found no deployed Soviet ICBMs
- SAC and CIA still provided defensive wall
- Reconnaissance satellites about to be launched in profusion
- Project Mercury as a “stop-gap measure” before Apollo?
  - Hardly!
MiDAS, Discoverer, and SAMOS

The Shape of Things to Come

- Follow-on to Mercury required F-1 engine (?)
  - Hardly!
  - Project Gemini didn’t need it
- But F-1 engine was critical for Saturn V
- Missile gap myth used to advantage during “critical years”
- After Republicans lost 1960 election, Eisenhower
  - killed the NASA Apollo Program
  - increased funding for USAF spy satellites
- Importance of a non-secret civilian space program
- As Ike wrote his Farewell Address, JFK and staff planned for the New Frontier
Part III Conclusion

- Sputnik made the Cold War “total”
- Its impact was confusing at first
- Important to prevent the military from going “hog wild”
- Still, space policy should shield spy satellites and other military systems
- Open space policy, real but disingenuous in part
- Flight to the Moon seen as a self-justifying feat
- Big technology not inherently un-American, seen as inevitable replacement of individual innovation
- One writer dismissed Ike “as a military hero who revolted against war”
- Edwin Land: “More and more we tend to resemble the Soviets”

Destination Moon

- Kennedy men: "The generation that fought the war...”
- Eisenhower: Little faith in centralized management
- LBJ’s National Aeronautics and Space Council, formed in 1958
- Robert Kerr, oil millionaire
- Project Mercury: Weisner committee's conclusions: crash program to put man in space cannot be justified
- NASA administrators
  - James Webb
- NASA’s focus
- USAF Space Study Committee
Politics and Advice

- USAF expectations
- **Feb 1961**: Webb finds BoB in dark about space programs, returns to JFK in March
- **Mar 1961**: Weisner Report
- Visionary statements by Lloyd Berkner
- JFK reaction to Gagarin orbit, Bay of Pigs (4/61)
- Johnson report, only one possible conclusion
- LBJ then worked on Webb for backup
- Webb-McNamara report
- Health of aerospace industry
- Congress: $20B on command technology for political goal

Maneuvering to Assure All Constituencies Were Onboard

- "Of all those who contributed to the moon decision, the ones farthest in the background were the engineers of Langley and Goddard and Marshall."
- Parallels to Constantine's conversion to Christian church.
- Kennedy years: space policy falling captive to the image makers.
- Apollo was the greatest peace-time commitment by Congress in history.
- At the same time, McNamara was imposing strict management reforms at the Pentagon
  - possibly playing off NASA against the aerospace industry
  - keeping it alive without paying as much from the DoD budget
A Man on the Moon
### Table: Phases and Design Criteria

<table>
<thead>
<tr>
<th>Phase</th>
<th>Initial Event</th>
<th>Design Criteria</th>
</tr>
</thead>
<tbody>
<tr>
<td>Braking</td>
<td>PDI</td>
<td>Minimize propellant usage</td>
</tr>
<tr>
<td>Approach</td>
<td>High Gate</td>
<td>Crew visibility</td>
</tr>
<tr>
<td>Landing</td>
<td>Low Gate</td>
<td>Manual control</td>
</tr>
</tbody>
</table>

### Diagram Description
- **CSM Orbit**: 160 N. Mi.
- **PDI**: 50,000 ft
- **High Gate**: 260 N. Mi.
- **Low Gate**: Landing point

**Braking Phase**: 8 min: 20 sec

**Approach or Visibility Phase**: 11 min: 48 sec

**Terminal Landing Phase**: 11 min: 30 sec

**Start LR Alt Updating**
- Alt: 50,000 ft
- Vel: 3,548 fps
- R₂₀₅₈: 240 nm
- Time: 4 min: 18 sec

**Start LR Alt Updating**
- Alt: 31,200 ft
- Vel: 3,065 fps

**Start LR Velocity Updating**
- Alt: 22,645 ft
- Vel: 1,325 fps
- Time: 6 min: 42 sec

**High-Gate Point**
- Alt: 7,400 ft
- Vel: 434 fps
- R₂₀₅₈: 8.3 nm

**Low-Gate Point**
- Alt: 500 ft
- Vel: 70 fps
- R₂₀₅₈: 8.3 nm

**Braking Phase**: 8 min: 20 sec

**Approach or Visibility Phase**: 11 min: 48 sec

**Terminal Landing Phase**: 11 min: 30 sec
Apollo 11

July 16, 1969

- Powered descent, LM flip at 46 Kft
- Maneuvering jets – Reaction Control System (RCS)
- 1202 Alarm: Rendezvous Radar
- Switch to manual control

July 20, 1969
Cause of 1202 Alarm
Charles “Pete” Conrad, ’51
(1930-1999)

Apollo 13
Apollo 13 Trajectory

Apollo 14
January 1971

Apollo 15
July 1971

Apollo 16
April 1972
Apollo 17, December 1972

Apollo 18-20: Cancelled

Orbit Transfer and Adjustment
Single Impulse Maneuver

- 4-stage launch to orbit, e.g., *Scout*
  - Three stages burn sequentially
  - 3\(^{rd}\)-stage coasts to apogee = perigee of desired orbit
  - 4\(^{th}\)-stage burn @ 3\(^{rd}\)-stage apogee provides orbit insertion

- 4\(^{th}\)-stage burn time is short compared to orbital period
- Assume that final stage provides required $\Delta v$ instantaneously

Assumptions for Impulsive Maneuver

- Instantaneous change in velocity vector
  \[ \mathbf{v}_2 = \mathbf{v}_1 + \Delta \mathbf{v}_{\text{rocket}} \]

- Negligible change in radius vector
  \[ \mathbf{r}_2 = \mathbf{r}_1 \]

- Therefore, new orbit intersects old orbit
- Velocities different at the intersection
Geometry of Impulsive Maneuver

Change in velocity magnitude, $|v|$, vertical flight path angle, $\gamma$, and horizontal flight path angle, $\xi$

Single Impulse Orbit Adjustment

Coplanar (i.e., in-plane) maneuvers

- Change eccentricity
- Change energy
- Change angular momentum

Required velocity increment

\[ e = \sqrt{1 + 2 \frac{E h^2}{\mu^2}} \]
\[ E = (e^2 - 1) \frac{\mu^2}{h^2} = \frac{1}{2} v^2 - \mu/r \]
\[ h = \sqrt{\frac{\mu^2 (e^2 - 1)}{E}} = \sqrt{\frac{\mu^2 (e^2 - 1)}{v^2/2 - \mu/r}} \]

\[ v_{\text{new}} \triangleq v_{\text{old}} + \Delta v_{\text{rocket}} = \sqrt{2 (E + \mu/r)} \]
\[ = \sqrt{2 \left[ (e^2 - 1) \frac{\mu^2}{h^2} + \mu/r \right]} \]
\[ \Delta v_{\text{rocket}} = v_{\text{new}} - v_{\text{old}} \]
Single Impulse Orbit Adjustment
Coplanar (i.e., in-plane) maneuvers

- Change semi-major axis
  - magnitude
  - orientation (i.e., argument of perigee); in-plane isosceles triangle

\[ a = \frac{h^2}{\mu} \]

- Change apogee or perigee
  - radius
  - velocity

\[
\begin{align*}
  r_{\text{perigee}} &= a(1-e) \\
  r_{\text{apogee}} &= a(1+e) \\
  v_{\text{perigee}} &= \sqrt{\frac{\mu}{a} \left( \frac{1+e}{1-e} \right)} \\
  v_{\text{apogee}} &= \sqrt{\frac{\mu}{a} \left( \frac{1-e}{1+e} \right)}
\end{align*}
\]

Orbit Circularization

- Initial orbit is elliptical, with apogee radius equal to desired circular orbit radius

\[
\begin{align*}
  a &= \frac{r_{\text{cir}} + r_{\text{insertion}}}{2} \\
  e &= \frac{r_{\text{cir}} - r_{\text{insertion}}}{2a} \\
  v_{\text{apogee}} &= \sqrt{\frac{\mu}{a} \left( \frac{1-e}{1+e} \right)}
\end{align*}
\]

- Velocity in circular orbit is a function of the radius
  - “Vis viva” equation:

\[
v_{\text{cir}} = \sqrt{\frac{2}{r_{\text{cir}} a_{\text{cir}}} - \frac{1}{a_{\text{cir}}}} = \sqrt{\frac{2}{r_{\text{cir}} a_{\text{cir}}} - \frac{1}{r_{\text{cir}}}} = \sqrt{\frac{\mu}{r_{\text{cir}}}}
\]

- Rocket must provide the difference

\[ \Delta v_{\text{rocket}} = v_{\text{cir}} - v_{\text{apogee}} \]
Single Impulse Orbit Adjustment
Out-of-plane maneuvers

- Change orbital inclination
- Change longitude of the ascending node
- $v_1$, $\Delta v$, and $v_2$ form isosceles triangle perpendicular to the orbital plane to leave in-plane parameters unchanged

Change in Inclination and Longitude of Ascending Node
(from Sellers, Understanding Space)
Escape from a Circular Orbit

- Velocity in circular orbit is a function of the radius
  - “Vis viva” equation:

\[ v_{cir} = \sqrt{\frac{2}{r_{cir}} - \frac{1}{a}} \]

- Minimum escape trajectory shape is a parabola

Escape from a Circular Orbit

- Semi-major axis goes to \( \infty \)
- Total specific energy of the parabolic orbit is \textbf{zero}

\[ E_{par} = -\frac{\mu}{2a} \xrightarrow{a \to \infty} 0 \]

- Eccentricity of the parabolic orbit is \textbf{one}

\[ e = \sqrt{1 + \frac{E_{par} P_{par}}{\mu}} = 1 \]

- Perigee velocity of the parabolic orbit

\[ v_{p_{par}} = \sqrt{\frac{2}{r_{p_{par}}} - \frac{1}{a}} = \sqrt{\frac{2\mu}{r_{p_{par}}}}, \quad a \to \infty \]
Velocity Increment Required for Escape

Velocity increment supplied by the spacecraft

\[ \Delta v_{\text{rocket}} = v_{\text{par}} - v_{\text{cir}} \]

\[ = \sqrt{\frac{2\mu}{r_{\text{cir}}}} - \sqrt{\frac{\mu}{r_{\text{cir}}}} = \left(\sqrt{2} - 1\right)\sqrt{\frac{\mu}{r_{\text{cir}}}} \]

\[ \approx 0.414 \sqrt{\frac{\mu}{r_{\text{cir}}}} \]

Two Impulse Maneuvers

- 1\textsuperscript{st} \( \Delta v \) produces target orbit intersection
- 2\textsuperscript{nd} \( \Delta v \) matches target orbit
- Minimize \( (|\Delta v_1| + |\Delta v_2|) \) to minimize propellant use
- Rendezvous with trailing spacecraft in same orbit
- At perigee, increase speed to increase orbital period
- At future perigee, decrease speed to resume original orbit
Hohmann Transfer between Coplanar Circular Orbits
(Outward transfer example)

Transfer Orbit
\[ a = \frac{r_{\text{cir}_1} + r_{\text{cir}_2}}{2} \]
\[ e = \frac{r_{\text{cir}_2} - r_{\text{cir}_1}}{2a} \]
\[ v_{\text{ptransfer}} = \sqrt{\frac{\mu}{a \left( \frac{1 + e}{1 - e} \right)}} \]
\[ v_{\text{attransfer}} = \sqrt{\frac{\mu}{a \left( \frac{1 - e}{1 + e} \right)}} \]

Outward Transfer Orbit
Velocity Requirements

\[ \Delta v_{\text{at 1st Burn}} = v_{\text{ptransfer}} - v_{\text{cir}_1} \]
\[ = v_{\text{cir}_1} \left( \sqrt{\frac{2r_{\text{cir}_2}}{r_{\text{cir}_1} + r_{\text{cir}_2}}} - 1 \right) \]

\[ \Delta v_{\text{at 2nd Burn}} = v_{\text{cir}_2} - v_{\text{attransfer}} \]
\[ = v_{\text{cir}_2} \left( 1 - \sqrt{\frac{2r_{\text{cir}_2}}{r_{\text{cir}_1} + r_{\text{cir}_2}}} \right) \]

Hohmann Transfer is energy-optimal for transfer between circular orbits

\[ \Delta v_{\text{total}} = v_{\text{cir}_1} \left[ \sqrt{\frac{2r_{\text{cir}_2}}{r_{\text{cir}_1} + r_{\text{cir}_2}} \left( 1 - \frac{r_{\text{cir}_1}}{r_{\text{cir}_2}} \right)} + \frac{r_{\text{cir}_1}}{r_{\text{cir}_2}} - 1 \right] \]
Rendezvous Requires Phasing of the Maneuver
(from Sellers, Understanding Space)

- Transfer orbit time equals target’s time to reach rendezvous point

Interplanetary Travel
Launch Opportunities for Fixed Transit Time: The Synodic Period

- **Synodic Period, $S_n$:** The time between conjunctions
  - $P_A$: Period of Planet A
  - $P_B$: Period of Planet B

- **Conjunction:** Two planets, A and B, in a line or at some fixed angle

$$S_n = \frac{P_A P_B}{P_A - P_B}$$

---

### Launch Opportunities for Fixed Transit Time: The Synodic Period

<table>
<thead>
<tr>
<th>Planet</th>
<th>Synodic Period with respect to Earth, days</th>
<th>Period</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mercury</td>
<td>116</td>
<td>88 days</td>
</tr>
<tr>
<td>Venus</td>
<td>584</td>
<td>225 days</td>
</tr>
<tr>
<td>Earth</td>
<td>-</td>
<td>365 days</td>
</tr>
<tr>
<td>Mars</td>
<td>780</td>
<td>687 days</td>
</tr>
<tr>
<td>Jupiter</td>
<td>399</td>
<td>11.9 yr</td>
</tr>
<tr>
<td>Saturn</td>
<td>378</td>
<td>29.5 yr</td>
</tr>
<tr>
<td>Uranus</td>
<td>370</td>
<td>84 yr</td>
</tr>
<tr>
<td>Neptune</td>
<td>367</td>
<td>165 yr</td>
</tr>
<tr>
<td>Pluto</td>
<td>367</td>
<td>248 yr</td>
</tr>
</tbody>
</table>
Transfer Orbits and Spheres of Influence

- **Sphere of Influence (Laplace):**
  - Radius within which gravitational effects of planet are more significant than those of the Sun
- **Patched-conic section approximation**
  - Sequence of 2-body orbits
  - Outside of planet’s sphere of influence, Sun is the center of attraction
  - Within planet’s sphere of influence, planet is the center of attraction
- **Fly-by trajectories dip into intermediate object’s sphere of influence for gravity assist**

Interplanetary Mission Planning

- **Example: Direct Hohmann Transfer from Earth Orbit to Mars Orbit (No fly-bys)**
  1) Calculate required perigee velocity for transfer orbit - Sun as center of attraction: **Elliptical orbit**
  2) Calculate $\Delta v$ required to reach Earth’s sphere of influence with velocity required for transfer – Earth as center of attraction: **Hyperbolic orbit**
  3) Calculate $\Delta v$ required to enter circular orbit about Mars, given transfer apogee velocity – Mars as center of attraction: **Hyperbolic orbit**
Solar Orbits

- Same equations as used for earth-referenced orbits
  - Dimensions of the orbit
  - Position and velocity of the spacecraft
  - Period of elliptical orbits
  - Different gravitational constant $\mu_{Sun} = 1.3327 \times 10^{11} \text{ km}^3 / \text{s}^2$

Solar System Spheres of Influence

For $\frac{m_{Planet}}{m_{Sun}} \ll 1$, $r_{SI} \approx r_{Planet-Sun} \left( \frac{m_{Planet}}{m_{Sun}} \right)^{2/5}$

<table>
<thead>
<tr>
<th>Planet</th>
<th>Sphere of Influence, km</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mercury</td>
<td>112,000</td>
</tr>
<tr>
<td>Venus</td>
<td>616,000</td>
</tr>
<tr>
<td>Earth</td>
<td>929,000</td>
</tr>
<tr>
<td>Mars</td>
<td>578,000</td>
</tr>
<tr>
<td>Jupiter</td>
<td>48,200,000</td>
</tr>
<tr>
<td>Saturn</td>
<td>54,500,000</td>
</tr>
</tbody>
</table>
**Conic Sections**

<table>
<thead>
<tr>
<th>Orbit Shape</th>
<th>Eccentricity, $e$</th>
<th>Energy, $E$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Circle</td>
<td>0</td>
<td>&lt;0</td>
</tr>
<tr>
<td>Ellipse</td>
<td>$0 &lt; e &lt; 1$</td>
<td>&lt;0</td>
</tr>
<tr>
<td>Parabola</td>
<td>1</td>
<td>0</td>
</tr>
<tr>
<td>Hyperbola</td>
<td>&gt;1</td>
<td>&gt;0</td>
</tr>
</tbody>
</table>

**Hyperbolic Orbits**

<table>
<thead>
<tr>
<th>Orbit Shape</th>
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<tr>
<td>Hyperbola</td>
<td>&gt;1</td>
<td>&gt;0</td>
</tr>
</tbody>
</table>

Hyperbolic Orbits

\[ E = \frac{v^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a}, \quad \therefore a < 0 \]

Velocity remains positive as radius approaches ∞

\[ v_{\to \infty} \rightarrow v_{\infty} \]

\[ \therefore E_{\infty} = \frac{v_{\infty}^2}{2}, \quad \text{and} \quad v_{\infty} = \sqrt{-\frac{\mu}{a}} \quad \text{or} \quad a = -\frac{\mu}{v_{\infty}^2} \]
Hyperbolic Encounter with a Planet

- Trajectory is deflected by target planet’s gravitational field
- Velocity w.r.t. Sun is increased or decreased

Δ: Miss Distance, km
δ: Deflection Angle, deg or rad

Hyperbolic Orbits

Asymptotic Value of True Anomaly

Polar Equation for a Conic Section

\[ r = \frac{p}{1 + e \cos \theta} = \frac{a(1 - e^2)}{1 + e \cos \theta} \]

\[ \cos \theta = \frac{1}{e} \left[ \frac{a(1 - e^2)}{r} - 1 \right] \]

\[ \theta \xrightarrow{r \rightarrow \infty} \theta_\infty \]

\[ \theta_\infty = \cos^{-1}\left( -\frac{1}{e} \right) \]
Hyperbolic Orbits

Angular Momentum

\[ h = \text{Constant} = v_\infty \Delta \]
\[ \mu \rho = \mu a (1 - e^2) = \frac{\mu^2 (e^2 - 1)}{v_\infty^2} \]

Eccentricity

\[ e = \sqrt{1 + \frac{2 h^2 E}{\mu^2}} = \sqrt{1 + \frac{v_\infty^4 \Delta^2}{\mu^2}} \]

Perigee Radius

\[ r_p = a (1 - e) = \frac{\mu}{v_\infty^2} (e - 1) \]

Eccentricity

\[ e = \left[ 1 + \frac{r_p v_\infty^2}{\mu} \right] \]

Effect of Target Planet’s Gravity on Probe’s Sun-Relative Velocity

Deflection – Velocity Reduction
Effect of Target Planet’s Gravity on Probe’s Sun-Relative Velocity

*Deflection – Velocity Addition*

- $\Delta v$ to increase speed to escape velocity
- Velocity required for transfer at sphere of influence
Planet Capture Trajectory

- Hyperbolic approach to planet’s sphere of influence
- $\Delta v$ to decrease speed to circular velocity

Earth-Moon Sphere of Influence

$$r_{SI} = r_{Earth-Moon} \left( \frac{m_{Moon}}{m_{Earth}} \right)^{2/5} \approx 66,100 \text{ km}$$

... but $r_{SI} \sim \frac{1}{4} r_{Earth-Moon}$

Actual “sphere” of influence is not a sphere (Battin, 1964)
Lunar Trajectory

- Direct transfer orbit may be an ellipse wrt Earth
- Travel time reduced for parabolic or hyperbolic transfer
Michielsen
Chart for Lunar Encounter
(Kaplan, Modern Spacecraft Dynamics and Control, 1976, Sec 3.5)

- Orbital velocity of the moon (wrt earth), $v_{\text{Moon}} \approx 1$ km/s
- Probe’s approach velocity specified by transfer trajectory
- Probe’s approach deflection angle specified by moon-relative hyperbolic trajectory
- Vector triangles predict probe’s departure velocity vector
- Transfer time to moon shown
- “Earth intercept zone” connotes return to earth without thrusting maneuver
- Earth escape also possible

Apollo Free-Return Trajectory

- Trajectories to lunar orbit or landing typically pass in front of the moon
- Thrusting maneuver on the far side required for lunar orbit or landing
- With proper approach velocity, trajectory is deflected to Figure 8 pattern for “free return”
Next Time:

- **Space Vehicle Design:** *Understanding Space* Ch 11, Sec 13.3, 13.4
- **Building Spacecraft & Launch Vehicles:**
  - *Chariots for Apollo* Ch 4 to 8
  - *Stages to Saturn*, Ch 3 to 8
  - *Wikipedia* pages for contemporary launch vehicles, e.g., Saturn IB, Saturn V, Space Launch System, Delta 4, Atlas 5, Ariane 5, Falcon 9, Long March 5, and PSLV

Supplemental Material
Mercury MESSENGER
Fly-By Trajectories

Mercury MESSENGER Mission
https://www.youtube.com/watch?v=otF2FjpCyZk

Cassini Fly-by Trajectories
Early Lunar Spacecraft

• Mission
  – Scientific discovery
  – Preparations for human voyages to the moon
• Robotic exploration of the moon

Lunar Spacecraft

• Lunar spacecraft launched by US, ESA, Russia, and Japan
• Future launches planned by China, Germany, and India